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AIRWORTHINESS CRITERIA DEVELOPMENT FOR POWERED-LIFT AIRCRAFT

A Program Summary

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16. Abstract

A four-year simulation program to develop airworthiness criteria for powered-lift aircraft is summarized. All flight phases affected by use of powered lift (approach, landing, takeoff) are treated with regard to airworthiness problem areas (limiting flight conditions and safety margins: stability, control, and performance; and systems failure). A tutorial discussion of each aspect is given in which the general features of powered-lift aircraft are compared to conventional aircraft. This is followed by a presentation of findings based on the simulation experiments of this program as well as on other appropriate sources. Qualitative and, in many cases, quantitative criteria are proposed. Where criteria cannot be defined, problems are discussed and subjects for further study are recommended.

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FOREWORD

The research reported here was done under NASA Contract NAS2-7926 as part of a joint NASA/FAA program. The Contract Technical Manager was Jack E. Cayot, the FAA Project Manager was Barry C. Scott, and the Systems Technology, Inc. Project Engineer was Robert L. Stapleford. The work on this report was accomplished in the period from October 1975 through July 1976, but the report covers work which began in June 1972.

Successful completion of this project was due to the contributions and cooperation of many individuals besides the authors. Major contributions were made by: Jack E. Cayot (FAA), Barry C. Scott (FAA), Charles S. Hynes (NASA), Paul W. Martin (FAA), Ralph B. Bryder (Civil Aviation Authority [CAA], United Kingdom), and Gilles Robert (Centre D'Essais en Vol [CEV], France). Special thanks are due the pilots for their patience through many long simulator sessions and their many helpful suggestions. They were: Don Alexander (FAA), John A. Carrodus (CAA), Bryant M. Chestnutt (FAA), Robert Chubboy (formerly FAA), Richard M. Gough (FAA), Gordon H. Hardy (NASA), Robert J. Kennedy (FAA), George Lyddane (FAA), Michel Robardet (CEV), John Ryan (FAA), and J. P. Van Acker (CEV).

ABSTRACT

A four-year simulation program to develop airworthiness criteria for powered-lift aircraft is summarized. All flight phases affected by use of powered lift (approach, landing, go-around, takeoff) are treated with regard to airworthiness problem areas (limiting flight conditions and safety margins; stability, control, and performance; and systems failure). A tutorial discussion of each aspect is given in which the general features of powered-lift aircraft are compared to conventional aircraft. This is followed by a presentation of findings based on the simulation experiments of this program as well as on other appropriate sources. Qualitative and, in many cases, quantitative criteria are proposed. Where criteria cannot be defined, problems are discussed and subjects for further study are recommended.

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LIST OF ABBREVIATIONS

AMST	Advanced Medium STOL Transport
AWJSRA	Augmentor Wing Jet STOL Research Aircraft
c.g.	Center of gravity
CAA	Civil Aviation Authority (U.K.)
CEV	Centre D'Essais en Vol (France)
CTOL	Conventional takeoff and landing aircraft
DDC	Direct drag control
DLC	Direct lift control
EBF	Externally blown flap
FAR	Federal Aviation Regulation
FP	Flight path
FR	Flight reference
FSAA	Flight Simulator for Advanced Aircraft
IBF	Internally blown flap
IFR	Instrument flight rules
ILS	Instrument landing system
IVSI	Instantaneous vertical speed indicator
MF/VT	Mechanical flap/vectored thrust

MOT	Ministry of Transport (Canada)
OEI	One engine inoperative
OPUI	One propulsion unit inoperative
PIO	Pilot induced oscillation
RMS	Root mean square
SAS	Stability augmentation system
SCAS	Stability and control augmentation system
SSDWG	STOL Standards Development Working Group
STAI	STOL Tactical Aircraft Investigation
STOL	Short takeoff and landing aircraft
USB	Upper surface blowing
VASI	Visual approach slope indicator
VFR	Visual flight rules

LIST OF SYMBOLS

AR	Aspect ratio
a_z	Acceleration normal to the flight path
b	Reference wing span
C_J	Blowing coefficient, $T_b/S\bar{q}$
C_L	Lift coefficient, $LIFT/S\bar{q}$
C_{L_α}	Lift curve slope, $\partial C_L/\partial\alpha$
C_ℓ	Rolling moment coefficient
C_Y	Side force coefficient
d	Linear excursion normal to flight path
e	2.71823 ...
g	Gravity acceleration
h	Altitude
I_x	Moment of inertia about x-axis
I_y	Moment of inertia about y-axis
I_z	Moment of inertia about z-axis
L	Rolling moment
L_a	Rolling moment derivative with respect to variable a, $\frac{1}{I_x}\frac{\partial L}{\partial a}$
L_u	Horizontal gust scale length, rolling moment derivative with respect to airspeed
m	Aircraft mass
$\dot{m}_a$	Engine air mass flow
M	Pitching moment
M_a	Pitching moment derivative with respect to variable a, $\frac{1}{I_y}\frac{\partial M}{\partial a}$

M Safety margin

M_n Lift margin

M_{u_g} Horizontal gust margin

M_{w_g} Vertical gust margin

M_α Angle of attack margin

N Yawing moment

N_a Yawing moment derivative with respect to variable a, $\frac{1}{I_x}\frac{\partial N}{\partial a}$

n_{x_α} Tangential acceleration due to an angle of attack change

n_{z_α} Normal acceleration due to an angle of attack change

p Roll rate

p_e Probability of exceedance

q Pitch rate

$\bar{q}$ Dynamic pressure, $\frac{1}{2}\rho V^2$

r Yaw rate; also jet flap recovery factor

R Range from runway aim point

s Laplace operator

S Reference wing area

t Time

$t_{1/2}$ Rise time to one half peak

T Thrust, gross thrust; also transparency (BR 941)

T_b Thrust used for jet flap blowing

T'_c Propeller thrust coefficient, $T/S\bar{q}$

T_d Thrust furnishing a direct applied force

T_i Laplace transform time constant

u Velocity perturbation along x-axis

u_g Horizontal gust velocity

V	True airspeed
V_1	Critical engine failure speed
V_2	Takeoff safety speed
V_{app}	Approach speed
V_{MC}	Minimum control speed
V_{min}	Minimum speed
V_R	Takeoff rotation speed
V_s	Stall speed
V_{s_o}	Stall speed in landing configuration with power off
w	Velocity perturbation along z-axis
w_g	Vertical gust velocity
W	Aircraft weight
x	Body fixed axis, aligned with the flight path under steady state conditions
X	x-force
X_a	x-force derivative with respect to variable a, $\frac{1}{m}\frac{\partial X}{\partial a}$
y	Body fixed axis normal to the plane of symmetry, positive along right wing
Y	y-force
Y_a	y-force derivative with respect to variable a, $\frac{1}{m}\frac{\partial Y}{\partial a}$
a	Body fixed axis, normal to flight path in aircraft plane of symmetry under steady state conditions
Z	z-force
Z_a	z-force derivative with respect to variable a, $\frac{1}{m}\frac{\partial Z}{\partial a}$
$Z_w^{\dagger}$	Trimmed Z_w, $Z_w - (Z_{\delta_e}/M_{\delta_e})\ M_w$
α	Angle of attack
β	Sideslip angle
γ	Flight path angle

δ_a	Lateral control surface deflection
δ_e	Pitch control surface deflection
δ_T	Throttle deflection
ζ	Damping ratio
η_p	Powered lift factor, $1 + \frac{VZ_u}{2g}$
θ	Pitch attitude
θ_A	Effective inclination of forces due to an attitude change, $\arctan \frac{-Z_\alpha}{X_\alpha - g}$
θ_T	Effective thrust inclination, $\arctan \frac{-Z_{\delta_T}}{X_{\delta_T}}$
μ^{STOL}	Cross coupling parameter
ρ	Air density
σ	Standard deviation
φ	Bank angle
ψ	Yaw angle
Ω	Spatial frequency, $\frac{\omega}{V}$
ω	Temporal frequency
ω_p	Phugoid natural frequency

SUBSCRIPTS

app	Approach
c	Command
ec	Engine failure
eng	Engine
FL	Flare
max	Maximum
min	Minimum

o	Initial condition
p	Roll rate
q	Pitch rate
r	Yaw rate
s	Stall
u	x-velocity
u_g	Horizontal gust
v	y-velocity
w	z-velocity
w_g	Vertical gust
α	Angle of attack
β	Sideslip
δ_a	Roll control
δ_e	Pitch control
δ_r	Yaw control
δ_T	Throttle

SUPERSCRIPTS

eng	Engine

SECTION 1

INTRODUCTION

1.1 BACKGROUND

This report summarizes a four-year program to develop airworthiness criteria for powered-lift aircraft. The program consisted of a series of simulator experiments which were conducted on the Flight Simulator for Advanced Aircraft (FSAA) at the NASA Ames Research Center.

The initial simulations concentrated on determining the major problem areas regarding airworthiness criteria for powered-lift aircraft and especially those areas where existing airworthiness standards might not be appropriate for powered lift. Later simulations addressed specific problems and potential criteria. The last simulation was primarily an evaluation of tentative criteria which had been developed. A more detailed review of the whole program is given in the next subsection.

The simulation efforts were supported by analytical studies and detailed reviews of other programs which were also concerned with powered-lift airworthiness or handling qualities. The following paragraphs describe those efforts which had a major impact on this program.

Reference 1, commonly known as FAR Part XX, was the initial attempt to formulate airworthiness standards especially for powered-lift aircraft. A document of this nature is the ultimate goal for this criteria development program. Reference 2 was a systematic review of Part XX which more directly addressed the special problems related to powered-lift airplanes. Another formal attempt to set forth civil airworthiness standards for powered-lift aircraft was done in the British counterpart to Part XX, Section P (Reference 3).

A large body of research literature on powered-lift aircraft was available. Efforts which were directly aimed at civil airworthiness criteria were described in References 4 through 9. Reference 4 was specially noteworthy because it summarized several NASA flight test programs involving actual powered-lift airplanes and proposed a number of criteria.

In addition, a large body of literature representing programs of a more limited scope than those above influenced the conduct of this program. One concurrent simulation research program having a significant impact on the work reported here is described in Reference 10. The contributions of various programs are indicated throughout the sections of this report by specific reference.

We should also note two current flight test programs which can be expected to have a major influence on the ultimate airworthiness standards. The first of these is a research program being conducted at the NASA Ames Research Center with the Augmentor Wing Jet STOL Research Aircraft (AWJSRA). The second is part of the Air Force competition to develop an Advanced Medium STOL Transport (AMST). The two competing designs are the Boeing YC-14 and the McDonnell Douglas YC-15.

1.2 PROGRAM HISTORY

This program originated in mid 1972 as a joint FAA/NASA effort to use manned simulation to develop STOL airworthiness criteria. Major milestones in this program are listed in Table 1-1 and are discussed below.

The first formal simulation period in this program was begun in October 1972 using an STI-developed model of the Breguet 941S as the subject airplane. This model was intended to reflect the general characteristics of deflected-slipstream powered-lift airplanes. During this first simulation, general test procedures were developed which were used in subsequent simulations. A relatively broad range of operations was investigated including transition from cruise to approach, approach and landing, go-around, and takeoff. Several approach cases were examined by considering different approach speeds with and without "transparency," (differential inboard/outboard propeller pitch which redistributed lift and thus varied the aerodynamic characteristics).

TABLE 1-1

PROGRAM MILESTONES

July	1972	NASA/FAA/CEV Flight Familiarization with BR 941S
October/November	1972	First BR 941 Simulation
April/May	1973	Second BR 941 Simulation
July/August	1973	AWJSRA Simulation
January/February	1974	First Generic STOL Simulation
June/July	1974	Second Generic STOL Simulation
September	1974	First Meeting of the STOL Standards Development Working Group
November/December	1974	STOL-X Simulation
April/May	1975	Second Meeting of the STOL Standards Development Working Group
April	1976	NASA/FAA Report on Progress Toward Criteria Development
September	1976	Summary Report

After an analysis period, the BR 941 simulation was continued in April 1973. The approach and landing became a more central area of investigation. The combined results of the BR 941 simulations were reported in Reference 11.

In July 1973 a second subject airplane model was investigated. This model was based on the NASA Augmentor Wing Jet STOL Research Aircraft (AWJSRA) and allowed us to view a design employing a different form of powered lift, i.e., augmentor wing. The same set of flight phases were considered as with the BR 941. The approach and landing, however, took on an increased emphasis. Several approach speeds were examined as well as use of different flight path controls (i.e., throttle, nozzle angle, direct lift control, and direct drag control). The documentation of this simulation, Reference 12, included initial attempts to formalize a theory of flight path/flight reference control in the approach and establish criteria based on this theory.

The BR 941 and AWJSRA simulations examined relatively complete models over a range of piloting tasks. The approach and landing emerged as potentially the most critical piloting task for powered-lift airplanes and the area most lacking in effective airworthiness criteria. At the same time, there were indications that various forms of powered lift (deflected slipstream, augmentor wing, externally blown flap, internally blown flap, etc.) all yielded relatively similar flight path control dynamics in a generic sense. This was more formally developed in a parallel FAA program to study STOL transport flight path control (Reference 10). Thus the next simulation focused on airworthiness problems in the approach using a generic STOL model rather than a specific airplane or specific type of powered lift. This generic representation allowed direct variation of many individual features of interest.

The first Generic STOL simulation was begun in January 1974. The objectives of this simulation were to study speed margins, flight path control power requirements, and flare and landing techniques. The general form of the model used allowed a direct variation of specific airplane characteristics which we wanted to study.

A second Generic STOL simulation was conducted in June 1974 to examine still other specific topics of interest. These included flight path/flight reference cross coupling and short-term flight path response. In addition, in order to answer questions concerning turbulence realism, the effects of the low altitude turbulence model were studied by comparing the previously used MIL-F-8785B model and an alternative turbulence model. This involved use of a familiar subject airplane, the Twin Otter. The turbulence model comparison showed no clear distinction with regard to realism and use of the MIL-F-8785B model continued. The results of both Generic STOL simulations were reported in Reference 13.

The first STOL Standards Development Working Group* (SSDWG) meeting was convened in September 1974 at the NASA Ames Research Center. The meeting was attended by representatives of the FAA, NASA, MOT (Canada), CAA (United Kingdom), CEV (France), and STI. The objective of this meeting was to review the results of the simulation exercises conducted over the prior two years in order to propose revisions to FAR Part XX, Reference 1. It was proposed that the results of this meeting be used as the basis for the subsequent simulation phase, i.e., that known as the STOL-X simulation.

The STOL-X simulation centered about an aircraft design contrived to just meet a number of the criteria discussed in the first working group meeting. This hypothetical design was based on an actual preliminary design of a powered-lift transport, but modifications were made to tailor the characteristics to the proposed criteria levels. During the simulation period a number of minor variations were made to better define the criteria limits. The results of this simulation experiment are given in Reference 14.

A second working group meeting was convened during April 1975 to further discuss development of airworthiness criteria especially in the light of the STOL-X simulation. The outcome of these discussions is presented in Reference 15. This document reflects not only the data

* During the second meeting of this group, the members agreed that the working group title was somewhat misleading. It would be more correct to change "STOL" to "Powered-Lift" as the group was concerned only with powered-lift aircraft and not low-wing-loading STOL aircraft.

collected in the program reported here but also important inputs from each of the working group members and participating agencies.

The final program milestone is the issuance of this summary report. Although reports had been written covering each simulation experiment, this overall summary was considered necessary. The basic objective was to collect and interpret the results from all the simulations as well as outside data sources. Another objective was to clarify conflicts between findings and hypotheses given in the earlier and later simulation reports. This report therefore supersedes References 11, 12, 13, and 14 but does not replace them as detailed descriptions of the experiments and results are not repeated here.

1.3 REPORT ORGANIZATION

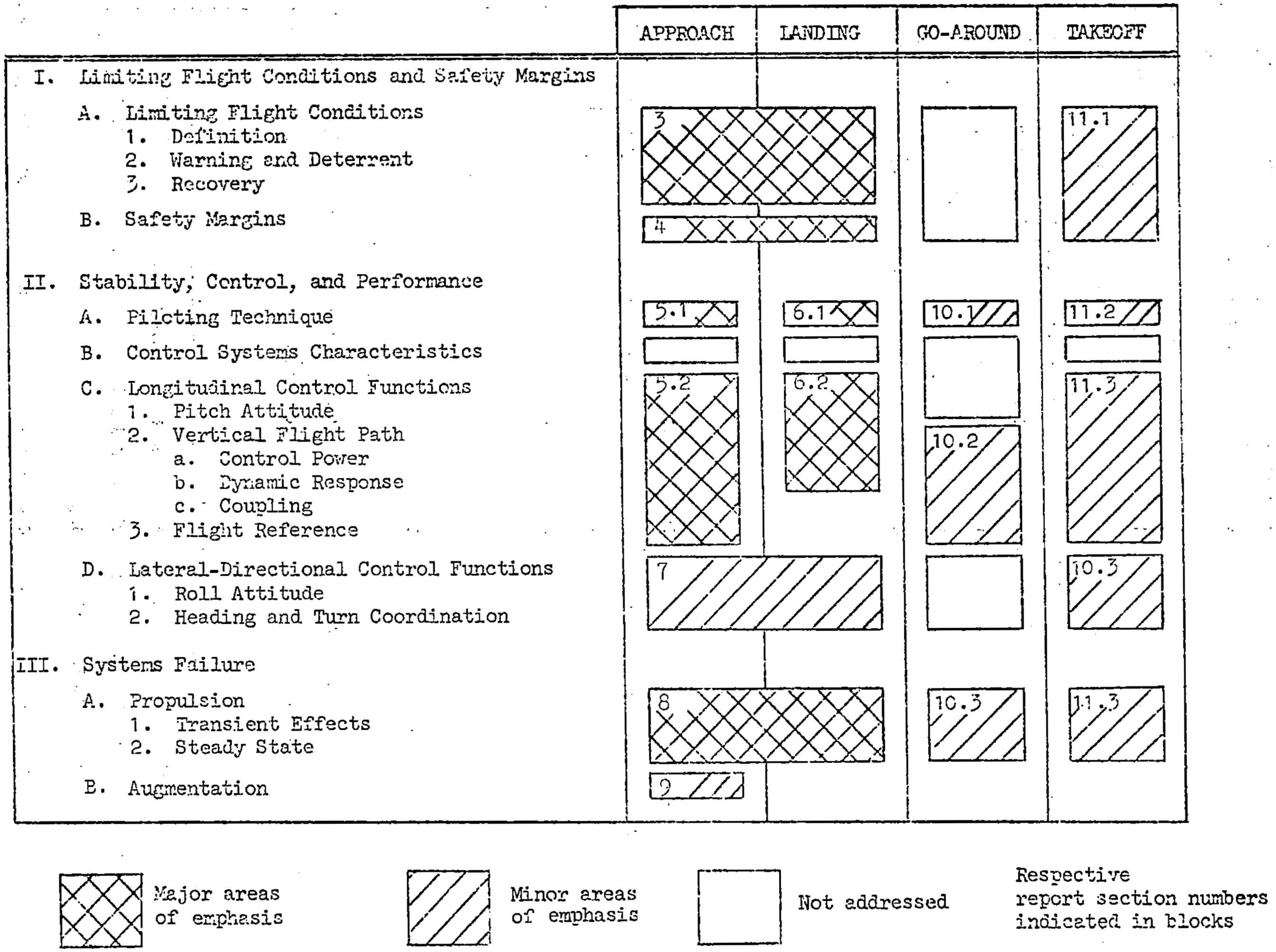
Section 2 is a general description of the simulation facilities and test procedures used throughout this program. The remainder of this report is organized into sections which cover the important aspects of several key flight phases. Figure 1-1 shows this organization in a schematic manner and gives an indication of the program emphasis. The shaded blocks indicate the areas of major and minor concern and give the respective report section.

Each section begins with a general tutorial discussion of the subject matter and ends with a presentation of findings from this program. Where possible, the results of this program are correlated with data obtained from other research programs. Specific criteria are suggested where the data warrant this.

The final section, Section 12, gives a summary for each of the areas considered. This includes an assessment of our current ability to define appropriate airworthiness criteria and recommendations for additional research where it seems necessary.

Because of the special importance of the longitudinal flight path dynamics and of atmospheric disturbance modeling, appendices dealing with these subjects are included.

Figure 1-1: Report Organization



SECTION 2

DESCRIPTION OF THE EXPERIMENTAL APPROACH

The following pages summarize the main features of the experimental approach taken in this simulator program. The details of each respective part of this program are covered in the simulation reports (References 11 through 14). The elements of the experimental approach which are described here are:

- Test procedure
- Airplane models
- Operational environment
- Simulator apparatus
- Subject pilots
- Data acquisition.

Each of these is taken as a subsection topic. The objective is to define the important features of the experimental approach.

2.1 TEST PROCEDURE

The test procedure used in this simulation program was, briefly stated, to examine powered-lift vehicles operating in the terminal areas. This was accomplished through a consideration of pilot opinion, overall pilot/vehicle performance, and engineers' observations. There was no strict reliance on any one of these.

When we considered any given flight phase, it was done within the proper context; for example, the go-around flight phase was always preceded by a realistic approach flight phase. The tasks themselves were made as realistic as possible although the pilots were requested to carry out tasks in sometimes unusual ways. For example, flare and landings using power instead of pitch attitude were examined. While this was a new experience for many of the pilots, it was conducted within a realistic setting and with realistic landing constraints. In some cases, flight test procedures were examined rather than specific flight phases or tasks. One notable example of this was approach to stall or high angle of attack conditions.

The specific flight phases which were considered in this program included:

- Takeoff
- Transition from cruise
- Approach
- Landing
- Go-around.

After examining all of these in the initial program simulations, it became clear that special emphasis should be placed on the approach and landing flight phases.

When conducting the simulation experiments, the pilot was normally given guidance as to the appropriate piloting technique, special performance objectives, and the nature or objectives of the particular experiment. This was found preferable to keeping the pilot unaware of test objectives and having him search out problems and solutions on his own. Formality in defining the experiment and pilot instructions varied over the course of the program. Initially a complete set of detailed flight cards were used but later only oral briefings were given to the pilot, along with greater latitude in examining the problem. In the final simulation period, however, we reverted to the use of detailed flight cards. Not only was this more desirable to the pilots but it also forced the experimenter to follow more closely the program plan and objectives.

It was found that it was efficient to have an engineer accompany the pilot during the runs to take notes and interrogate the pilot for the taped record. In general, the most valuable information obtained was through direct observations made by the pilots and engineers.

2.2 AIRPLANE MODELS

In this simulation program, several airplane models were used. We will describe each of these briefly with special notes on model construction and significant problems encountered.

The airplane models which will be described in the following paragraphs are:

- BR 941
- AWJSRA
- Generic STOL series of configurations
- STOL-X.

The BR 941 model (Volume III of Reference 11) was developed by STI specifically for this program. The objective was to develop a highly detailed model of the BR 941S, which was flown by four of the pilots prior to the simulation program. Special emphasis was given to modeling the propulsion system including the individual effects of propellers, governor, and engines. In addition to previously available flight test data on the BR 941.01, we also used additional flight test data from the BR 941S. These latter data were collected during the flights made in preparation for this program. Development of the model included one short simulation period with the same four pilots to fine tune the model.

One important feature of the BR 941 model was use of analytical functions to describe the aerodynamic, propulsion, and landing gear parts of the model. This was in contrast with the usual use of tables to define characteristics. One important advantage in doing this was the reduction in the total number of parameters to define the model. For example, tables of lift coefficient values involving several hundred numbers could be replaced by a few coefficients to describe an analytic function. Another advantage was that it was easier to adjust parameters to fine tune the model.

The BR 941 model was successful in defining a complex and sophisticated airplane model with an economy of parameters compared to use of a tabular definition scheme. This model was a forerunner to the Generic STOL model which was developed at a later phase and is described shortly.

The AWJSRA model used in this program was an existing NASA computer model (Reference 16). It was an early version of a model which has been subsequently refined and updated. This model made wide use of tabulated aerodynamic data in contrast to the aforementioned program. Some minor modifications were made to the AWJSRA model for this program. These included the removal of some propulsion system non-linearities, implementation of a separate control DLC and DDC option, and substitution of the BR 941 landing gear model. This model, like the BR 941, was a highly detailed simulator model. It allowed operation throughout the entire flight envelope of the airplane.

One important addition to the AWJSRA model was an all-axes stability augmentation system plus a flight director and a configuration management system. The SAS was essentially the same as that used in the latter stages of the BR 941 simulation. The flight director and configuration management system was one specially designed for the AWJSRA in another program, described in Reference 17.

The Generic STOL simulator model consisted of a general computer program capable of modeling a wide range of powered-lift and conventional aircraft types. During the two Generic STOL simulation periods a single basic airplane model was used but certain longitudinal characteristics were varied in order to examine a large number of configurations. These various longitudinal models were devised to carry out specific experiments related to safety margins, flare and landing, flight path control power, dynamic response, speed/path cross coupling, and examination of turbulence model effects. In some cases the configurations which were used did not reflect realistic powered-lift aerodynamics. They were deliberately contrived to examine certain important handling features.

The Generic STOL computer program itself was developed in another project and is described in Reference 10. The model provides the basic framework for an airplane simulation and has been used in simulation programs involving both powered-lift and conventional airplanes. The model was based on an analytic function description of aerodynamic and propulsion characteristics much the same as the BR 941 model.

The STOL-X model (Volume II of Reference 14) was a realistic powered-lift airplane configuration utilizing the Generic STOL simulator program. This configuration was developed by STI to explore certain tentative airworthiness criteria. This was done by making the characteristics of the airplane just meet these criteria. The configuration was based on a preliminary design developed in an Air Force study program (Reference 18). In more specific terms, this design employed an EBF powered-lift concept utilizing four turbo-fan engines and was in a weight class just slightly lighter than the current Air Force AMST (Advanced Medium STOL Transport) designs. One of the special objectives was to operate at minimum safety margins; hence the design flight condition involved a lift coefficient of about six, a relatively high value compared to most powered-lift designs to date.

2.3 OPERATIONAL ENVIRONMENT

The operating environment important to this program consisted of two main parts: the ground environment including the airport and associated terrain and the atmospheric environment. The important aspects of each of these will be discussed in the following paragraphs.

The central feature of the ground environment for this simulation program was a 600 to 1 scale replica of a STOL port and surrounding countryside. Runway dimensions and markings were largely in accordance with Reference 19, but minor modifications were made at various times over the span of this simulation program. Details are reported in the respective simulation reports.

The forms of approach guidance provided to the pilot included electronic, VASI, and a normal visual scene. Electronic guidance consisted of normal cockpit instruments with varying ILS glide slope and localizer angles and sensitivities depending upon the particular experiment and aircraft. Variation of some of the electronic guidance parameters was the subject of some minor experiments. For example, a localizer offset angle of 6 deg from the runway centerline was evaluated during the first series of BR 941 experiments. Some BR 941 and STOL-X experiments involved significant variations in glide slope angle. The majority of the experiments, however, employed a straight in approach on a 6 deg glide slope.

Two types of VASI were utilized during this program. During some of the BR 941 and AWJSRA experiments, the VASI consisted of a pair of fixed sighting bars located on the side of the runway. These were later replaced by a two-color light system. In general, the VASI systems were used to enhance the pilot's outside visual reference during the latter stages of the approach. This was an attempt to offset a sometimes marginal video display. In later experiments the VASI was not used.

The visual display consisted of a closed-circuit color TV system using a 600 to 1 scale model of the terrain. The angular field-of-view provided was 37 deg vertically by 48 deg horizontally. The most severe constraint imposed by this was the limit on crab angle in a crosswind without losing sight of the runway. A standard color TV monitor was used with the picture sharpness apparently having some effect on altitude and altitude rate perception, especially during the flare and landing.

The features describing the atmospheric environment included random turbulence, deterministic winds and shears, and visibility.

Visibility was adjusted by setting the cloud ceiling. This ranged from a totally visual approach to an IFR approach in instrument meterological conditions with the ceiling set near the decision height. Missed approaches were forced by setting the ceiling to zero.

The random turbulence model used throughout this program is fully described in Appendix B. The turbulence intensity was the only parameter which was independently varied. Usually the turbulence level was characterized as either "calm" or "turbulent". Early in the program the calm condition was perfectly calm air. During latter experiments it was found that a level of $\sigma_{u_g} \doteq 0.46$ m/s (1.5 ft/s) was more satisfactory because it would not permit the pilot to fly totally hands off. The standard turbulent condition corresponded to $\sigma_{u_g} = 1.37$ m/s (4.5 ft/s). This level is exceeded only 10% of the time according to Reference 20. In a few cases a 1% turbulence level was used, σ_{u_g} = 2 m/s (6.5 ft/s).

The main source of atmospheric disturbance in this simulator program was random turbulence, but the question of its validity lingered throughout the program. Some of the subject pilots felt that the turbulence level which was characterized as having a 10% probability of exceedance was unrealistically severe in the simulator. Prior to the second Generic STOL simulation, a study of various random wind models was made, and a large number of sources dealing with low-level turbulence models were surveyed. Based on analysis, there did not appear to be a significant difference in the net effect of any of the models surveyed. Nevertheless, a short simulation experiment was run to study the more widely varying turbulence model parameters. This seemed to confirm that the model originally used was as realistic as any of the alternatives, and its use continued throughout the remainder of the simulator program.

It should be noted that in a subsequent evaluation of the standard turbulence model involving use of the Princeton Variable Stability Navion, the model again appeared realistic. This result is reported in Reference 10.

Since the turbulence model used here seems to have been shown reasonably valid, at least quantitatively, the main problem may have been in the subject pilots' interpretation of a given probability of exceedance. At any rate, this should not alter the validity of the data obtained from the simulator experiments run. Since pilot ratings were obtained using well-defined disturbance levels, it is possible to reassess minimum acceptable boundaries for other levels of disturbance.

Various combinations of deterministic winds and wind shears were used along with the random turbulence in some of the experiments in this program. During the first two simulation series each set of runs contained a variety of wind profiles and turbulence. This was found awkward. The shear magnitudes were somewhat arbitrary because no probability estimates could be made. At the same time it was found that the random turbulence model provided random wind shears (as it appeared to the pilots) due to the significant low frequency content. For this reason, during the Generic STOL and STOL-X simulations the large deterministic wind shears were removed from the test matrix and replaced by a 1/6 power law wind profile to provide only boundary layer wind shear effects.

It was clear from initial BR 941 simulations that wind shears were at least as important disturbance factors as was random turbulence, but a more recent simulation program (Reference 21) provides a better insight to the effects of wind shears on powered-lift vehicles. This involved a systematic variation of shear magnitude and duration. The vehicles included a relatively conventional airplane (a low wing loading STOL) along with a selection of powered-lift configurations with various augmentation devices. The results of this experiment are discussed in later sections (safety margins and longitudinal flight path/flight reference control).

2.4 SIMULATOR APPARATUS

The entire simulation program described here was carried out on one simulator facility at the NASA Ames Research Center. This consisted of the Flight Simulator for Advanced Aircraft (FSAA) and a Redifon visual display system. The following is a brief description of each of these devices.

The FSAA is a six-degree-of-freedom moving base simulator with an unusually large lateral motion capability. The FSAA provided generally realistic motion for simulated flight including the effects of turbulence and maneuvering by the pilot. Its most apparent limitation was its inability to provide a good vertical acceleration cue during flare and at touchdown. It was necessary to augment the touchdown motion cue by advising the pilot, after landing, of his touchdown sink rate.

The cockpit of the FSAA was specially configured for the BR 941 and the AWJSRA airplane simulations. For the Generic STOL and STOL-X simulations a cockpit representative of conventional transports was used. This cockpit was generally similar to those specific airplanes mentioned above but differed mainly in having a center console throttle quadrant.

The Redifon visual display was a crucial element of the simulator apparatus. This was because most of the critical piloting tasks occurred at a time when visual reference was required, such as flare and landing. The Redifon device used was adequate for conducting this program but had to be maintained at its maximum potential. It was considered important to frequently check the altitude and longitudinal position calibrations. The sharpness of the picture displayed to the pilot seemed to be the most critical aspect of the visual display. This was especially important during the flare maneuver where sink rate and altitude perception was of special concern to the pilot. The subject pilots were sensitive to even the slightest degradation in picture quality.

2.5 SUBJECT PILOTS

A relatively large number of subject pilots participated in this simulation program. These pilots represented NASA and the civil aviation agencies of France, the United Kingdom, and the United States. Most of the pilots had experience in flight testing and experience with various types of STOL or powered-lift aircraft. At the same time there was a good deal of diversity in the backgrounds of these individuals. Table 2-1 briefly describes the experience of each of the subject pilots along with any special qualifications pertinent to this simulation program.

2.6 DATA ACQUISITION

The data collected during the simulation program were of three forms: written comments, oral (taped) comments, and recorded performance.

Standard questionnaire forms were the basis for written pilot comments. Frequently, though, extensive written reports were prepared on specific items encountered during simulation runs.

Oral interrogation of subject pilots was found to be the most important source of information. This could be accomplished during or immediately following individual runs.

TABLE 2-1

SUBJECT PILOT BACKGROUND

DON ALEXANDER FLIGHT TEST PILOT FAA, WESTERN REGIONAL OFFICE	JOHN CARRODUS ASSISTANT CHIEF TEST PILOT CAA (United Kingdom)	BRYANT CHESTNUTT FLIGHT OPERATIONS SPECIALIST FAA
• Current operational experience in C-141 (7 years). • Extensive flight test experience (7 years) in the basic certification of airplanes, e.g., DC-10, Super Guppy, L-1011, all types of general aviation airplanes. • Limited STOL experience -- Twin Otter. • No helicopter experience. • Limited research simulation experience (FSAA).	• Some STOL experience as a certification test pilot of smaller twin turboprop types (e.g., Skyvan) plus a limited amount of heavier twin turboprop types (AVRO 748) and a jet V/STOL type (Harrier). • Certification experience with multi-engine commercial aircraft (L-1011). • Limited experience in helicopters and light aircraft. • Considerable simulator experience. • Military experience as naval fighter pilot and as test pilot (primarily fighters).	• Current flight experience in conventional light twin and DHC-6. Majority of time in heavy multi-engine (DC-3, DC-4, DC-9). • Participated in STOL evaluation at NAFEC using DHC-6. • No helicopter experience. • FAA instructor and check pilot in conventional light and heavy multi-engine aircraft. Extensive experience as navigation facilities flight check pilot (DC-3 and DC-4).

TABLE 2-1 (Continued)

LTC. ROBERT CHUBBOY (U.S. ARMY) R & D SPECIALIST FAA	RICHARD GOUGH FLIGHT TEST PILOT FAA	GORDON HARDY RESEARCH PILOT NASA
• Current rotary wing and light single and twin engine fixed wing. • Extensive STOL test and operational experience (DHC-2, 4, 5, and XC-142). Limited experience in BR 941S. • Extensive rotary wing test and operational experience in a wide range of helicopters. • Extensive research simulator experience in a wide variety of aircraft.	• Current experience in conventional airplane airworthiness certification programs (DC-10, L-1011, etc.). • Research test pilot for USAF flying wide range of conventional fixed wing aircraft. (Fighter, bomber, trainer, utility, light STOL). • Limited STOL experience (YC-134, BR 941S, AWJSRA). • Little rotary wing experience. • Considerable ground based simulator experience. • R and D subject in TIFS (Concorde simulation).	• Current flight experience largely in conventional aircraft (CV-340, CV-990, Lear Jet). • Limited experience in several STOL aircraft (DHC-5, DHC-6, AWJSRA, BR 941S) as research pilot. • No helicopter experience. • Extensive light aircraft experience. • Military experience in conventional single engine fighter/ attack aircraft. • Research simulator experience in a range of handling qualities experiments (space shuttle, DHC-6, AWJSRA, AMST, STOLAND, etc.).

TABLE 2-1 (Continued)

ROBERT KENNEDY FLIGHT TEST PILOT FAA	GEORGE LYDDANE FLIGHT TEST PILOT FAA, WESTERN REGIONAL OFFICE	MICHEL ROBARDET FLIGHT TEST PILOT CEV (France)
• Seven years experience as FAA flight test pilot. (Participated in STOL project at NAFEC using DHC-6 and Heliporter). • Experienced test pilot for Piasecki and Vertol in ducted fan aircraft and helicopters. • Considerable ground based simulator experience in STOL programs. • Military experience in wide range of aircraft (fighter, bomber, transport, helicopter, etc.).	• Extensive light aircraft flight test experience, basic certification (performance, S and C, and systems testing). • 11 years military flight experience (7 years as a flight test pilot — Primary flight test programs include C5A, B-52, U2, B-57, Boeing 737 [T-43A]). • Very limited STOL experience, C-130, Skyvan. • No helicopter experience. • Limited research simulator experience - FSAA.	• Considerable experience in multi-engine transport aircraft as airline captain, military transport command instructor, and certification test pilot. • Some STOL experience as TRANSALL test pilot. • Little rotary wing experience, limited experience in light aircraft. • Substantial experience in jet fighters and bombers (at CEV). • Limited experience in research simulators. (Extensive experience in modern training simulators).

TABLE 2-1 (Concluded)

JOHN RYAN FAA	J. P. VAN ACKER TEST PILOT CEV (France)
• Current experience as flight test pilot in NAFEC curved path MLS program. • Experienced in BR 941S and DHC-6. • Helicopter experience. • Extensive simulation experience.	• Current flight experience in military aircraft (fighter, transport) and airbus certification program. Research pilot for variable stability Mirage. • Considerable experience with TRANSALL C160 modified for STOL operation and limited experience in BR 941S. • Military experience with fighter/attack aircraft. • Extensive simulation experience.

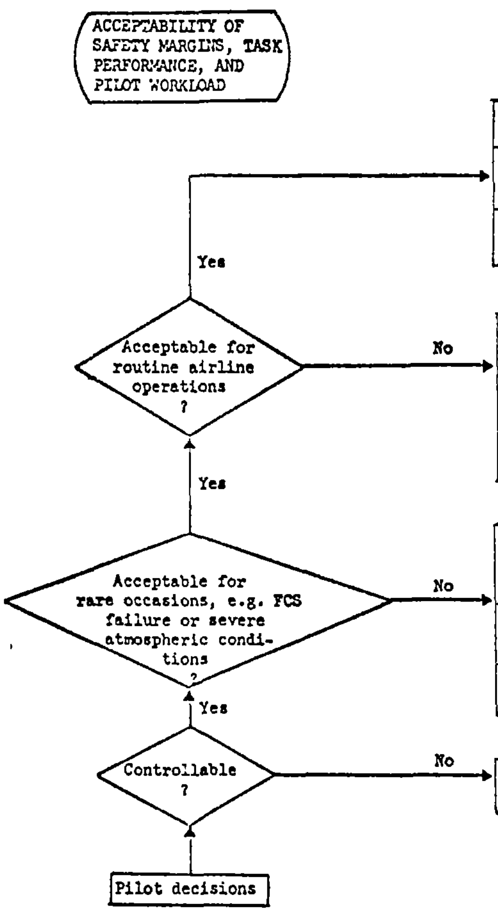
Various forms of recorded performance data were gathered. Analog strip charts of a large number of variables were always taken. Various forms of statistical performance data were also available but sometimes were not taken because the data output increased simulator run time. The forms of performance records are shown in detail in the individual simulation reports. In general, verbal data were relied on more heavily than performance data. Early in the program it was found that pilot ratings and comments would reflect degraded conditions before the performance data would.

During this program a modified Cooper-Harper rating scale was used as a quantitative indication of task difficulty. This scale is shown in Figure 2-1. The modifications from the standard Cooper-Harper scale reflect the need to better address the matter of airworthiness. Specific modifications are wording changes in the decision tree of column one and addition of the safety margin aspects of column three.

One unavoidable difficulty connected with the use of the rating scale concerns the role of atmospheric disturbances. Although levels of severity of disturbances are not explicitly addressed in the scale, there is an effect on ratings depending upon an individual pilot's assessment of probability of occurrence or exceedance. As mentioned earlier, this assessment was not entirely understood.

Figure 2-1

Modified Cooper-Harper Rating Scale



GENERAL CHARACTERISTICS	SAFETY MARGINS	DEMANDS ON THE PILOT	PILOT RATING
Excellent Highly desirable	Clearly adequate	Pilot compensation not a factor for desired performance	1
Good Negligible deficiencies	Clearly adequate	Pilot compensation not a factor for desired performance	2
Fair - Some mildly unpleasant deficiencies	Clearly adequate	Minimal pilot compensation required for desired performance	3
Minor but annoying deficiencies	Clearly adequate	Desired performance requires moderate pilot compensation	4
Moderately objectionable deficiencies	Adequate	Adequate performance requires considerable pilot compensation	5
Very objectionable but tolerable deficiencies	Marginal	Adequate performance requires extensive pilot compensation	6
Major deficiencies	Inadequate	Adequate performance not attainable with maximum tolerable pilot compensation Controllability not in question	7
Major deficiencies	Inadequate	Considerable pilot compensation is required for control	8
Major deficiencies	Inadequate	Intense pilot compensation is required to retain control	9
Major deficiencies	None	Control will be lost during some portion of required operation	10

SECTION 3

LIMITING FLIGHT CONDITIONS; APPROACH AND LANDING

This section covers the subject of flight at, or approaching, limiting conditions of the flight envelope which are related to high angles of attack and low airspeeds. In the case of conventional aircraft this would correspond to the region near aerodynamic stall. For powered-lift aircraft, this region may also be characterized by aerodynamic stall, but requires a more complex treatment.

In order to cover the subject, this section is organized in the following manner:

- Definition of limiting flight conditions
- Approach to and recovery from limiting flight conditions
- Warning of and deterrent to limiting flight conditions.

We begin by giving a background discussion of limiting flight conditions for powered-lift aircraft and then present related simulation results.

A key difference between conventional and powered-lift aircraft is the strong effect of power setting on the relationship between lift and angle of attack. This is shown in Figure 3-1 in which typical plots of lift coefficient versus angle of attack are shown. Note that for a conventional aircraft there is nearly a one-to-one relationship between lift coefficient and angle of attack; the variation between power-off and maximum power is relatively insignificant. On the other hand, for a powered-lift aircraft, a wide range of lift coefficients is possible at any given angle of attack depending upon power setting. Of particular interest is the fact that $C_{L_{max}}$ for a powered-lift aircraft can vary greatly depending upon thrust. Also, it is possible for the angle of attack at $C_{L_{max}}$ to vary with thrust.

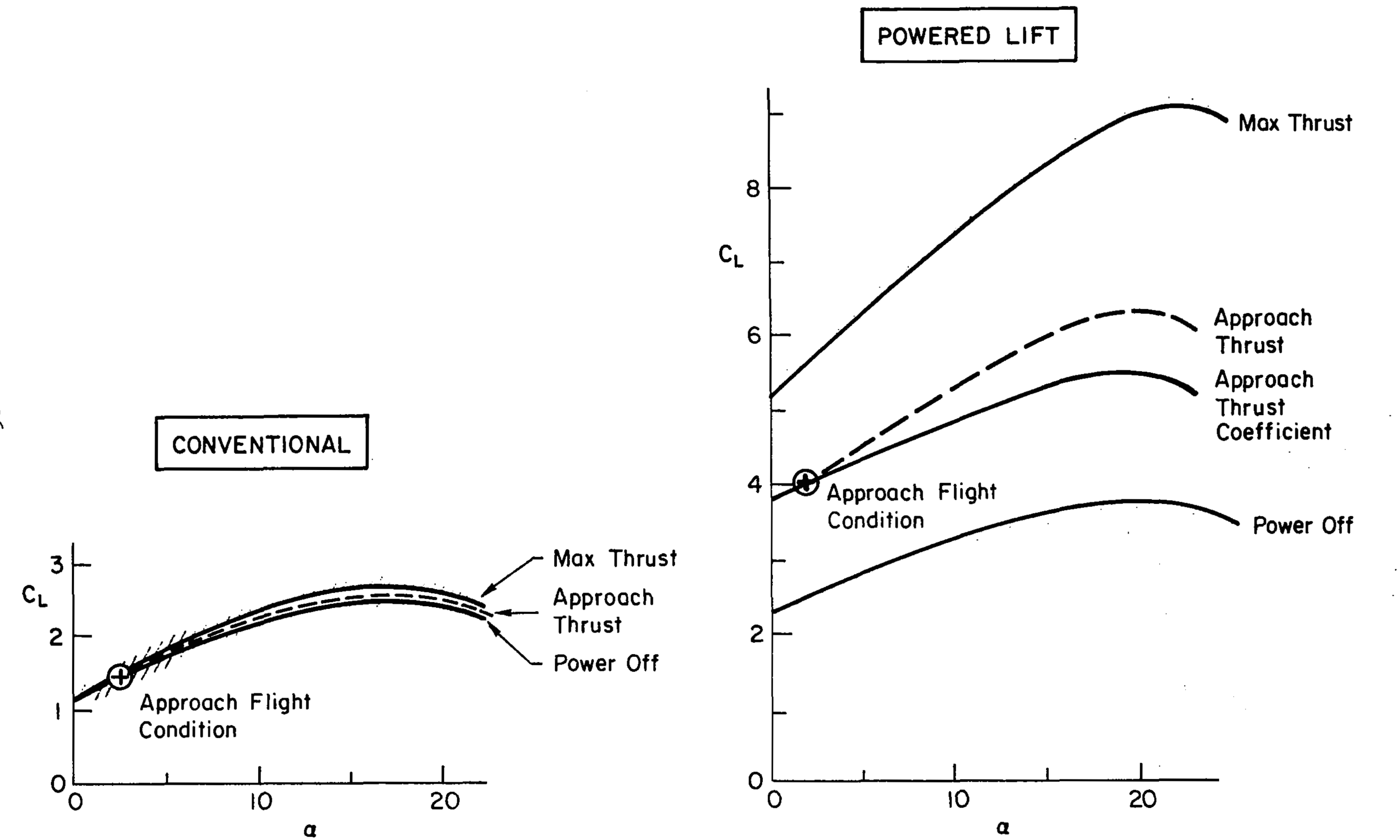


Figure 3-1: Typical Plots of Lift Coefficient versus Angle of Attack and Thrust

The following pages will dwell on the behavior of a powered-lift airplane in the region of aerodynamic stall. The relationships shown in Figure 3-1 will be discussed more fully and the simulation results which relate to limiting flight conditions in the region of aerodynamic stall will be presented.

3.1 DEFINITION OF LIMITING FLIGHT CONDITIONS; APPROACH AND LANDING

The objective of this subsection is to enumerate, in some detail, the conditions which constitute limits of the flight envelope for a powered-lift airplane and to give related simulator observations and findings.

A general definition of limiting flight conditions is that they form the boundary of the usable flight envelope (we are considering, in particular here, the high angle of attack/low airspeed boundary). Beyond the boundary of the usable flight envelope it is assumed that there would be a substantial change in flight characteristics which may be totally uncontrollable or, at the very least, a cause of major problems in aircraft operation.

Let us begin by considering the case of a conventional aircraft. For most conventional aircraft, the power-off stall is likely to be the defining feature of its limiting flight condition. At or near aerodynamic stall, conventional flight dynamics cease to exist and a large percentage of aerodynamic lift may be lost with only a small angle of attack increase. In some cases the adversity which dominates is related to loss of control in the lateral-directional axes. These limiting flight conditions can normally be associated with an angle of attack. In addition, there can also be a limiting flight condition created by inadequate dynamic pressure, e.g., the minimum control speed related to propulsion failure. This, then, would be tied to airspeed as opposed to angle of attack. But, a single equivalent airspeed is all that is needed to essentially define the 1 g limiting flight condition for a conventional aircraft (for a given wing loading) whether it be primarily a function of angle of attack or of airspeed. The nearly one-to-one relationship between C_L and angle of attack allows this simplification.

For the powered-lift aircraft, the same kinds of adversities can form limiting flight conditions. If, however, the limiting flight condition is related to aerodynamic stall there can be a wide range of airspeeds and angles of attack at which this can occur. This can be described in terms of a variety of types of aerodynamic stall. Figure 3-2 illustrates a variety of limiting flight conditions stemming from aerodynamic stall and how they depend upon the power setting.

The first condition we consider is power-off stall. While this is meaningful for conventional aircraft because it is well defined and fairly invariant, it has less significance in the operation of a powered-lift airplane. This is because the approach speed is likely to be below the power-off stall speed* as shown in Figure 3-2.

Next, consider the condition of aerodynamic stall with approach power (or throttle setting). The two cases we will consider here are 1 g stall at approach power and an accelerated stall at approach power and approach speed. The latter follows a contour of constant blowing coefficient since speed and power stay constant. In the case of a 1 g stall at approach power the blowing coefficient increases as the airplane slows. Thus, $C_{L_{max}}$ for a 1 g stall will generally be greater than $C_{L_{max}}$ for an abrupt, constant speed stall. This kind of relationship will be of particular importance in the discussion of safety margins in Section 4.

The final case to be considered is aerodynamic stall occurring at maximum power in unaccelerated flight. As shown in Figure 3-1, this represents the maximum obtainable lift coefficient and consequently the lowest trim airspeed for a given configuration.

It should be noted that any of the types of aerodynamic stall mentioned above can be characterized in terms of a maximum angle of attack which is, in turn, a function of thrust coefficient. At the same time, any of the 1 g stall conditions can also be characterized as a minimum airspeed which is a function of power. Only the accelerated stall must be defined strictly

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* Power-off stall speed, as used here, refers to the 1 g stall speed and not the V_{min} normally associated with the FAR Part 25 certification process.

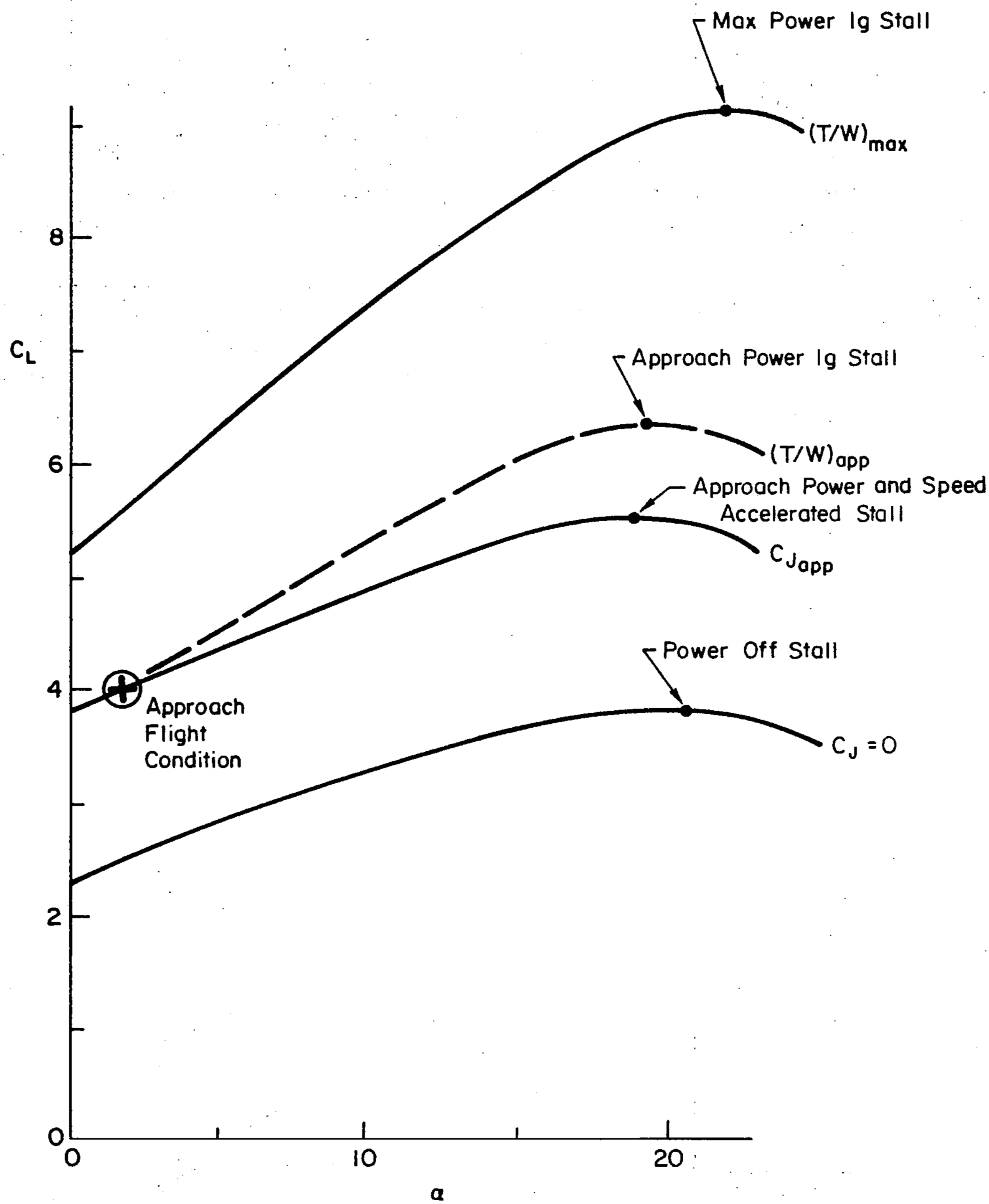

Figure 3-2: Variety of Stall Types
for Powered-Lift Aircraft

in terms of maximum angle of attack. In all cases, angle of attack may be the best defining parameter because it is independent of load factor and may not be particularly sensitive to throttle setting.

A limiting flight condition purely due to a lack of dynamic pressure is an important factor for powered-lift airplanes. This kind of limiting flight condition is usually associated with inadequate aerodynamic control power. Power setting can effect this minimum dynamic pressure just as it does aerodynamic stall. This would be true if the ailerons or the elevators were blown in order to increase effectiveness and would probably be most critical following an asymmetric propulsion failure. As such, it will be discussed at greater length in Section 8.

The following are the results of the simulation program which relate to definition of limiting flight conditions and how they appear to the pilot of a powered-lift vehicle.

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FINDING:

The effects of a 1 g stall at approach power may not be nearly as severe in powered-lift aircraft as in conventional aircraft.

DISCUSSION:

In the BR 941 simulation it was noted that the effects of a 1 g stall were relatively mild. The pilot felt no abrupt loss of lift and controls continued to be effective beyond $C_{L_{max}}$. This was experienced in both the simulator and in the actual aircraft. The reason for this is probably most strongly related to thrust effects on the air flow over the wing. The thrust effects prevent an abrupt flow separation and associated lift loss. Thus, even beyond stall the variations in C_L with angle of attack can be mild.

The implication is that the 1 g stall may not need to be considered as severe a limiting flight condition as in conventional aircraft. While flight path dynamics themselves are severely degraded at $C_{L_{max}}$, it may be that going beyond stall is not particularly hazardous so long as an abrupt loss of lift is not experienced. In any event, operating above the stall angle of attack serves no useful purpose and does cause some piloting difficulties. For example, increasing angle of attack increases airspeed and can cause a large, rapid increase in sink rate. Thus, sustained flight in this regime should be avoided, although a momentary excursion into it may not be hazardous.

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FINDING:

In the case of powered-lift aircraft it was found convenient to split limiting flight conditions into two categories: "soft" limits and "hard" limits.

DISCUSSION:

These two cases have been normally referred to as the V_{min} and the α_{max} limits. It may be, however, somewhat misleading to use these specific terms. Hence, in the following paragraphs the distinction will be made using the terms "soft" and "hard".

The hard limit was considered to be a point beyond which catastrophe was likely. Examples of this were considered to be loss of control in any axis or an abrupt force or moment change which could lead to a loss of control. Some examples of a hard limit would include:

- A sharp loss of lift following aerodynamic stall
- An uncontrollable nose slice or wing drop associated with stall
- Uncontrollable pitch up to a deep stall condition
- Severe aerodynamic buffet
- Stalling of an aerodynamic control surface.

The common element in each of these conditions is that they are unsafe to encounter.

It is significant that aerodynamic stall, per se, is not included in the above list. It was found that aerodynamic stall, under certain conditions, could be considered a soft limit. Such would be the case if stall were not accompanied by any of the large discontinuities in forces, moments, or control as mentioned previously. Hence the pilot would suffer only the degradation of normal flight path control because of near zero damping. Attitude control about all axes and even lateral flight path control would remain. Thus, recovery from such a soft limit would be easier and safer than recovery from a hard limit. If a hard limit were to occur prior to aerodynamic stall, any soft limit would have no significance.

Treatment of aerodynamic stall as a soft limit is an outgrowth of the previous finding. Because of the likelihood that powered-lift vehicles can exhibit relatively docile behavior at aerodynamic stall, it seems reasonable to define safety margins from an α_{max} which could be greater than that for $C_{L_{max}}$.

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FINDING:

There is some evidence that a limiting flight condition exists beyond $C_{L_{max}}$ which involves a flight path divergence.

DISCUSSION:

This limiting flight condition corresponds to the point at which a steady-state flight condition is no longer possible. Even though the pilot holds constant attitude and power, airspeed and sink rate continue to increase. In effect, this condition is a combination of pitch attitude and power for which there is no stable trim.

A flight path divergence was first observed in the BR 941 simulation. During stall demonstrations, the pilot slowly increased pitch attitude as he approach $C_{L_{max}}$ (at constant power). Further pitch increases would cause gradual sink rate and airspeed increases. Finally, he would reach a point where a slight pitch up would cause a rapid divergence in sink rate. Recovery was possible if it was initiated promptly. The procedure was to pitch down and add full power.

The cause for this divergence was not immediately diagnosed. It was first thought to be some deficiency in the model since the model accuracy at high angles of attack was somewhat questionable due to a scarcity of data. The condition was later recognized as a divergence which could occur in many powered-lift aircraft.

The most direct means found to define the point of divergence is to plot trim pitch attitude versus angle of attack for a constant throttle setting. Where the slope $\partial\theta/\partial\alpha$ is positive a normal stable trim condition is possible. If the slope becomes negative, then only an unstable trim is possible and a path divergence will occur if attitude and throttle are held fixed. An example is given in Figure 3-3 for the STOL-X simulator model. Note that the path divergence condition occurs beyond the point of aerodynamic stall, $C_{L_{max}}$.

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3.2 APPROACH TO AND RECOVERY FROM LIMITING FLIGHT CONDITIONS

One of the key differences between powered-lift and conventional aircraft is the rate at which limiting flight conditions are approached following a power reduction. In a conventional jet airplane, retarding the throttle does not significantly reduce lift but does cause the aircraft to decelerate. If the pilot holds the pitch attitude, the airspeed decreases and the sink rate increases. Thus he approaches the limiting flight condition (stall) relatively slowly as airspeed decays. But, for normal approach attitudes, even a power reduction to flight idle normally does not lead to a stall or even come close.

The situation is quite different in a powered-lift aircraft. A power reduction causes a large, immediate loss in lift. The angle of attack increases rapidly and a reduction to flight idle would, in all likelihood, take the aircraft into a limiting flight condition.

The key differences are then the magnitude and rate of safety margin reductions after reducing thrust. The conventional airplane is inherently more gradual and forgiving, while the powered-lift airplane may approach

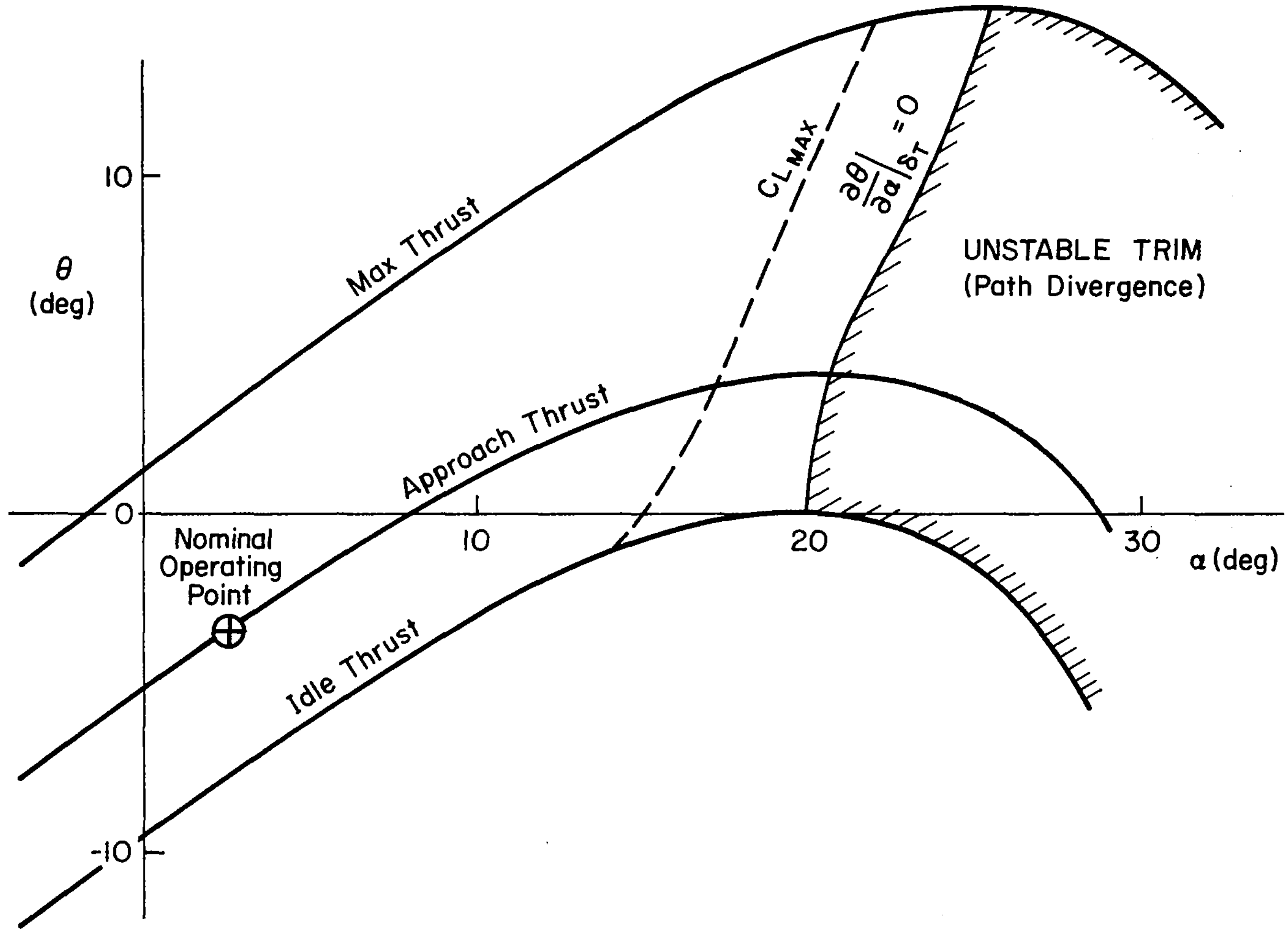

Figure 3-3: Region of Unstable Trim Conditions
(STOL-X)

limiting flight conditions nearly as quickly as the pilot can retard the throttle.

Approach to limiting flight conditions using pitch attitude, on the other hand, is more similar between powered-lift and conventional airplanes. A pitch increase of 1 deg will cause about the same speed decay whether a powered-lift or conventional aircraft (see the example time responses in Figure 5-4). While there is an accompanying vertical acceleration to serve as a warning of the speed reduction, the acceleration cue in a powered-lift aircraft is smaller in magnitude and of shorter duration. For similar vertical accelerations, the speed decay would be much more rapid in a powered-lift aircraft. Thus, the acceleration cue would be a less useful warning.

The simulation findings in this area are relatively qualitative. There was no formal variation of the parameters which determine how a powered-lift airplane approaches limiting flight conditions. Rather, we simply viewed the characteristics of specific simulation models, in particular the BR 941 and the STOL-X. In all of the cases considered there was no loss of attitude control at high angles of attack. The intent was to view only flight path problems related to the use of powered-lift. The following findings, then, are limited to those characteristics which are likely to be unique to powered-lift airplane designs.

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FINDING:

Aircraft behavior during the approach to limiting flight conditions using one control can depend upon the specific setting of the other control.

DISCUSSION:

The approach to a limiting flight condition via a power reduction depends upon what attitude is held, or the approach to a limiting flight condition using an attitude increase depends upon the power setting. The case of power reduction is the more interesting of the two. The variation in behavior depending upon pitch attitude held is illustrated in Figure 3-4.

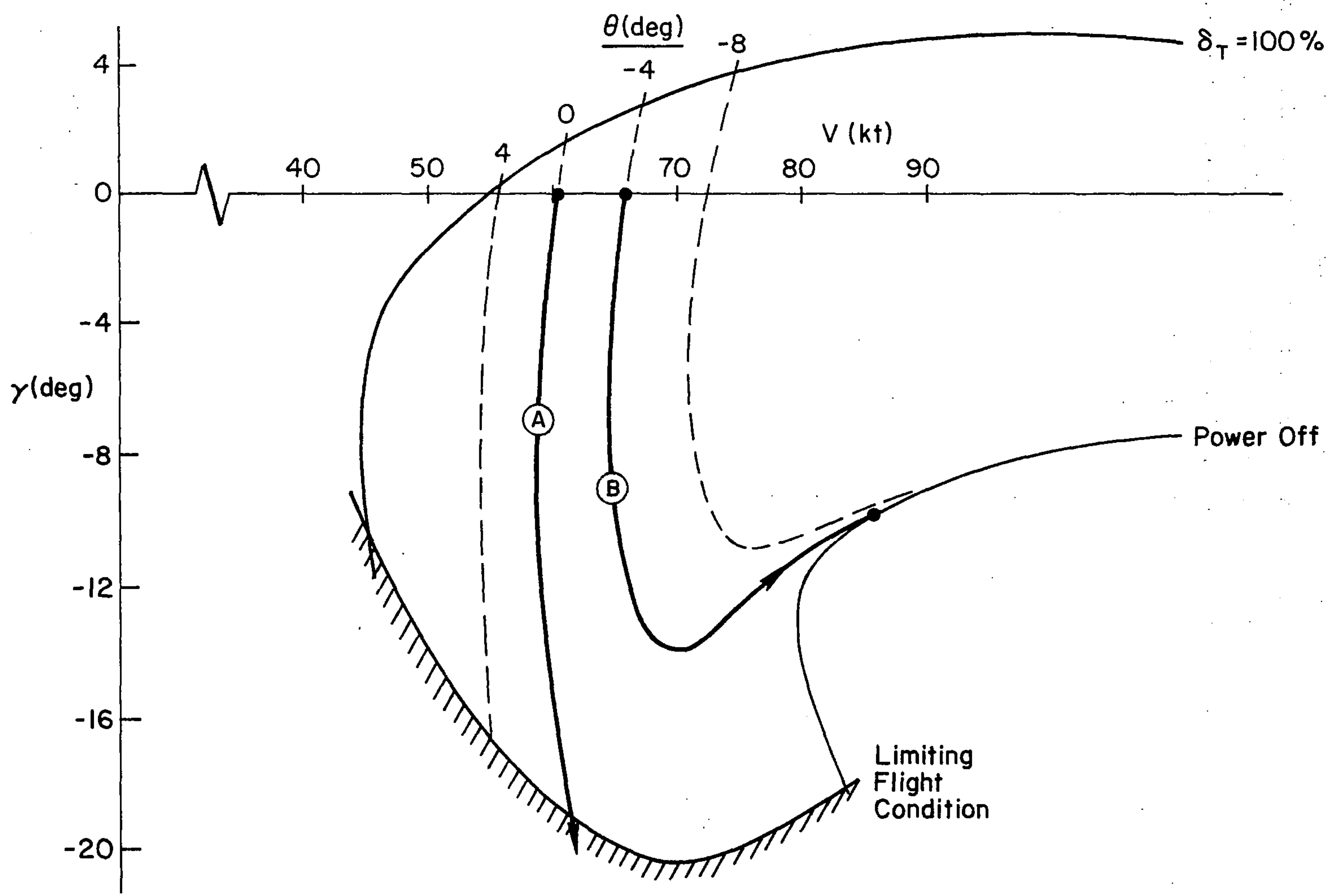

Figure 3-4: γ-V Trajectories for Approach to α_{max}

If the pilot of this aircraft were to hold the nose level ($\theta = 0$) and slowly reduce power he would follow the trajectory labeled A. The limiting flight condition, as shown by the shaded boundary, would be approached directly and nearly as rapidly as the rate at which the pilot retards the throttle. If, on the other hand, the pitch attitude were held at 4 deg nose down, then a power reduction would result in following the trajectory labeled B. Instead of approaching the limiting flight condition, the aircraft would, at some point, begin to accelerate and end in a gliding condition safely above the power-off stall. This general behavior would hold fairly independently of how slowly or rapidly the throttle is moved.

The trajectories shown in the figure actually represent the STOL-X simulation model. The behavior described was clearly observable by the subject pilots and the general behavior is likely to be present in other powered-lift designs. The main point, though, is that the way in which limiting flight conditions are approached can depend upon how the flight path controls are set.

The corresponding case for approach to limiting flight conditions using attitude with power fixed is somewhat trivial. The main feature is simply that the stall speed or maximum angle of attack which defines the limiting flight condition can vary significantly with power setting.

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FINDING:

Approach to a limiting flight condition appears to be more rapid and hazardous with power reduction than with a pitch up.

DISCUSSION:

In most of the powered-lift designs simulated in this program the subject pilots noted that following an abrupt power reduction the angle of attack began to build rapidly and that if the attitude were sufficiently high an aerodynamic stall occurred almost immediately. As mentioned previously, an abrupt approach to stall by pitching up is accompanied by a substantial increase in normal acceleration. At the same time there is an opposing force from the attitude controller. If power is rapidly reduced, the pilot experiences only a decrease in normal acceleration. He has no corresponding force cues in the throttle controller. This problem has a direct impact on the requirements for warning and deterrent to limiting flight conditions. A discussion on this will be continued in the next subsection.

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FINDING:

Recovery using a power application and holding attitude is usually effective.

DISCUSSION:

The subject of recovery from limiting flight conditions was studied most during the BR 941 simulations. For that particular simulation model it was found that, where the throttle is an effective device for rapidly approaching limiting flight conditions, it is conversely an effective device for reversing the process and effecting a recovery. The effectiveness of power as a recovery device does, however, depend on how deeply the limiting flight condition has been penetrated. In the case of the BR 941 simulation model there was a flight path divergence, as mentioned previously, which could preclude successful recovery unless initiated promptly. In an actual airplane, this same condition could be present. However, there could also be a serious degradation of the propulsion system in the vicinity of the limiting flight condition. For example, at a high angle of attack an actual airplane may suffer a propulsion system failure because of inadequate inlet air flow and advancing throttles would have no effect. Under such conditions a recovery using a pitch down would be the only alternative, providing that pitch control were still effective.

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FINDING:

Under some special conditions, recovery using power could aggravate a limiting flight condition situation.

DISCUSSION:

This is a feature that was observed in the STOL-X simulation model. It was a subtle feature of that model but it is mentioned here because it could exist in other powered-lift airplane designs and possibly be more prominent.

The condition referred to above is illustrated in Figure 3-5. This is a γ - V plot similar to the previous one. Consider the case of trajectory B in which attitude is held at 4 deg nose down and power is reduced to idle. As noted previously, this ends in a power-off glide above the stall speed. If, in that gliding condition, the nose is then raised slightly to level and power is then increased a small amount the aircraft will follow trajectory D which, in turn, approaches the limiting flight condition. Thus, instead of improving the situation, the power increase actually aggravates it.

The physical explanation for such unusual behavior is related to the effective thrust inclination at a very low thrust setting. In any jet flap vehicle with near zero blowing, the effective thrust inclination is dominated by the induced drag term. Hence, increasing lift slightly with the thrust is nearly the same as increasing lift slightly by pitching up. Either way, the airplane is slowed and sink rate is increased. If this persists an aerodynamic stall could occur. Normally, however, the application of power would be large enough to give a more forward component of effective thrust, thus reducing sink rate and returning to the normal approach condition.

It appears that this very special case of aggravated recovery would occur for a minimum power setting, a speed just above aerodynamic stall, and a very slow application of power while holding attitude. The tendency would be significantly reduced, if not eliminated, by pitching the nose over or more rapidly advancing the throttle to maximum power. Since this particular problem was never encountered in any of the simulated landings, its seriousness is purely speculative, but it could be more severe for another powered-lift design.

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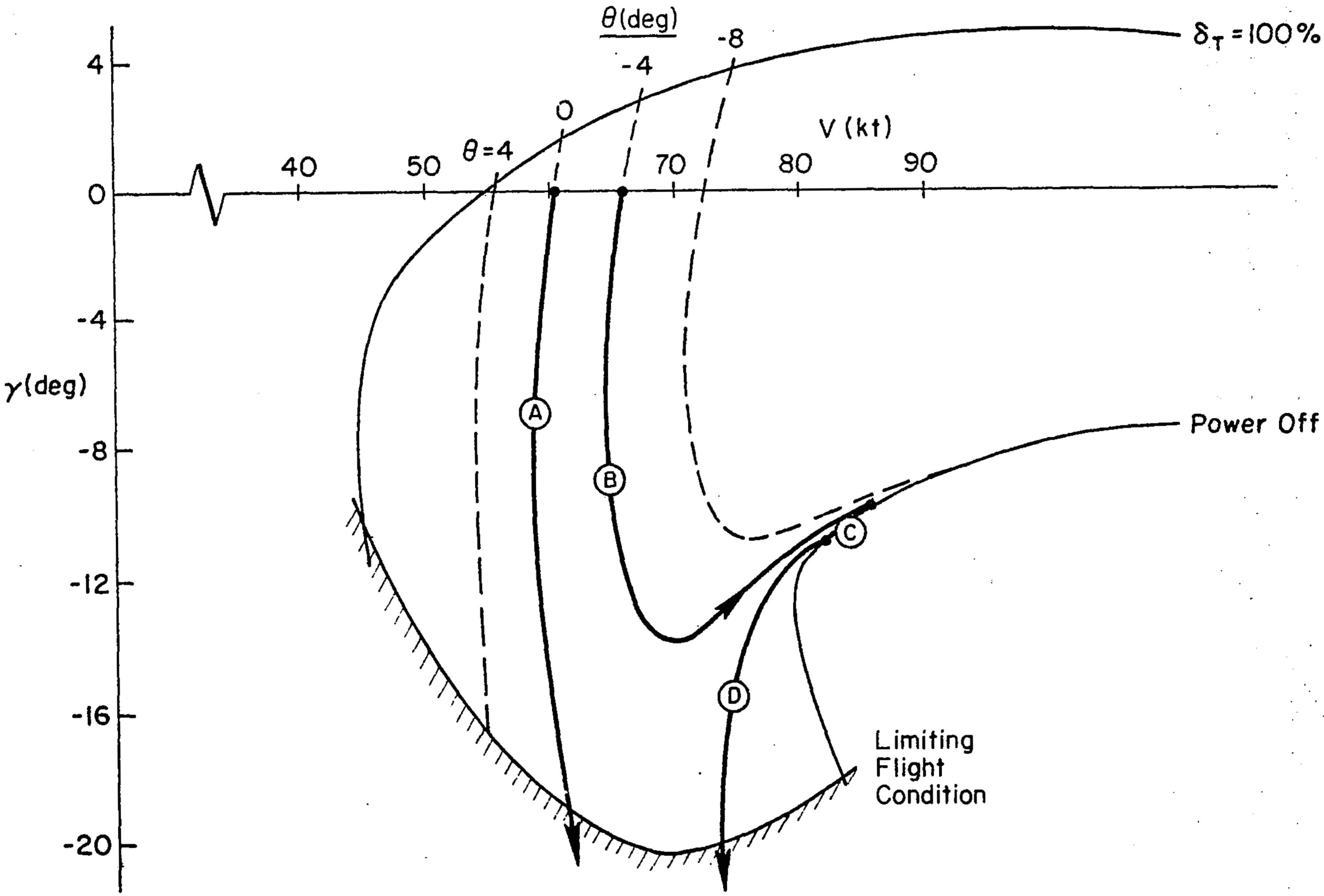

Figure 3-5: γ-V Trajectories for Approach to α_{max} From Low Thrust Condition

3.3 WARNING AND DETERRENT TO LIMITING FLIGHT CONDITIONS

From the point of view of current practice, it is necessary, in powered-lift aircraft, to have (i) a warning that a limiting flight condition is being approached, and (ii) a deterrent to encountering it. While the simulation program did not include a formal investigation of limiting flight condition warning and deterrent, there were some resulting ideas which are worth reporting.

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FINDING:

Warning can be characterized as an alert to the pilot that the basic control characteristics are starting to change and this warning can be made without interfering with operation of the aircraft.

DISCUSSION:

Typical examples of warning to limiting flight conditions are, for example, a stick shaker or the encounter of light aerodynamic buffet. In the case of powered-lift aircraft warning is always appropriate at the soft limit (usually V_{min}) even though the aircraft could go to a higher angle of attack without hazard, i.e., when the hard limit is beyond the aerodynamic stall. A warning should always be made at (or before) stall because it signals the loss of heave damping and the associated fundamental change in control of flight path.

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FINDING:

A deterrent, as opposed to a warning, should be associated with a potentially hazardous event and should interfere with the pilot's action in continuing past the limiting flight condition.

DISCUSSION:

Typical examples of current deterrents to limiting flight conditions are: stick pushers, heavy aerodynamic buffet, or a large nose-down pitching moment. For powered-lift aircraft the same approach to deterrent devices seems reasonable. An automatically varying throttle stop might be particularly effective in view of the likely use of throttle to control flight path.

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FINDING:

Warning and deterrent devices for approach to limiting flight conditions by reducing power may require more sophistication than currently-used devices.

DISCUSSION:

This is primarily because of the potentially rapid approach to limiting flight conditions from a power reduction as was described previously. In the STOL-X simulation, a column shaker and pusher were used as warning and deterrent respectively. The shaker was actuated at an angle of attack 5 deg prior to an assumed α_{max} and the pusher was actuated 2 deg prior to that assumed α_{max}. This warning and deterrent combination was reasonably effective for approach to limiting flight condition via pitching up, but was completely ineffective for protection against a rapid power reduction. One pilot, who evaluated this system, noted that even a slow throttle closure caused column shaker, pusher, and encounter of α_{max} in rapid succession. With a rapid throttle closure, catastrophe was virtually instantaneous. He felt that throttle closure inhibiting would seem to be the only solution to this situation, however, this might also interfere with normal pilot use of the throttle during an approach.

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SECTION 4

SAFETY MARGINS; APPROACH AND LANDING

This section addresses the subject of safety margins for powered-lift aircraft operating in the approach and landing flight phases. The use of powered lift poses significant complications in establishing safety margins compared to conventional aircraft. In the first part of this section, these complications will be described and discussed to set the stage for the simulator findings reported in the second part.

A safety margin is the separation between a given operating point and a limiting flight condition. The purpose of safety margins is to prevent excursions into limiting flight conditions. The margins must tolerate flight condition excursions due to external disturbances and maneuvering by the pilot, as well as some reasonable variations and abuses in the nominal flight condition.

For any aircraft, angle of attack and airspeed are the primary flight condition variables. Angle of attack is the best measure of stall proximity for accelerated and 1 g flight. Airspeed is important because if it drops below the stall speed, 1 g flight is not possible at any angle of attack. Therefore, the important safety margins are the angle of attack and airspeed margins.

The things which affect airspeed and angle of attack, and thus their respective margins, are the pilot's controls (attitude and power), and external disturbances composed of vertical and horizontal gusts. This is shown schematically in Figure 4-1. Note that margins might be described in terms of these "input quantities" as well as the "outputs", angle of attack and airspeed. Another way to put this is that one can speak of margins in terms of how much pilot input or atmospheric disturbance can be tolerated, or in terms of airspeed and angle of attack margins remaining. Naturally, the dynamics of the airplane and the piloting techniques used

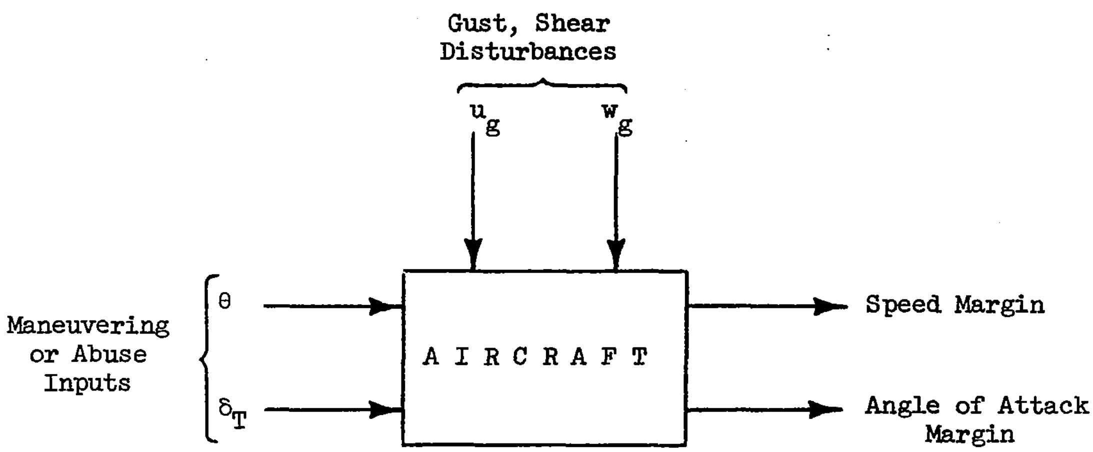

Figure 4-1: Schematic of Safety Margin Influences

to fly it are important in determining the relationship between the input controls and disturbances, and the resulting airspeed and angle of attack. The following paragraphs will describe these relations, first for conventional aircraft, then for powered-lift.

For conventional aircraft the main safety margin qualtity is expressed in terms of approach speed relative to a power-off stall speed or $\frac{V_{app}}{V_{min}}$*. We use the symbol M to represent margin, and M_V to represent specifically relative speed margin. The expression of safety margins in terms of a simple relative speed margin apparently has been adequate for virtually all conventional transport aircraft designs. We feel that the reason for this is a strong implied relationship between speed margin and other margin quantities such as angle of attack, horizontal and vertical gusts, and lift margins. The basis of these implied relationships are several aerodynamic and geometric quantities which tend to be relatively invariant. The following is a brief discussion of these relationships in preparation for trying to deal similarly with powered-lift aircraft.

For conventional jet transports, most of the implied margin relationships can be derived directly from a plot of lift coefficient versus angle of attack. This is shown in Figure 4-2. In this figure, the limiting flight condition is taken to be aerodynamic stall as characterized by $C_{L_{max}}$ and α. Safety margins, then, can be taken between stall and a given approach operating point. The implied margin relationships which shall be derived from this plot of C_L versus α will consist of angle of attack margin, M_α; lift margin, M_n; horizontal gust margin, M_{u_g}; and vertical gust margin, M_{w_g}. The independent variables to be used in each of these implied relations will be relative speed margin,

$$M_V \triangleq \frac{V_{app} - V_{min}}{V_{min}}$$

* Throughout this discussion we refer to the 1 g stall speed as V_{min}. For most jet transports this speed is considerably larger than the certified stall speed, V_{s_O}, because of certification test procedure used. For a typical jet transport, the FAR Part 25 limit of 1.3 V_{s_O} is about 1.22 times the 1 g stall speed according to Reference 22.

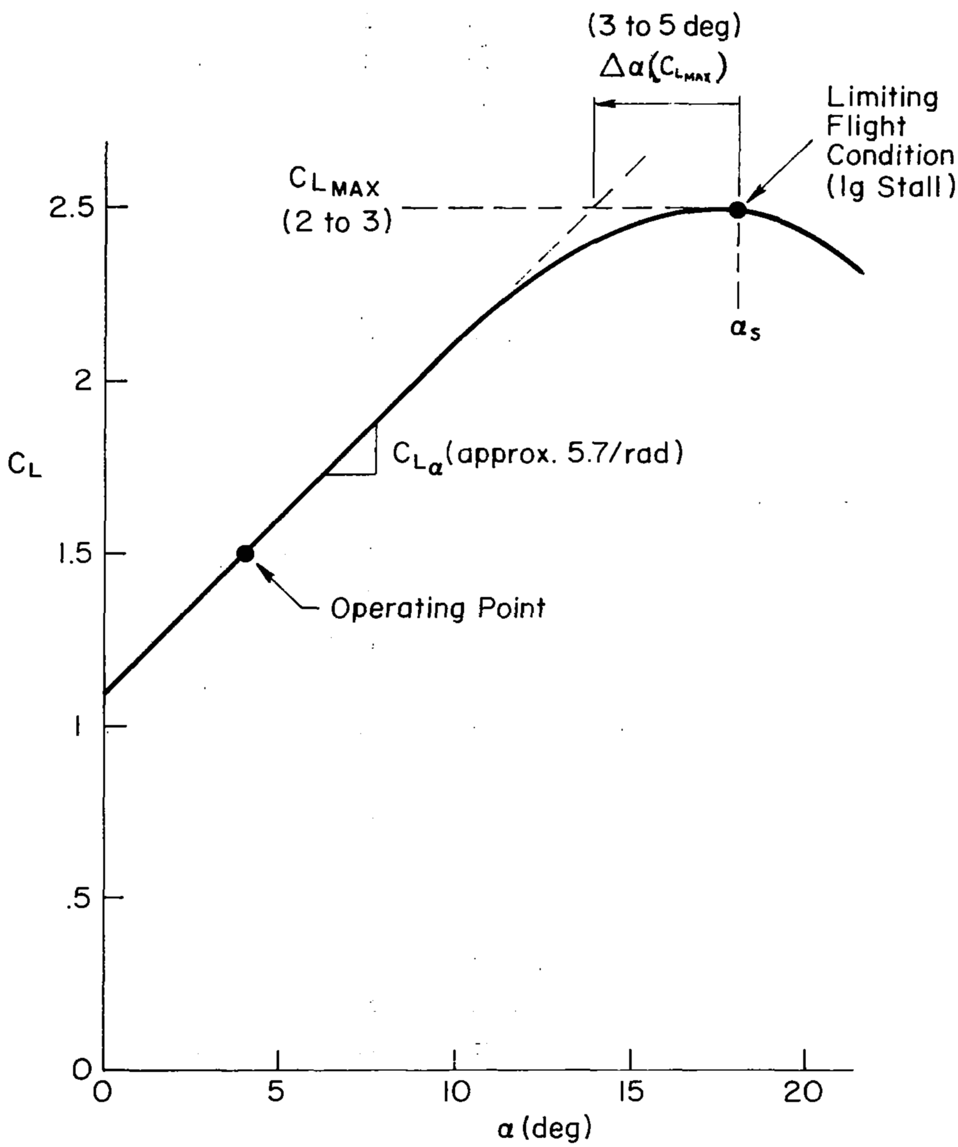

Figure 4-2: Typical C_L versus α
(Conventional Aircraft)

First consider the implied angle of attack margin for conventional airplanes. Referring back to Figure 4-2 we can see that the angle of attack margin between stall and a given 1 g operating point can be computed using the ratio of $C_{L_{max}}$ to $C_{L_{app}}$; the lift curve slope, C_{L_α}; and the factor describing the roundness of the lift curve near $C_{L_{max}}$, $\Delta\alpha(C_{L_{max}})$. Hence, angle of attack margin is:

$$\mathbb{M}_\alpha \triangleq \alpha_s - \alpha_{app}$$

and

$$\mathbb{M}_\alpha = \frac{C_{L_{max}}}{C_{L_\alpha}}\left[1 - \frac{1}{(1 + \mathbb{M}_v)^2}\right] + \Delta\alpha(C_{L_{max}})$$

Figure 4-3 shows the resulting angle of attack margin versus speed margin for a likely range of $C_{L_{max}}$, i.e., 2 to 3, and representative values of C_{L_α} and $\Delta\alpha(C_{L_{max}})$. This shows that for operation at minimum certification speed, 1.3 V_{s_o} (approximately 1.22 V_{min}), the angle of attack margin is 11 to 14 deg. Reference 22 shows that airline pilots usually approach with a 1 g speed margin more nearly 30%. At that speed, the angle of attack margin is roughly 12 to 16 deg. The point of this is that if a specific speed margin is imposed, a fairly well constrained angle of attack margin is implied, and it is not necessary to impose both speed and angle of attack margins. This same idea can be carried further with regard to other margins.

Implied safety margins in terms of atmospheric disturbances, in particular sharp-edged horizontal and vertical gusts, can be shown in a manner similar to that used above. It will be necessary, however, to consider the absolute airspeed in addition. The following margin relationships thus result:

$$\mathbb{M}_{u_g} = \mathbb{M}_v \cdot V_{min}$$

$$\mathbb{M}_{w_g} = (1 + \mathbb{M}_v)\, V_{min} \sin(\mathbb{M}_\alpha)$$

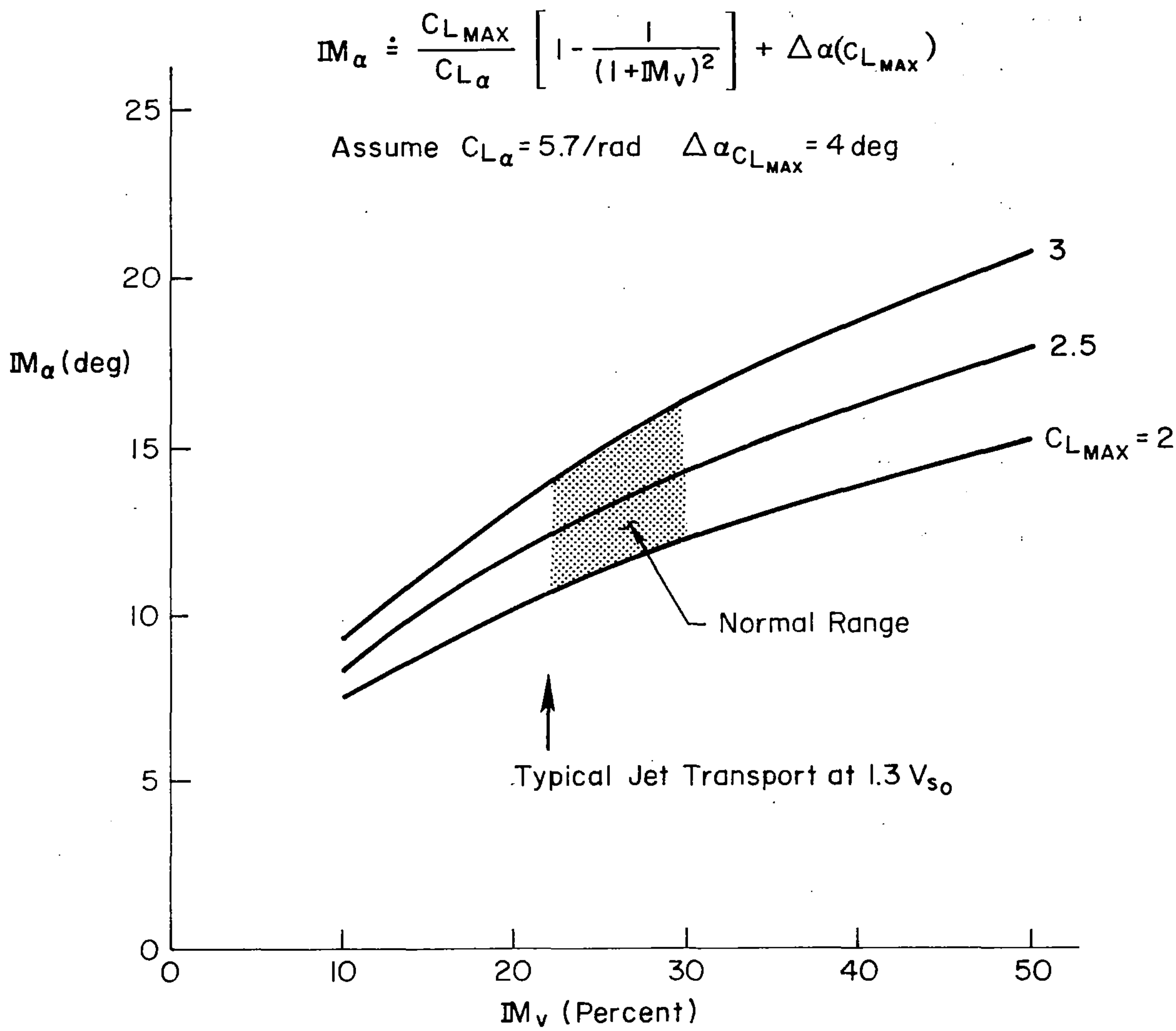

Figure 4-3: Angle of Attack Margin Implied by Relative Speed Margin (Conventional Aircraft)

Implied horizontal and vertical gust margins are plotted in Figures 4-4 and 4-5. Again, they are shown to be a strong function of relative speed margin. The curves assume a $C_{L_{max}}$ range of 2 to 3 and stall speeds ranging from 90 to 110 kt, which are representative of current jet transports. As before, the point of these plots is to show the strong implied relationship between various margins and the speed margin.

Finally, consider the safety margin which is indicative of the degree of maneuvering available to the pilot. This shall be termed lift margin, M_n, and can be defined as the maximum available normal acceleration resulting from a change in aerodynamic lift. This is, theoretically, the ratio of $C_{L_{max}}$ to $C_{L_{app}}$. If this ratio were 1.5 then the pilot could theoretically pull 1.5 g if he were to rotate the aircraft instantaneously to the point of aerodynamic stall. In practice this theoretical limit is never quite reached. The pilot cannot instantaneously rotate to stall angle of attack. Hence, there is some airspeed loss and thus a loss in the maximum absolute lift owing to the reduced dynamic pressure. To keep things simple, we shall neglect this factor. Then lift margin is approximately equal to:

$$M_n \doteq (1 + M_V)^2 - 1$$

This relationship is plotted in Figure 4-6.

To summarize, for <u>conventional transport aircraft</u> the establishment of a minimum <u>speed margin</u> effectively <u>implies</u> several <u>other margins</u>. These implied margin relationships are relatively invariant because the following are relatively invariant:

- Airplane geometry, in particular, aspect ratio
- $C_{L_{max}}$
- Limiting flight condition defined by stall.

Powered-lift aircraft have, in general, a different set of geometric and aerodynamic constraints. The result is a different set of implied

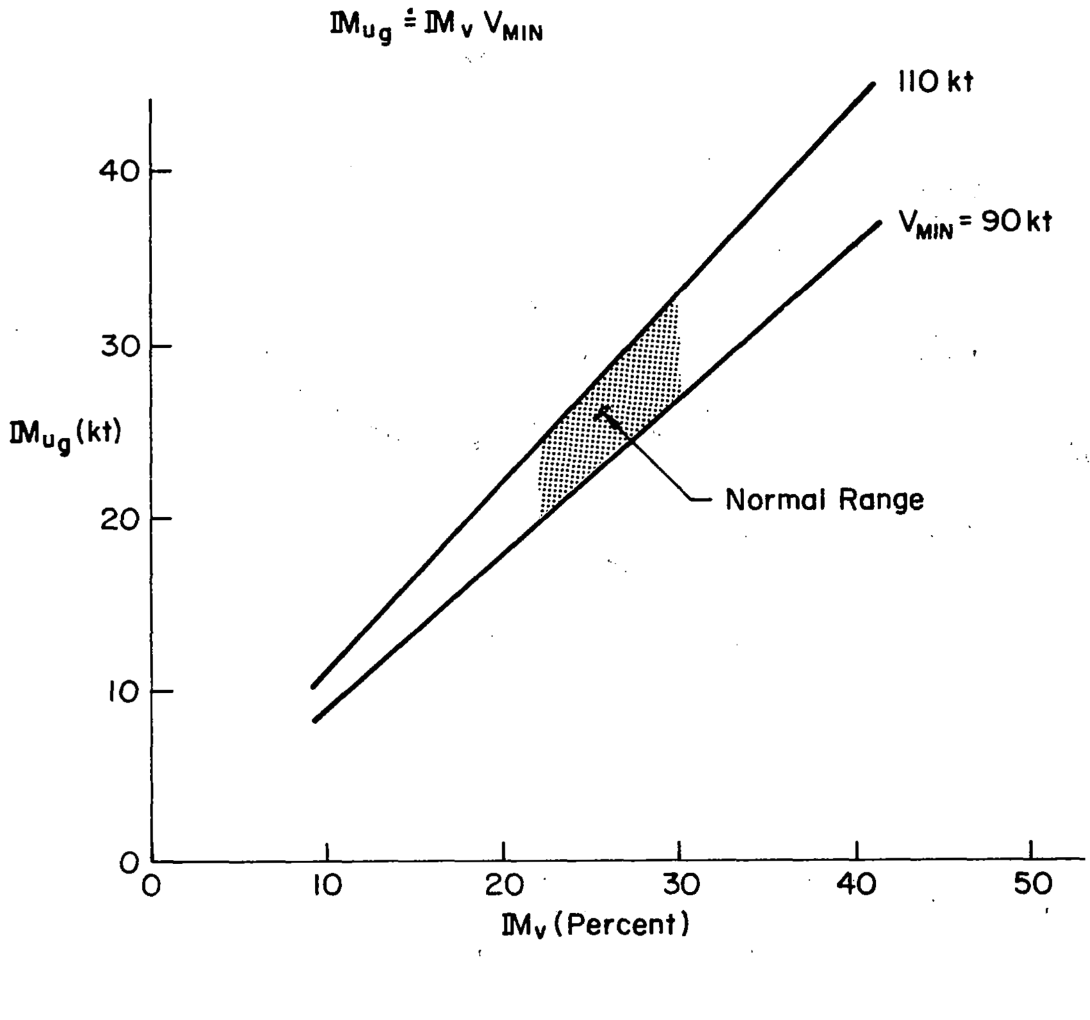

Figure 4-4: Horizontal Gust Margin Implied by Relative Speed Margin (Conventional Aircraft)

$$\mathbb{M}_{w_g} \doteq (1+\mathbb{M}_V)\, V_{MIN} \left\{ \frac{C_{L_{MAX}}}{C_{L_\alpha}} \left[1 - \frac{1}{(1+\mathbb{M}_V)^2} \right] + \Delta\alpha(C_{L_{MAX}}) \right\}$$

Assume: $C_{L_\alpha} = 5.7/\text{rad}$, $\Delta\alpha(C_{L_{MAX}}) = 4\,\text{deg}$

$2 < C_{L_{MAX}} < 3$, $90 < V_{MIN} < 110$ kt

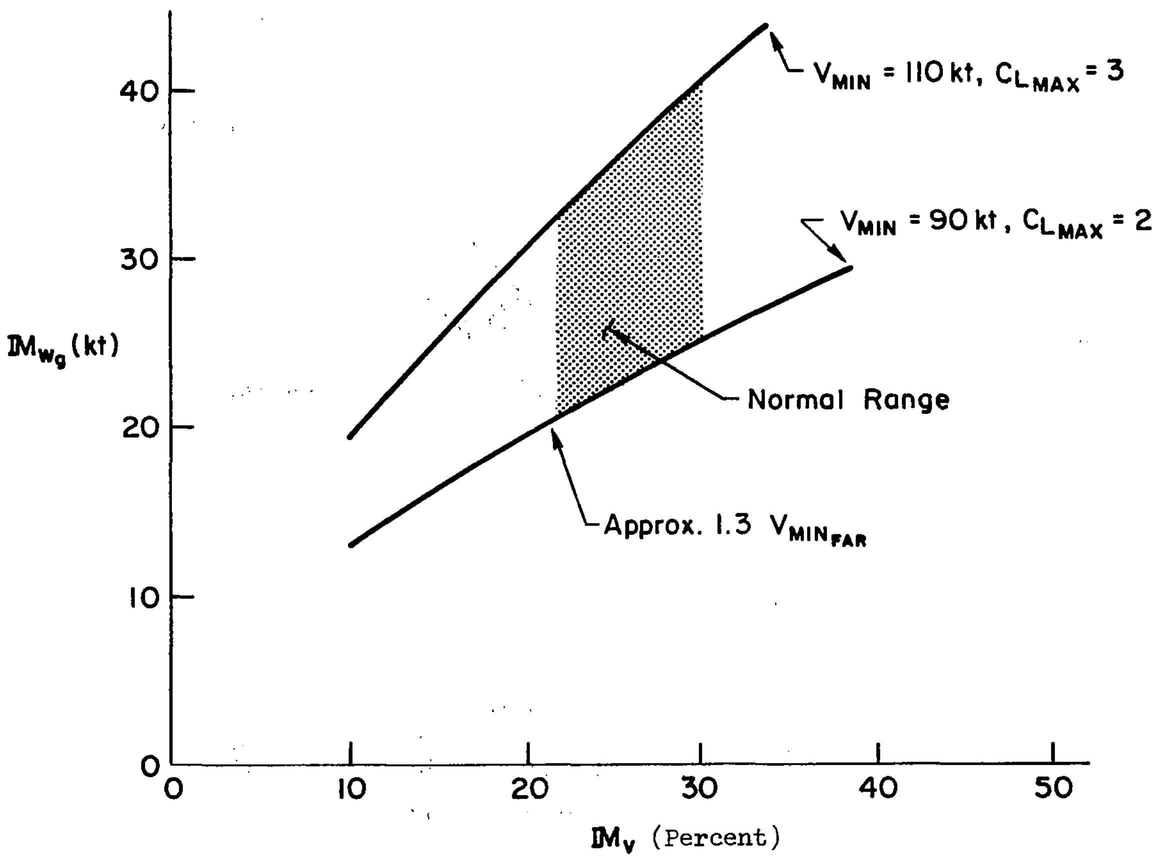

Figure 4-5: Vertical Gust Margin Implied by Relative Speed Margin (Conventional Aircraft)

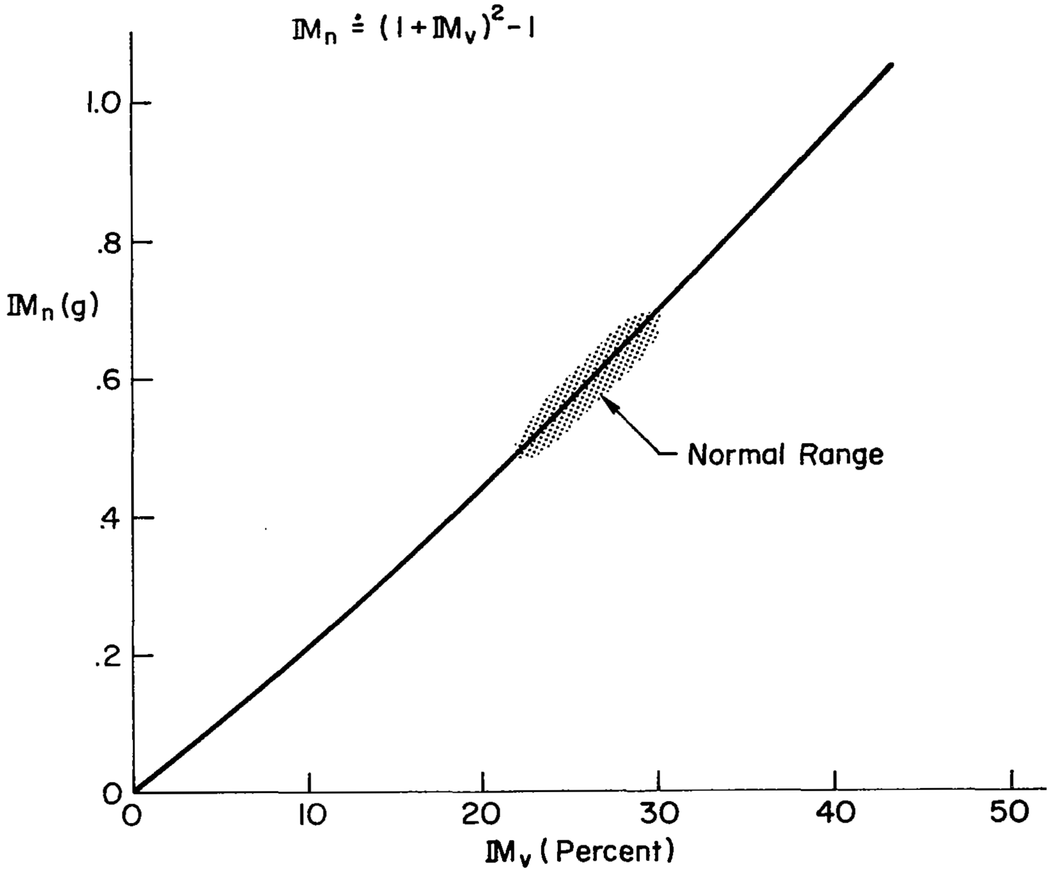

Figure 4-6: Lift Margin Implied by Relative Speed Margin (Conventional Aircraft)

margin relationships. While this simulation program did not try to take advantage of implied relationships for powered-lift airplanes, it is useful to discuss these briefly prior to presenting the simulation results. In fact, the simulation program tried to explore the explicit margin requirements for powered-lift aircraft.

The implied margin relations for powered-lift airplanes can be shown in precisely the same way that they were for conventional aircraft; that is, they can be based almost solely on the behavior of lift versus angle of attack. The added dimension will be, of course, the effect of thrust on lift. The following is a brief derivation of the various margin relationships for powered-lift airplanes.

<u>Relative Speed Margin</u>, $\mathbb{M}_V$:

$$\text{By definition, } \mathbb{M}_V \triangleq \frac{V_{app}}{(V_{min})_{app}} - 1$$

This is similar to the conventional aircraft speed margin definition except that the power setting is that of the trimmed approach condition rather than power off.

<u>Lift Margin</u>, $\mathbb{M}_n$:

Lift margin is defined at approach power also. It can be directly related to relative speed margin in at least two ways, but some measure of powered lift must be included. The metric used here is the parameter η_p, an indicator of the proportion of powered lift to total lift. (η_p is defined and discussed in Appendix A.)

The first way of approximating the lift margin is to assume a simple linear C_L versus α relationship between the trim condition and stall.

If $C_L = f(\alpha, C_J)$.

$$\Delta C_L = \frac{\partial C_L}{\partial \alpha} \Delta\alpha + \frac{\partial C_L}{\partial C_J} \Delta C_J$$

Since $C_J \sim \frac{T}{V^2}$

$$\Delta C_L = \frac{\partial C_L}{\partial \alpha} \Delta\alpha + \frac{\partial C_L}{\partial C_J} \left(-2\, C_J \frac{\Delta V}{V} + \frac{C_J}{T} \frac{\partial T}{\partial V} \Delta V \right)$$

For lift margin, ΔC_L is evaluated with $\Delta V = 0$

But, for relative speed margin, V is allowed to vary such that

$$\frac{\Delta C_L}{C_L} = -2 \frac{\Delta V}{V}$$

Thus for lift margin, $\Delta C_L = \frac{\partial C_L}{\partial \alpha} \Delta\alpha = C_{L_\alpha} \Delta\alpha$

and for speed margin*, $\Delta C_L = \frac{\partial C_L}{\partial \alpha} \Delta\alpha + \frac{\partial C_L}{\partial C_J} C_J \frac{\Delta C_L}{C_L}$

or $$\Delta C_L = \frac{\frac{\partial C_L}{\partial \alpha}}{1 - \frac{C_J}{C_L} \frac{\partial C_L}{\partial C_J}} \Delta\alpha = \frac{C_{L_\alpha}}{1 - \eta_p} \Delta\alpha$$

Each of these cases is illustrated in Figure 4-7. Based on the assumptions made,

$$M_n = \left[(M_v + 1)^2 - 1 \right] (1 - \eta_p)$$

This provides one implied margin relationship for powered-lift airplanes. Note that if $\eta_p = 0$, i.e., no powered lift, the expression reverts to that for conventional aircraft.

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* The thrust variation with speed, $\partial T/\partial V$, is assumed zero. This is a reasonable approximation for a jet engine.

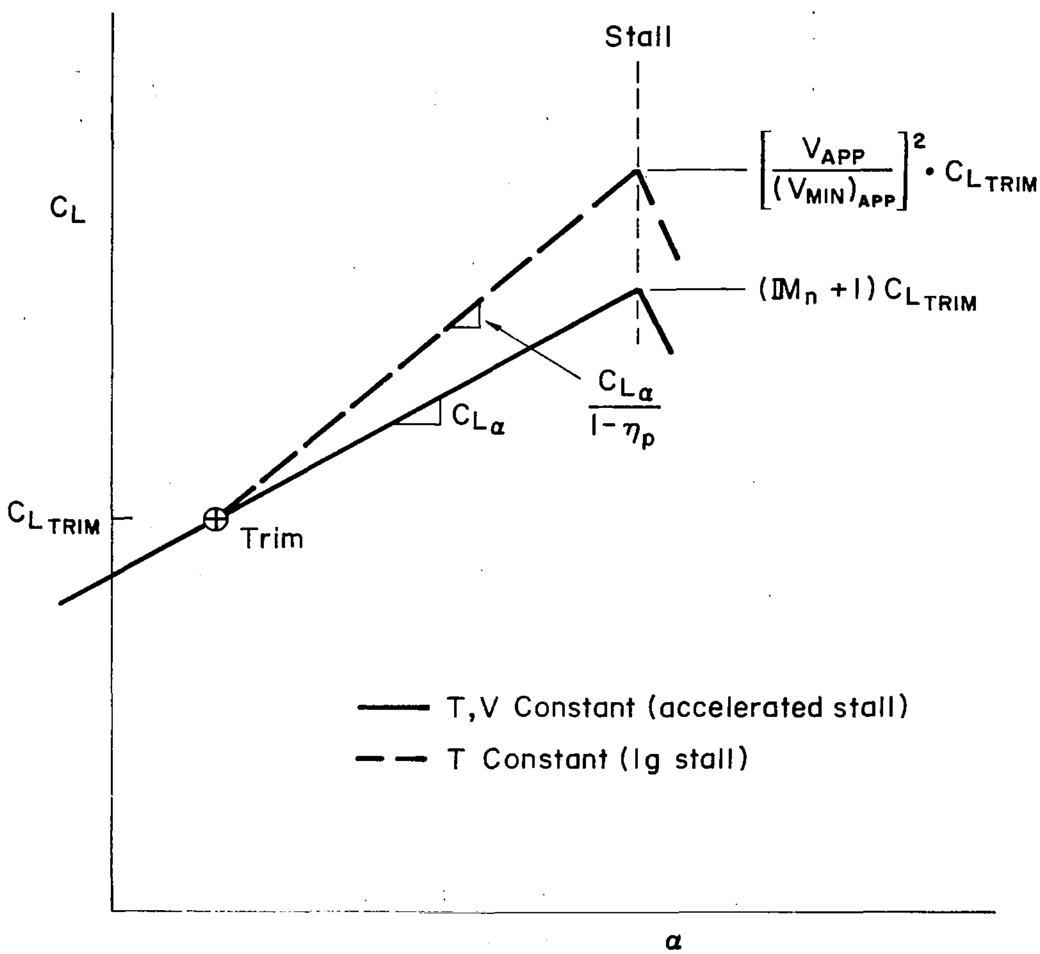

Figure 4-7: Lift Margin Relationship with Powered Lift

A second way of approximating the lift margin relationship is also worth noting. It may, in fact, prove to be a better approximation. This method is based on the sketch shown in Figure 4-8. If $C_{L_{max}}$ for an accelerated stall, i.e., $(M_n + 1)C_{L_{trim}}$, is related to $C_{L_{max}}$ for a 1 g stall, i.e., $(M_v + 1)^2 C_{L_{trim}}$, just as was done in the previous method, then:

$$M_n + 1 = \left(M_v + 1\right)^{2(1-\eta_{p_{stall}})}$$

Note that if $\eta_{p_{stall}}$ equals zero, M_n relative to M_v is the same as for a conventional airplane. If $\eta_{p_{stall}}$ equals 0.5, M_n equals M_v. If $\eta_{p_{stall}}$ equals η_p (which appears to be a good approximation) there is little difference between the implied lift margin relationships of either of the two methods described here for $M_v < 0.3$ and $\eta_p \lesssim 0.5$. This is shown in Figure 4-9.

Angle of Attack Margin, M_α:

The angle of attack margin relation, if based on lift margin, is unchanged from that for conventional aircraft:

$$M_\alpha \doteq \frac{M_n}{n_{z_\alpha}} + \Delta\alpha(C_{L_{max}})$$

$$\text{where} \quad n_{z_\alpha} = C_{L_\alpha}/C_{L_{trim}}$$

The values for n_{z_α} and $\Delta\alpha(C_{L_{max}})$ may differ, however.

Horizontal Gust Margin, M_{u_g}:

The horizontal gust margin differs from conventional aircraft only in that it is based on V_{min} at the approach power setting. Thus:

$$M_{u_g} = M_v \cdot (V_{min})_{app}$$

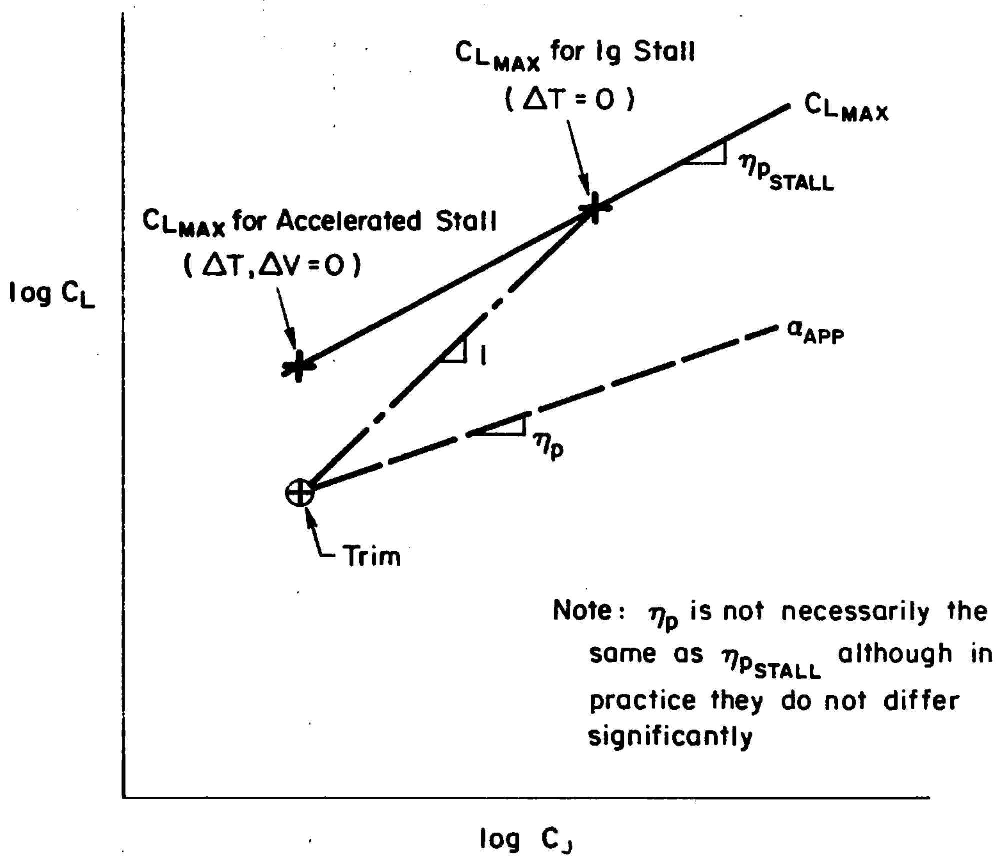

Figure 4-8: Lift Margin Relationship With Powered Lift
(Second Method)

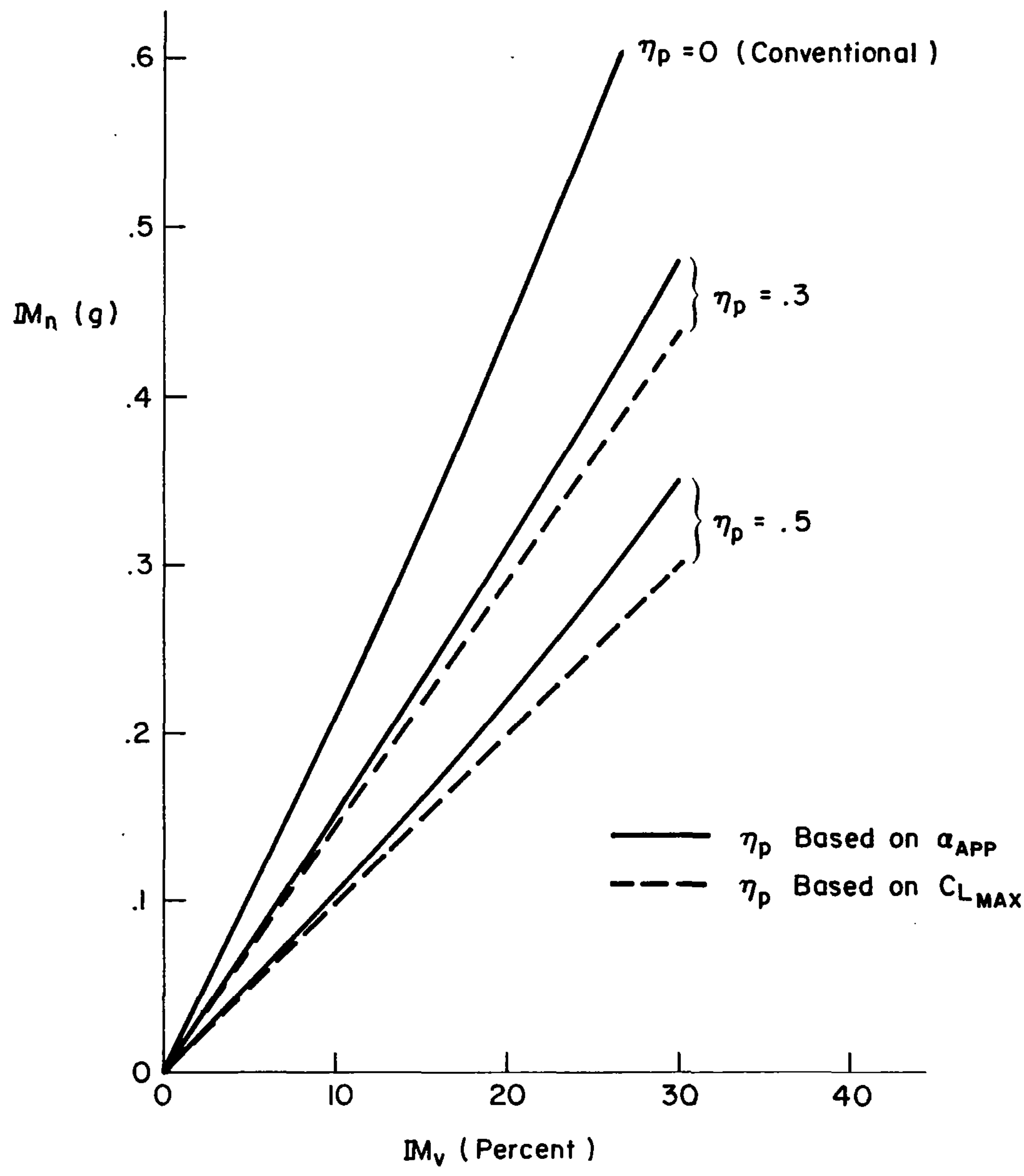

Figure 4-9: Lift Margin Implied by Relative Speed Margin (Powered-Lift Aircraft)

Vertical Gust Margin, M_{w_g}:

The vertical gust margin is identical to conventional aircraft if related to M_α:

$$M_{w_g} \doteq V_{app} \sin M_\alpha$$

These relationships are plotted against M_V in the following pages to show that there are corresponding implied margin relationships for powered-lift aircraft.

Figure 4-10 shows that the angle of attack margin, M_α, for powered-lift vehicles, appears to be in the same general vicinity as for conventional aircraft given the same relative speed margin. But it is important to note that with the use of powered lift there is more room for variation in η_p, $\Delta\alpha(C_{L_{max}})$, and n_{z_α}. Thus, although an implied margin relationship exists, it is less well defined.

The amount of M_α required for powered-lift may be less than for conventional due to use of a STOL piloting technique. This, in effect, is still another source of variation in the parameters which combine to form safety margin requirements. The trend of horizontal gust margin, M_{u_g}, for powered-lift aircraft is shown in Figure 4-11. This relationship is, of course, a unique function of M_V and V_{app}. M_{u_g} for slower aircraft must be inferior at the same relative speed margin.

It should be pointed out, though, that speed margins can be improved through use of powered lift. From the sketch of C_L versus α and ΔT in Figure 4-12 one can see how it may be possible to significantly improve relative and absolute speed margins when given credit for the ability to increase thrust. This is due to the vertical shift in C_L versus α for increased thrust. Note that this is useful only when there is a significant η_p and a rapidly available excess thrust, $\frac{\Delta T}{T_{app}}$.

Finally, in Figure 4-13 the implied margin relationship between M_{w_g} and M_V is shown. For flight path parameters assumed (i.e., η_p, $\Delta\alpha(C_{L_{max}})$,

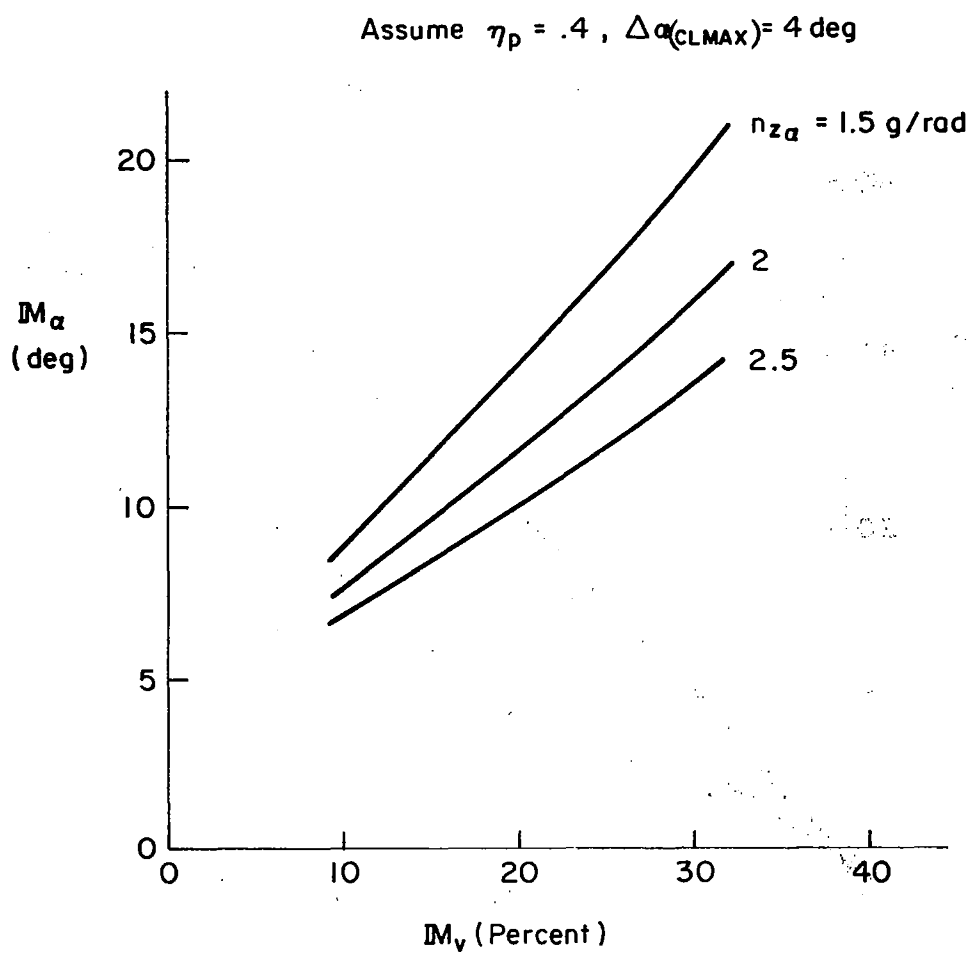

Figure 4-10: Angle of Attack Margin Implied by Relative Speed Margin (Powered-Lift Aircraft)

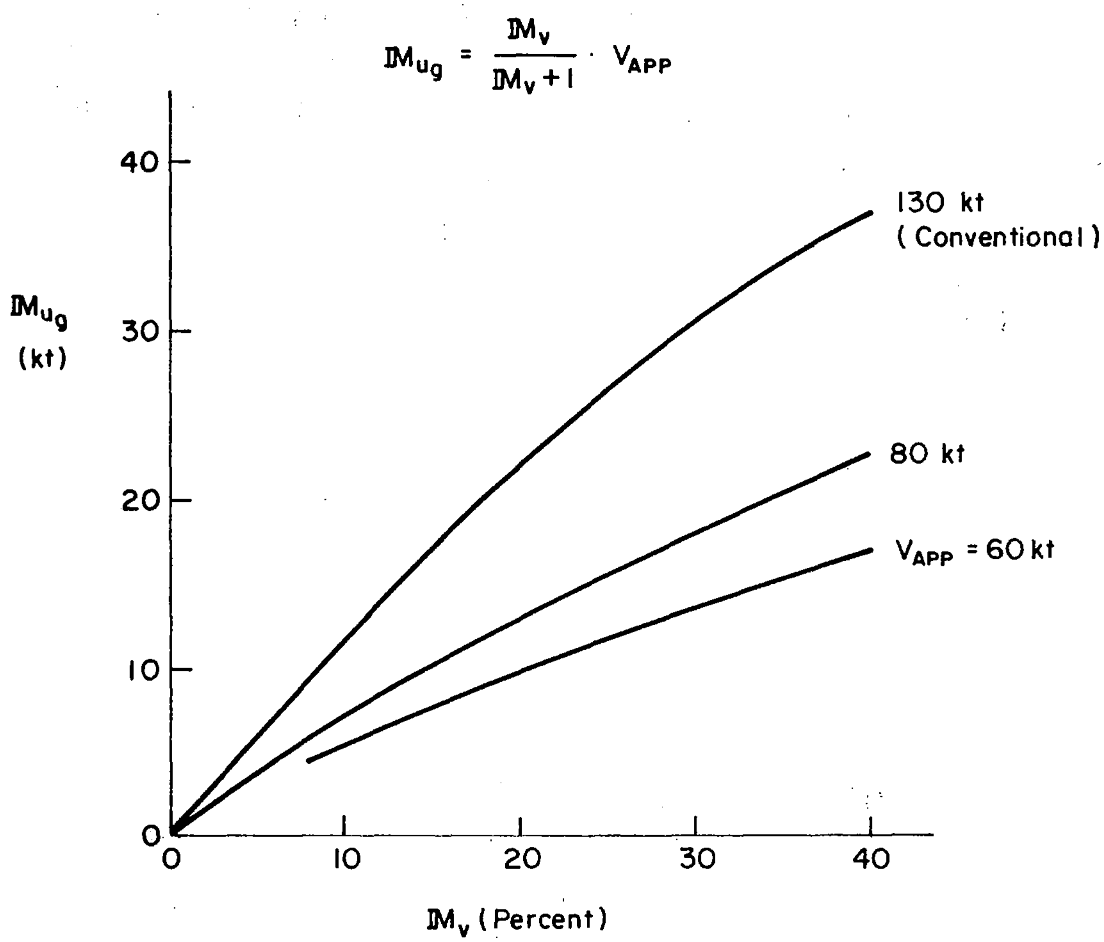

Figure 4-11: Horizontal Gust Margin Implied by Relative Speed Margin (Powered-Lift Aircraft)

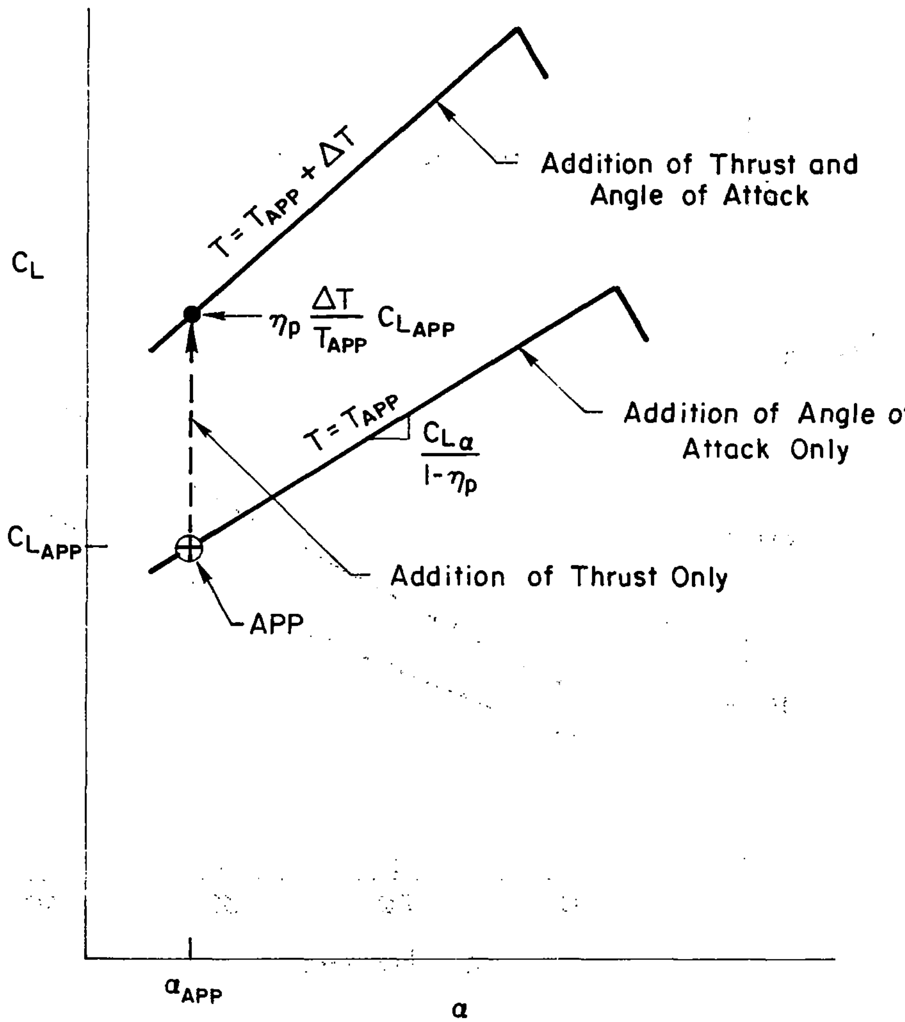

Figure 4-12: Effect of Thrust on Speed Margin

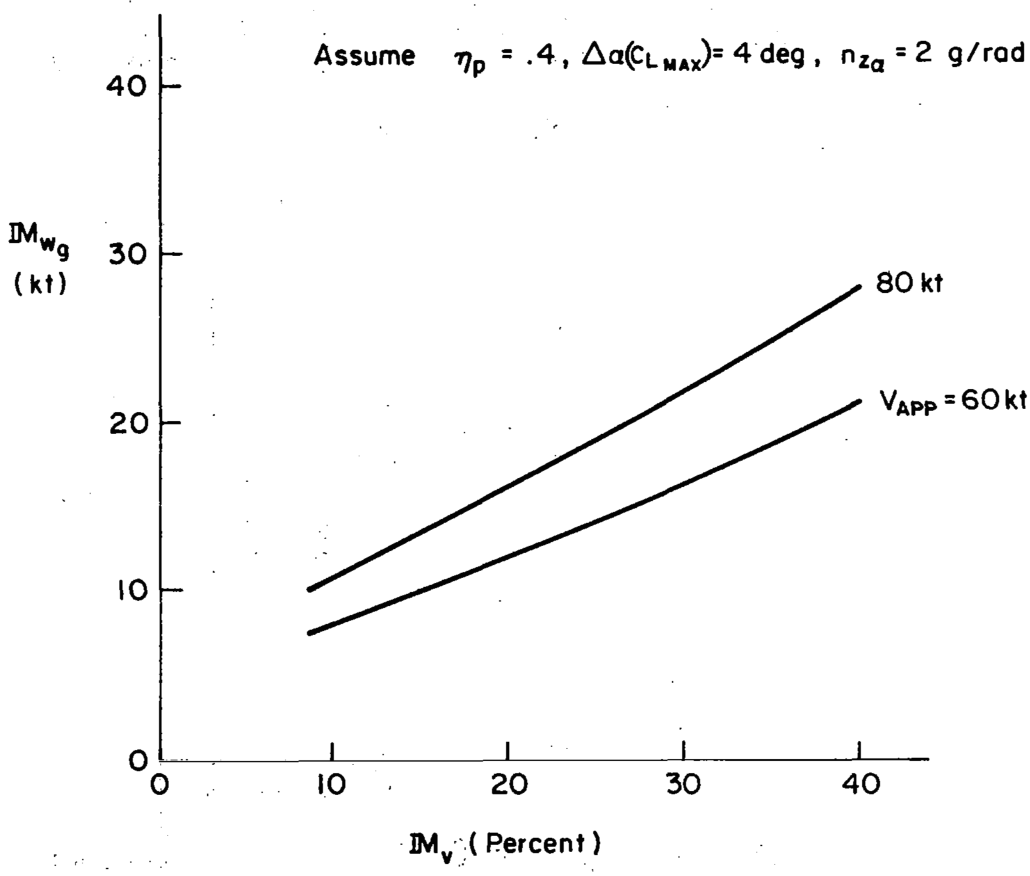

Figure 4-13: Vertical Gust Margin Implied by Relative Speed Margin (Powered-Lift Aircraft)

n_{z_α}, and V_{app}) the vertical gust margin is somewhat less than for conventional aircraft. The main source of this deficiency is the reduced approach speed. (Recall $M_{w_g} = V_{app} \sin M_\alpha$.) But a compensating factor may be present in this safety margin.

If the stall itself is non-hazardous then it may be possible to allow an α_{max} greater than α_{stall}. Thus, both M_α and M_{w_g} would increase correspondingly. (In the expressions given for M_α and M_{w_g} such an effect could be reflected by an increased $\Delta\alpha(C_{L_{max}})$.)

Note that the margin approximations given above are general, that is, they apply to both conventional and powered-lift aircraft. The main determining parameters are:

- n_{z_α}
- η_p
- V_{app}

The respective values of these parameters tend to set any implied relationships between relative speed margin and the other margins. These parameters are relatively restricted for conventional aircraft, hence there are relatively strong implied marginal relationships. In powered-lift aircraft, these parameters are less restricted in addition to having different ranges of values. The result is that implied margin relationships are looser and are of a different character than for conventional aircraft. Therefore, in trying to develop minimum safety margins for powered-lift aircraft it is more necessary to consider each of the various safety margins one by one as was done here.

During the simulation program safety margins were studied in two main ways. First, for several realistic powered-lift configurations the adequacy of their respective margins was noted in the context of routine approach and landing operations. Second, for a realistic general STOL configuration a scheme was devised to independently vary speed and angle of attack margins and examine them during routine approaches and landings. Thus, the results of this simulation include a set of specific configurations for which there is some pilot opinion relating to adequacy of safety margins. There was no attempt to sort out which specific margins may be deficient. It is not known if this is even possible. Due to the nature of the testing, it is not fair to characterize pilot opinion of the margins any more specifically than the terms of "adequate, marginal, or inadequate." The list of cases for which margins were considered is shown in Table 4-1. This table will be referred to in the findings which follow.

In addition to the simulator configurations of this program, it is helpful to also consider a number of actual flight tests in which safety margins were studied. Those used in the following discussion are listed in Table 4-2. Each case represents a minimum acceptable approach flight condition. Every attempt was made to make these cases as homogeneous and as accurate as possible.

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FINDING:

In this simulation program the smallest margins collectively associated with a clearly acceptable case were:

- Relative speed margin, 16%
- Absolute speed margin (horizontal gust margin), 9 kt
- Angle of attack margin, $\geq$ 11 deg
- Vertical gust margin, $\geq$ 13 kt
- Lift margin, 0.2 g

(These values do not necessarily correspond to the smallest margins taken on an individual basis as shown shortly.)

DISCUSSION:

This result was determined by considering the cases listed in Table 4-1. If all cases having comments indicative of being less than acceptable are set aside, then the case of the BR 941 operating with transparency at 65 kt has the set of smallest safety margins. It is somewhat coincidental that all of these minimum margins were for the same case but it is probably

TABLE 4-1

SAFETY MARGINS FOR SIMULATED CASES

CASE	V_{app} (kt)	α_{app} (deg)	M_V (%)	M_V (max) (%)	M_α (deg)	M_n (g)	M_{u_g} (kt)	M_{w_g} (kt)	COMMENT ON MARGINS	REFERENCE NUMBER
BR 941 95/0	60	+2	6	40	8		3	8	Inadequate	11
	65	-2	12	51	11		7	13	Marginal	11
95/12	55	+3	1	10	4	.01	1	4	Inadequate	11
	60	-1	8	20	8	.09	5	9	Marginal	11
	65	-4	16	30	11	.20	9	13	Adequate	11
AWJSRA 65/75	60	+7	26	69	17		13	18	Adequate	12
	65	+5	33	83	20	.37	16	22	Adequate	12
Generic STOL 203	75	+5	10	63	5	.04	7	6	Marginal to Inadequate	13
204	75	+5	15	72	11	.25	10	14	Marginal to Adequate	13
205	75	+5	22	81	14	.33	13	18	Adequate	13
210	75	+5	12	28	11	.24	8	14	Marginal to Inadequate	13
211	75	+5	18	32	14	.34	11	18	Marginal to Adequate	13
215	75	+5	21	87	10*	.33	13	12*	Inadequate	13
216	75	+5	32	97	15*	.50	18	19*	Adequate	13
STOL-X Baseline	63.5	+3	19	44	14*	.24	10	15*	Marginal to Adequate	14

* Margin from "catastrophic event"

TABLE 4-2

SAFETY MARGINS FOR SEVERAL POWERED-LIFT AIRPLANES

AIRCRAFT	V_{app} (kt)	M_v (%)	M_α (deg)	M_n (g)	M_{u_g} (kt)	M_{w_g} (kt)	REFERENCE NUMBER
YC-134A	60	15	10	--	10	13.5	23
NC-130B	67	16	8.5	.13	9	10	24
UF-XS	55	8	--	--	4	--	25
CV-48	55	17	--	.19	8	--	26
BR 941.01	60	15	--	.12	8	--	27
AWJSRA	65	--	15+	.15		17	28
Transall	81	25	9	--	16	13	29

-- Insufficient information
to compute margins

a direct result of the implied margin tendency described previously. Another fact which should be noted is that this case was examined extensively by a relatively large number of subject pilots.

The safety margins associated with this case should not be interpreted as the absolute minima required for powered-lift vehicles. They should, instead, be considered as a reasonably firm starting point. The effects of differences in piloting technique and the level of atmospheric disturbances need to be considered. This particular case will be used as the basis of a general discussion of safety margins for powered-lift aircraft. Data from other cases will be brought to bear for the purpose of making a more refined estimate of minimum acceptable safety margins.

If the above set of safety margins is, in fact, reasonably valid then it indicates that some powered-lift margins may be significantly less than those found in conventional jet transports, but, as we shall explain, they may not lack equivalent protection. First, however, consider the obvious differences. The relative speed margin of this case is one half to three quarters that of a conventional transport (if the conventional transport speed margin from 1 g stall is taken to be 22% to 30%). The absolute speed margin or horizontal gust margin is only one half that found in a conventional jet transport. Lift margin is less than one half, and vertical gust margin is about two thirds. Only the angle of attack margin is approximately equal to that found in a conventional jet transport. We shall try to explain the significance of these general trends in discussing the following findings.

Note that in general the margins associated with this case are similar in magnitude and relative proportion to many of the flight test cases in Table 4-2. This gives additional credence to the simulator data, but one basic point should be kept in mind. Much of the flight test experience may have been limited to lower levels of atmospheric disturbance than that used in this simulation program. Therefore it may not be entirely correct to make such a direct comparison.

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FINDING:

A speed margin of 15% $(V_{min})_{app}$ (M_V) or 10 kt (M_{u_g}), whichever is greater, appears adequate for the powered-lift vehicles and atmospheric conditions considered in this program.

DISCUSSION:

Based on the experiments conducted during this program, acceptable speed margins were found to be significantly lower than those of conventional aircraft. While these results do appear valid for the atmospheric conditions considered, the validity of the atmospheric conditions themselves are open to question. This opinion is based partly on the subsequent simulation results reported in Reference 21. Aside from this recent data, it is felt that any safety margins should ultimately be based on flight test experiments in actual atmospheric turbulence.

Before discussing the validity of the experiments conducted in this program, let us consider the results obtained. All the cases from Table 4-1 are plotted in Figure 4-14. This shows, first, a plot of relative speed margin versus a pilot opinion indication and second, a plot of absolute speed margin. These plots are intended to show the general trend with pilot opinion as well as some indication of a cut-off point. The flagged symbols are an indication of where other margin parameters are probably inadequate even though speed margins were reasonably large. The clearest grouping of the data appears to be for those cases which are better than "marginal." This leads us to interpret the minimum speed margins as being 15% $(V_{min})_{app}$ or 10 kt. These are clearly less than conventional transports which are 22 to 30% of V_{min} and about 20 kt.

It is not possible to say, categorically, that the minimum margins inferred do, in fact, provide protection equivalent to current safety margins of conventional aircraft. There are factors, however, which would explain why a generally lower level of speed margin could be satisfactory for powered-lift aircraft. These factors (explained below) are: the likely use of a STOL piloting technique, the ability to rapidly increase margins by applying thrust, and inherently higher speed damping in the bare airframe.

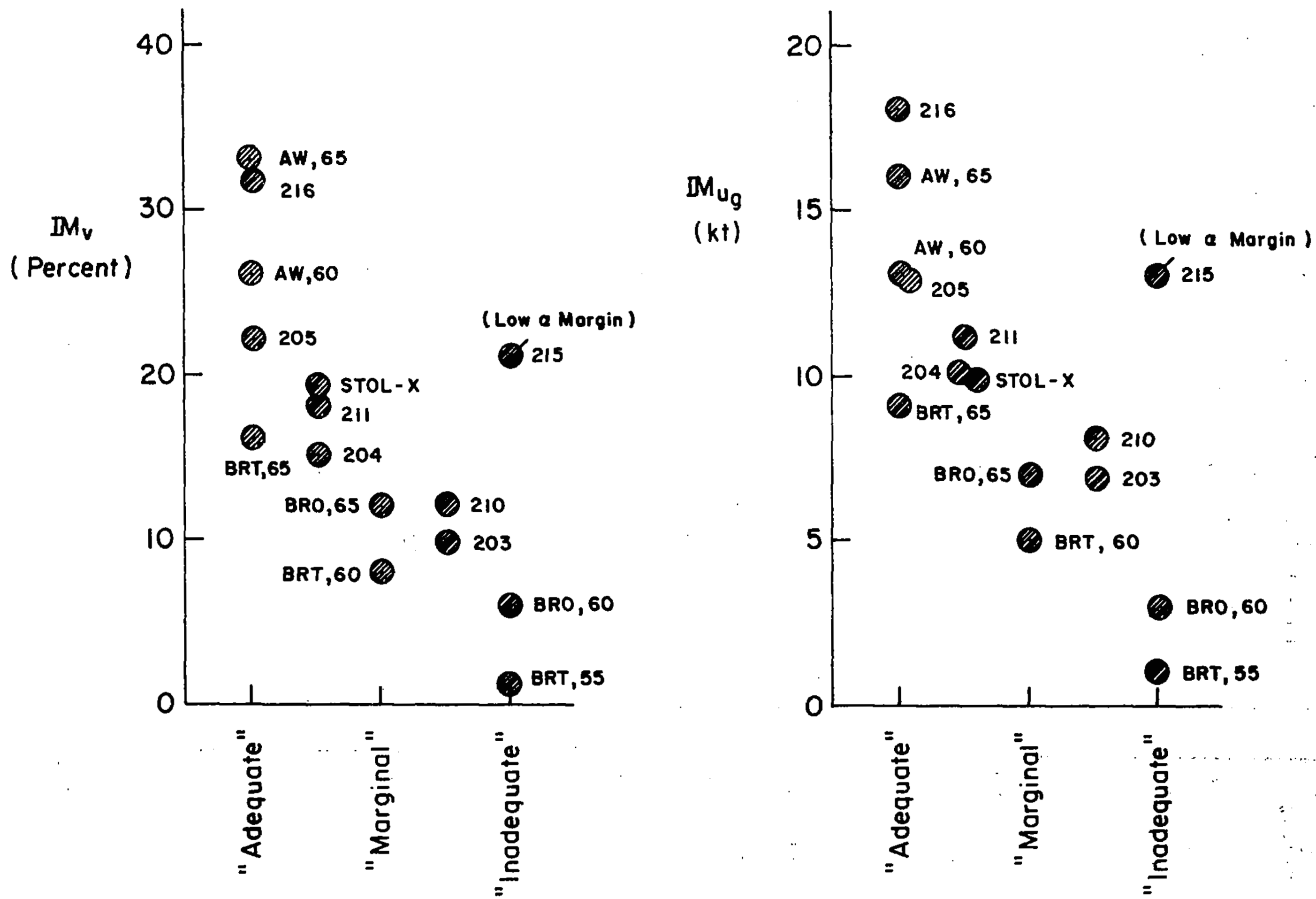

Figure 4-14: Speed Margin Trends

First consider the effect of piloting technique. In the case of conventional aircraft, the pilot is likely to maneuver using pitch attitude as primary control. This necessarily leads to some degree of airspeed excursion even though the pilot may be attempting to maintain airspeed by use of throttle. On the other hand, in a powered-lift aircraft, the pilot is likely to maneuver using the throttle (STOL technique) and, because of a largely vertical thrust orientation, such use of the throttle will not produce a significant amount of airspeed excursion. In addition, though, in the event a gust produces a large loss of airspeed rapid addition of thrust can significantly increase speed margins by a significant reduction in V_{min}. These ideas are shown schematically in Figure 4-15.

Another factor which acts to allow smaller speed margins is the inherently higher speed damping of powered-lift airplanes. The term speed damping refers to the specific restoring force due to an airspeed excursion. This effectively translates into a time constant for the exponential decay of an airspeed error provided pitch attitude and throttle are held fixed. Higher speed damping normally reduces this time constant (this will be explained in detail in the section on approach vertical path dynamics). Where a conventional aircraft is likely to have an airspeed decay time constant of approximately 15 sec, a powered-lift aircraft may have a time constant of less than half that. In effect, this is a measure of the convective tendency of the airframe without the action of the pilot. The effect of speed damping on airspeed excursions with throttle and attitude held fixed is shown by the following expression:

$$\frac{\sigma_{u_a}}{\sigma_{u_g}} \doteq \frac{1}{\sqrt{1 + \frac{1}{T_{\theta_1}} \frac{L_u}{V}}}$$

where

σ_{u_a} is the RMS airspeed excursion

σ_{u_g} is the RMS horizontal gust

L_u is the horizontal gust scale length.

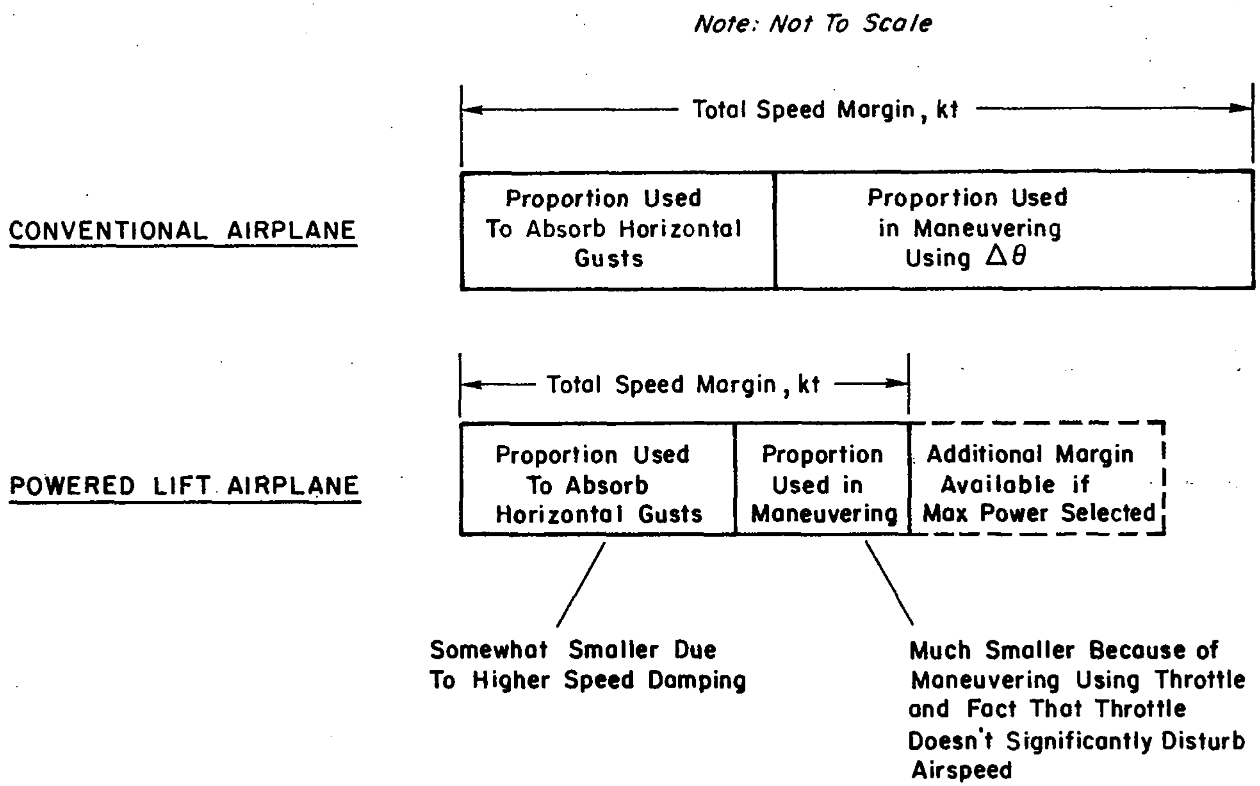

Figure 4-15: Speed Margin Considerations
(Conventional versus Powered Lift)

In most cases, this expression reveals a tendency for powered-lift airplanes to have somewhat smaller absolute speed excursions than convention airplanes.

Unfortunately, the factor which plays the biggest role in determining minimum speed margins may be inadequately defined. This is the atmospheric disturbance itself and, in particular, horizontal gusts and wind shear. The aspect which is not well known but is crucial for determination of safety margins is an appropriate relationship between atmospheric disturbance and its probability of occurrence. The speed margins determined in this program must be viewed as conditional to the atmospheric disturbances used in this program.

It is fair to say that the speed margins determined here are heavily weighted by the standard random atmospheric turbulence model used. While this level of turbulence seemed severe to the subject pilots, it may not have adequately addressed the very large wind shears which have occurred from time to time in airline operations.

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FINDING:

An angle of attack margin of 14 deg appeared adequate for the powered-lift vehicles considered in this program.

DISCUSSION:

Based on an experiment conducted during the first Generic STOL simulation program (Reference 13), there was evidence that some angle of attack margin was required independent of speed margin. The device used to reveal this was an unusually sharp break in lift at stall as opposed to the normally rounded break. This is illustrated in Figure 4-16. Thus, for any given speed margin there was correspondingly less angle of attack margin than for a more rounded lift curve. The specific cases which were involved in this comparison were: 203, 204, and 205 versus 215 and 216. The first group possessed the more rounded break at stall while the last two had a sharp break. Cases 205 and 215 both had a speed margin slightly in excess of 20%. The first was judged clearly adequate, the second, clearly

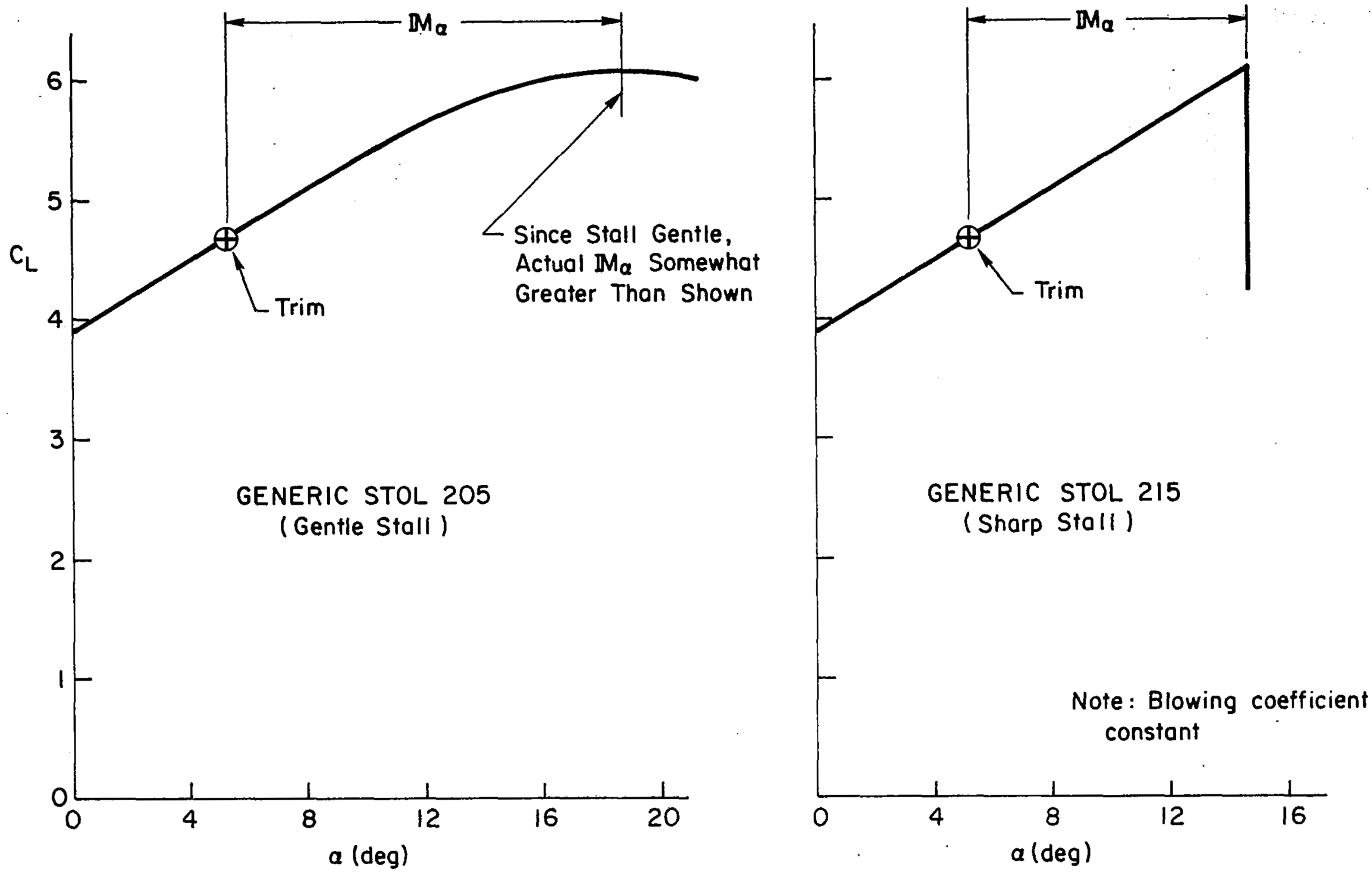

Figure 4-16: Lift versus Angle of Attack
(Cases used to show need for angle of attack margin)

inadequate, in terms of safety margins. The case which was inadequate had a sharp and unrecoverable limiting flight condition which occurred at an angle of attack 10 deg above the nominal flight condition. The other case had a relatively docile stall which occurred at 14 deg above the nominal flight conditions, and in being docile permitted angles of attack somewhat in excess of stall.

In view of other cases, all the safety margins in these two specific cases were adequate except angle of attack. While it is reasonably clear that lack of angle of attack margin was responsible for the inadequacy of case 215, it is not clear what particular aspect was deficient, i.e., vertical gust protection or ability to make attitude or angle of attack excursions in maneuvering. A combination of both was involved in this experiment because there was a fair level of atmospheric turbulence present, 1.4 m/sec (4.5 ft/sec) σ_{u_g}, and the pilots were intentionally abusing their pitch attitude and throttle controls.

All of the cases considered with regard to safety margins are shown in Figure 4-17, a plot of angle of attack margin versus pilot opinion. Note that a consistent trend exists and that the case which fell outside the speed margin trends, case 215, now falls in line with the angle of attack trends. This plot was used to infer the minimum angle of attack margin suggested above, i.e., 14 deg.

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FINDING:

A vertical gust margin of 15 kt appeared adequate for the powered-lift vehicles considered in this program.

DISCUSSION:

This level of vertical gust protection was inferred by considering the apparent vertical gust protection of all the cases from Table 4-1, in the absence of any maneuvering. A plot of the cases considered is shown in Figure 4-18.

Note:

- Shaded symbols indicate that margin taken with respect to "hard" limiting flight condition.
- All other cases for margin with respect to stall having varying degrees of gentleness, i.e., M_α somewhat greater than indicated.

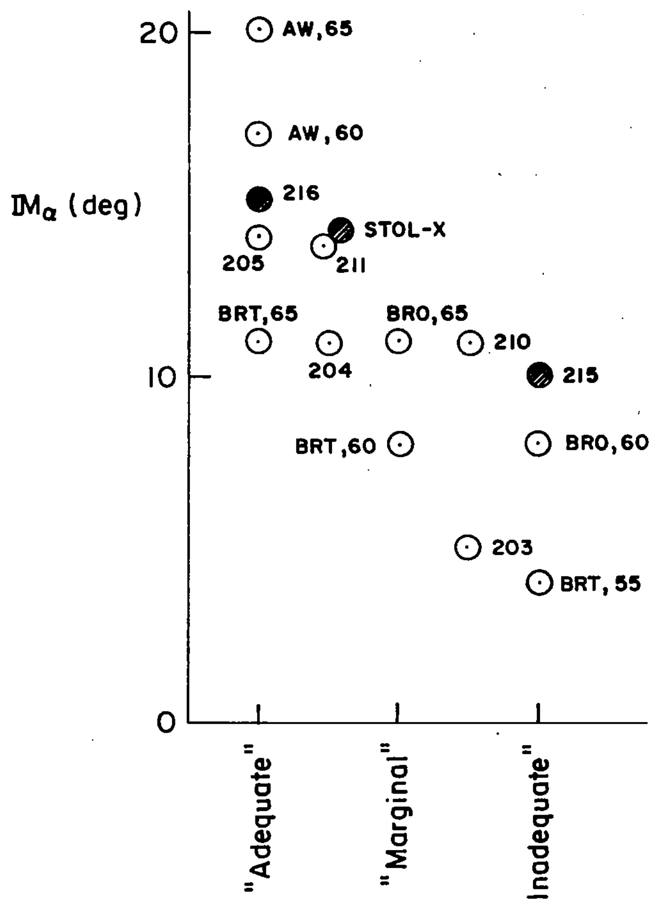

Figure 4-17: Angle of Attack Margin Trends

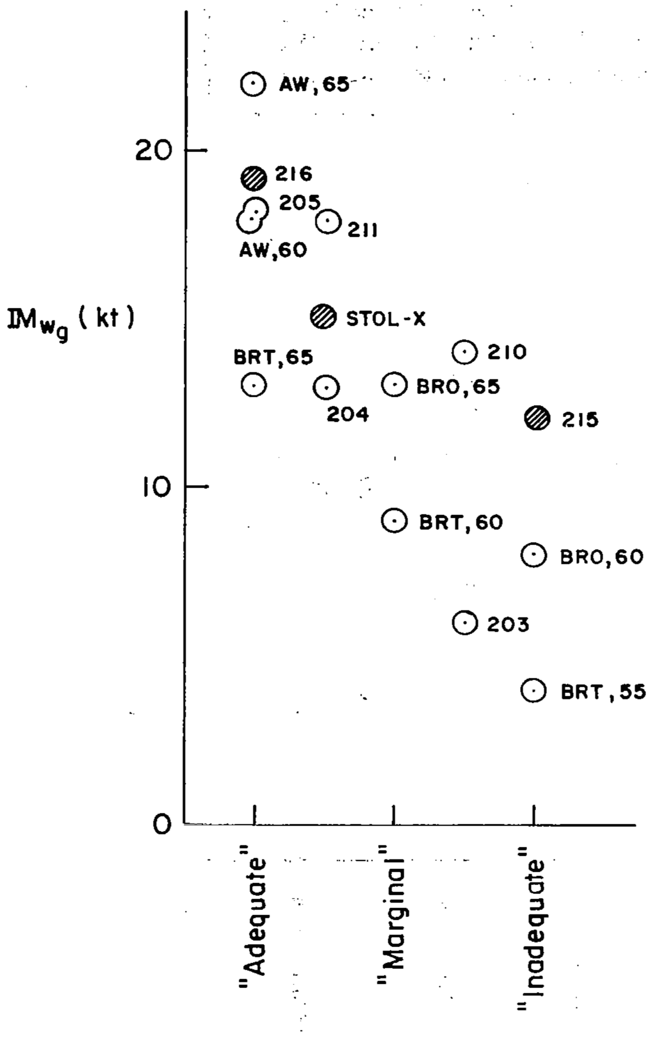

Figure 4-18: Vertical Gust Margin Trends

It is important to note that there is an overlap between angle of attack margin and vertical gust margin as determined here. Angle of attack margin, per se, seems more related to the pilot's use of the pitch attitude control, while vertical gust margin is obviously related to gust disturbances. Both, of course, concern a direct margin from an angle of attack limiting flight condition.

It is not possible at this time to determine a rational apportionment of M_α and M_{w_g}. For all practical purposes, though, use of one or the other is sufficient. Of the two, the most logical choice seems to be M_{w_g}. For the values determined here, $M_{w_g} \geq 15$ kt would be slightly conservative for approach speeds of 80 kt or less. Further, if minimum M_{w_g} were set equal to 20 kt as suggested by the SSDWG then it would probably remove any need for separate consideration of M_α.

We should note that the vertical gust margin suggested by the SSDWG (see Reference 15) is 20 kt. This larger value is based on matching current conventional aircraft capabilities.

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FINDING:

A lift margin of 0.15 g (for pitch attitude control use) appeared adequate for the powered-lift vehicles considered in this program.

DISCUSSION:

The necessity of a lift margin requirement was the subject of some debate by the SSDWG. For conventional aircraft, M_n does have a strong connection with maneuverability since pitch attitude is likely the primary flight path control. For powered-lift aircraft, however, M_n seems less important because:

i) Throttle is the most likely primary control

and ii) In any event, a minimum requirement on short term flight path control power (to be discussed in Section 5) would take the place of a lift margin.

For these reasons the SSDWG decided not to specify a lift margin at this time.

There is, however, at least one clear role for a lift margin requirement, and that is to provide for lateral maneuvering without increasing power. Thus a lift margin could be related to a turn rate capability by the following expression:

$$\dot{\psi} = \frac{g}{V}\sqrt{M_n\,(M_n + 2)}$$

The lift margins from the simulator experiments of this program are shown in Figure 4-19. A lift margin limit of 0.15 g was inferred from this, although it is not clearly delineated. This is close to the one flight test case (NC-130B) for which a reasonably accurate determination of M_n could be made. According to the above expression, M_n of 0.15 g would allow a turn of 9 deg/sec or 3 times standard rate at 70 kt in the absence of any other pilot or atmospheric induced angle of attack excursions.

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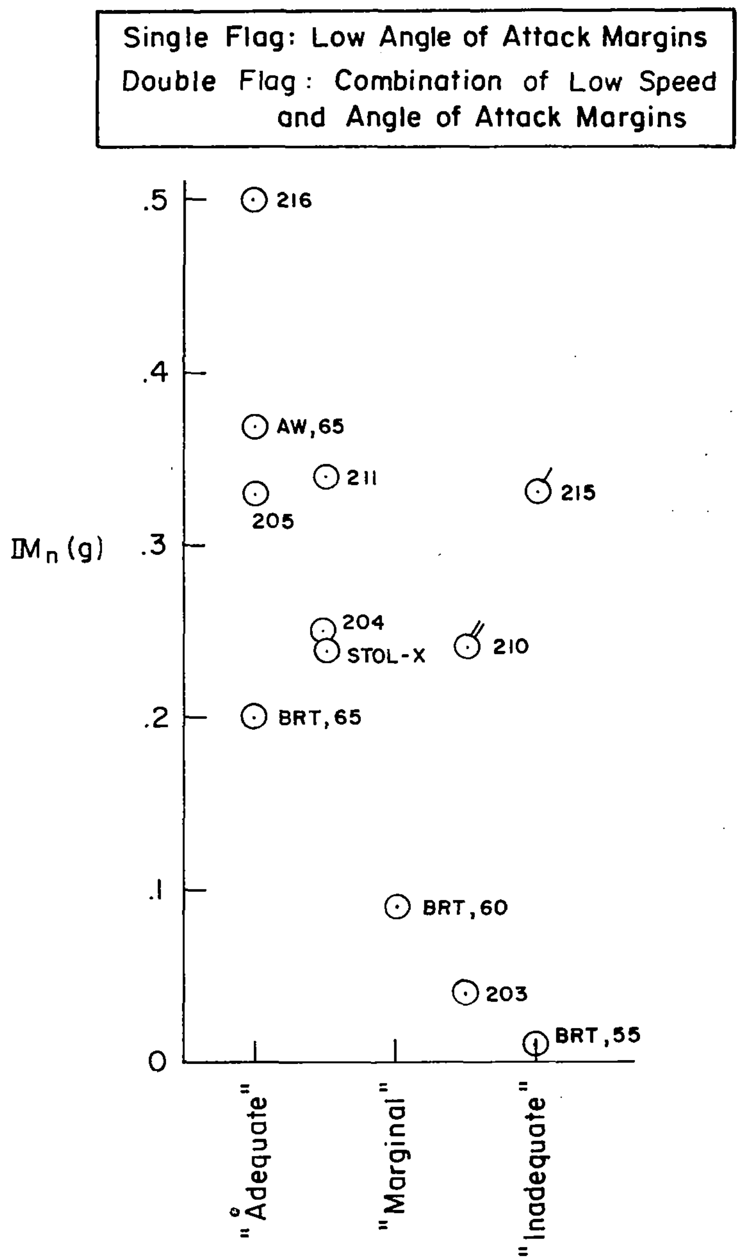

Figure 4-19: Lift Margin Trends

SECTION 5

LONGITUDINAL STABILITY, CONTROL, AND PERFORMANCE; APPROACH

This section deals with the broad spectrum of airworthiness considerations relating to longitudinal stability, control, and performance in the approach flight phase. The topics of stability and control, and of performance are treated together to preserve and emphasize their strong interrelation.

This section is divided into (1) piloting technique and (2) longitudinal control functions. The piloting technique subsection is relatively brief and discusses overall guidance and control objectives and various methods (piloting techniques) for achieving these. Only a few simulation results are given.

The subsection on longitudinal control functions constitutes the bulk of this section. It deals with the behavior of the airplane (i.e., stability, control, and performance) as it affects the various piloting tasks. This discussion is further subdivided according to the various longitudinal control tasks or feedback loops. In these subdivisions the majority of simulation results are presented and discussed.

Prior to discussing piloting technique, it may be helpful to review the general impact of powered lift on longitudinal stability, control, and performance. The two fundamental characteristics which tend to differentiate powered-lift aircraft from conventional ones are:

- Operation at low airspeed relative to wing-loading, i.e., high C_L
- A predominantly vertical force variation as a result of a throttle change.

Operation at a high C_L, the objective of powered lift, can be the result of (a) increased circulation by blowing the wing, (b) mechanical deflection of propeller slipstream or of jet exhaust using flaps or vectorable nozzles, or (c) a combination of both. The resulting high C_L has a direct effect on the responses of flight path and airspeed to attitude changes and gusts. These effects will be discussed in detail shortly.

A relatively large vertical force component due to a throttle change also comes about from the above mechanisms to get a high C_L. The effect is primarily on flight path and airspeed responses to a throttle change and can have a significant impact on piloting technique. This will be amplified in the control functions subsection.

The basic approach of the simulator study was to consider the airworthiness problems and possible airworthiness criteria relative to the overall pilot/vehicle system. This approach was reflected in the conduct of the simulator experiments. In early stages the program involved detailed simulations of actual aircraft, namely, the Breguet 941S (Reference 11) and the NASA Augmentor Wing Jet STOL Research Aircraft, AWJSRA, (Reference 12). This was done to observe the kinds of problems naturally occurring and their possible connection with a specific powered-lift concept. The program then progressed to direct variations of aircraft characteristics in the generic STOL simulations (Reference 13). Finally, after postulating possible criteria, a hypothetical aircraft design, which just met these criteria, STOL-X, was devised and simulated (Reference 14).

All of the approach simulations employed experienced pilots, a realistic approach task, realistic aircraft features, and a variety of realistic adversities. No heavily augmented configurations were considered. Pilots concentrated on one configuration at a time to provide adequate training in order to make fair comparisons.

5.1 PILOTING TECHNIQUE

The following is a discussion of the significance of piloting technique in evaluating longitudinal stability, control, and performance. It is a presentation of ideas and terms in preparation for the subsections on longitudinal control functions. The points covered are piloting objectives and various piloting techniques. This is followed by some of the more general findings relative to piloting technique.

The primary piloting task during the approach flight phase is that of vertical path control. The vertical flight path control during approach involves maintaining an acceptable proximity to a nominal glide path for the purpose of clearing potential obstacles and arriving at the point of flare such that a successful landing can be made. The term flight path, as used here, can refer to glide slope deviation, visually perceived flight path angle, altitude, altitude rate, etc., depending upon which is appropriate for a given situation.

Longitudinal tasks subordinate to vertical flight path control are pitch attitude and flight reference control. While pitch attitude is a specific unambiguous quantity, flight reference for a powered-lift airplane could be any one of several parameters, including the common ones such as indicated airspeed or indicated angle of attack. So far as the pilot is concerned, however, flight reference is the specific quantity indicated on the cockpit flight reference gauge.

The term "piloting technique" refers to the specific way in which the pilot uses the cockpit controllers to accomplish the tasks of vertical path control, attitude control, and flight reference control. The method of describing piloting technique which appears most useful is in terms of feedback loops. This is a meaningful concept for pilots and it allows use of powerful tools for engineering analysis. Perhaps most important, description in terms of feedback loops tends to simplify the mathematical approach to pilot/vehicle dynamics.

The basic loop structure involves three primary response variables:

- Pitch attitude, θ
- Flight path, FP
- Flight reference, FR

and two pilot inputs or cockpit controllers:

- Longitudinal control column
- Throttle (or some equivalent cockpit control lever).

Under some conditions, the pilot may make use of additional feedback variables (e.g., sink rate, engine RPM, or normal acceleration) but only to support and enhance the basic three-variable loop structure. Use of additional variables will be discussed later. The important point is that use of such additional variables or of pilot-generated compensation does not significantly affect the description of the piloting technique; likewise, for the case of additional cockpit controllers.

Before describing the ways of forming the pilot loop structure (i.e., defining piloting technique), consider the pitch attitude control task. Regardless of piloting technique, pitch attitude is regulated manually by the pilot or by an augmentation system. With attitude thus regulated, it can be considered as a control in itself in that the pilot can initiate and sustain changes in flight path and flight reference with a change in pitch attitude. Attitude regulation is an important simplifying function in that it allows the definition of piloting technique to involve only two controlled variables and two controls. That is, the controlled variables are flight path and flight reference; the controls are pitch attitude and a single cockpit control lever, normally the throttle.

There are two basic ways of forming the pilot loop structure or piloting technique: either pitch attitude can be used to control flight path, or throttle can be used to control flight path. If pitch attitude is used to control flight path, it is referred to as a conventional or CTOL piloting technique, also known as a frontside technique. Use of throttle to control flight path refers to the STOL technique which is also known as a backside piloting technique or slow flight technique. These two loop structures are shown in Figure 5-1. Note that in both piloting techniques a pitch attitude inner loop is indicated.

The above is, of course, an idealization of real pilot behavior. The most significant omission above is various control crossfeeds. A pilot

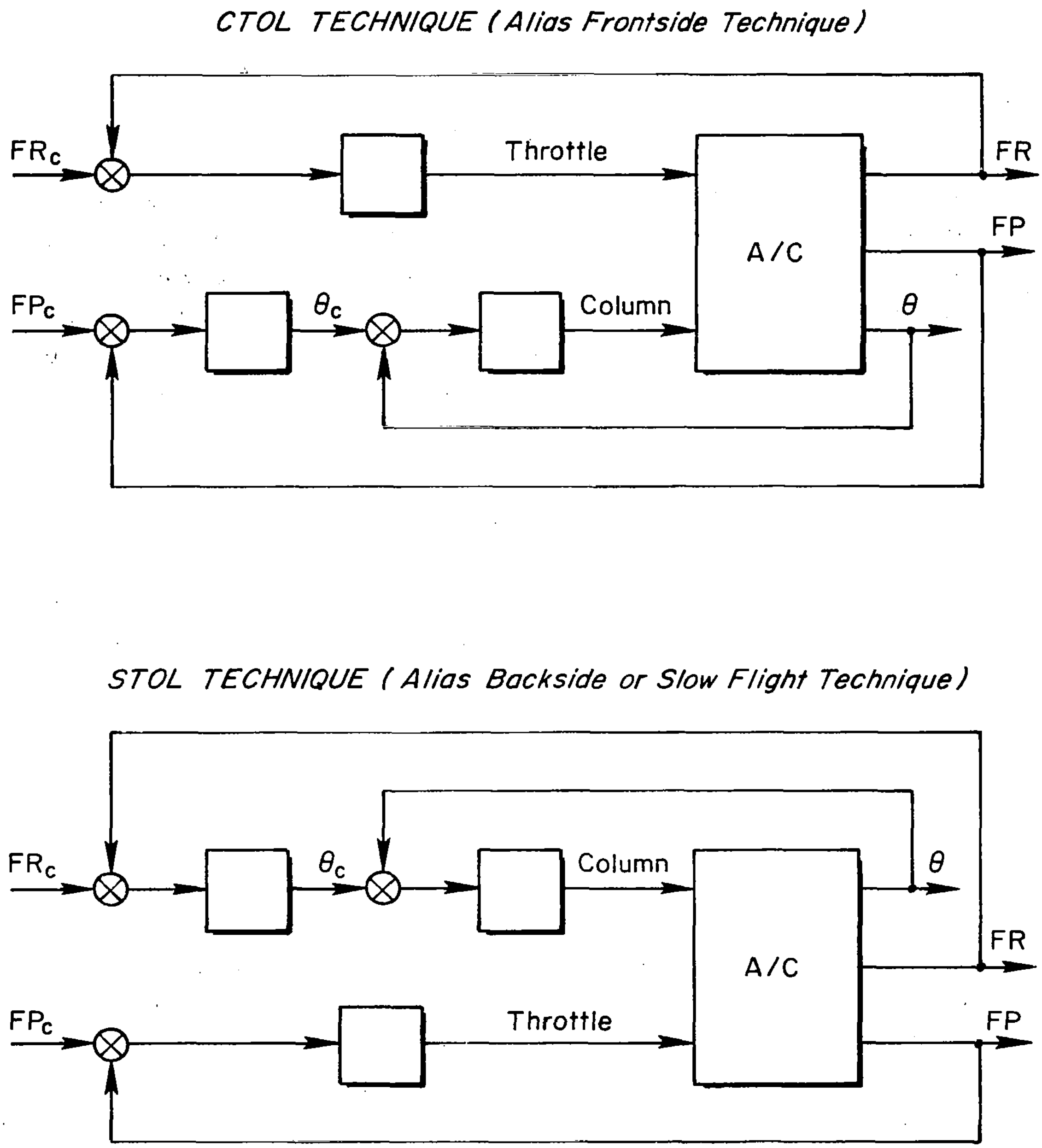

Figure 5-1: Pilot Loop Structure Forms

might learn that to go up he should add power and simultaneously pitch up. He would be using a control crossfeed or, in other words, coordinating his inputs. This might be done to enhance the flight path response or reduce the flight reference excursions. Nevertheless he should be able to identify either pitch or throttle as the primary control for regulating flight path. If he must use both controls for adequate performance, his opinion of the aircraft will probably be poor.

The reason for dwelling on the subject of piloting technique is that the behavior of an aircraft in terms of longitudinal stability, control, and performance is highly dependent on the pilot control loop structure. The choice of piloting technique can mean the difference between an acceptable airplane or an unacceptable airplane in the approach flight phase.

The foregoing ideas on piloting technique are supported by a number of simulation findings. Some of these findings are presented in the following paragraphs. Others will be presented in the subsections to follow.

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FINDING:

Subject pilots routinely were able to identify the piloting technique which they used as either STOL or CTOL.

DISCUSSION:

The loop structure appeared sufficiently obvious that the pilot could decide which one he was using. In the case of most pilots, however, piloting technique in the simulator was more obvious than for their flight experience in conventional aircraft. In the latter, some claimed a combination of controls was used, but if pressed to give the control used for rapid flight path regulation, the distinction could be resolved.

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FINDING:

The ease of adaption to the STOL technique depended to an extent on a pilot's background.

DISCUSSION:

The STOL technique was more readily used by helicopter pilots and Navy carrier pilots while the CTOL technique was preferred by pilots with conventional transport background. This phenomenon was observed and discussed in Reference 9, which involved a similar simulation experiment. While the pilot background effect was observed to some extent in this program, most of the subjects had no problem in using the STOL technique when desirable. The effect of pilot background was most noticeable when trying to flare using power rather than attitude. This is discussed in more detail in the flare and landing section, 6.

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FINDING:

The pilot tended to limit himself to only two longitudinal controls even though more were available.

DISCUSSION:

This was especially true in the latter stages of the approach. It was observed primarily in the AWJSRA experiments (Reference 12) where both nozzle and power control were available in addition to pitch attitude. The pilot was able to use both nozzle and power controls in the early stages of the approach where corrections could be made at a leisurely pace, but when near the ground, say below 60 m (200 ft), the nozzle control was set and maneuvering was done solely with throttle. Attitude was used for airspeed control. The subject pilots felt that the use of three controls to achieve acceptable performance presented an excessive workload.

It is likely that a three-control technique will be proposed for some airplane designs. It is suggested that in those cases the burden of proof of acceptability be left to the designer.

The remainder of results relating to piloting technique will be discussed in the context of longitudinal control functions.

5.2 LONGITUDINAL CONTROL FUNCTIONS

This section deals with the airworthiness considerations surrounding the three basic longitudinal control functions:

- Pitch attitude control
- Flight path control
- Flight reference control.

These three control functions are considered within the context of the pilot loop structure discussed in the preceding section. Approaching the airworthiness problem in this way aids in identifying the critical aspects of the longitudinal stability, control, and performance; and helps to sort out features important to powered-lift aircraft which may not be so with conventional aircraft.

Prior to considering the individual control loops, several important background concepts are presented. These include ways of representing the airplane behavior, elements of individual control functions, and dynamic features of the airplane.

In the following pages, several general means are used to represent the behavior of the airplane. The first of these is the input/output block diagram which was used in the piloting technique section and requires little explanation. Another way of describing the airplane is through the γ - V curve. This is useful in describing the steady state variations in flight path angle and airspeed (or flight reference) for a range of pitch attitude and power excursions. The main shortcoming of the γ - V curve is the lack of time response information. Airplane dynamics are thus described using linear equations of motion. Linear equations of motion require the use of dimensional stability and control derivatives and may be expressed, ultimately, as input/output transfer functions which relate any motion quantity to any given control input. An important set of linear equations of motion made use of in this study are the so-called "simplified" linear equations of motion, which result from constraining pitch attitude. A detailed account of these forms of describing airplane behavior is provided in Appendix A.

In order to describe the characteristics of control of any variable, certain elements are considered. The elements of a given control function include such features as control sensitivity, control power, response time, response shape, stability, and linearity.

Control sensitivity is the ratio of an aircraft response to the control input; for example, the ratio of pitch rate to control column deflection could be termed pitch rate sensitivity or column sensitivity. Sensitivity can also be considered in a short term or long term sense (i.e., high frequency or low frequency). Normal acceleration to thrust is an example of short term flight path control sensitivity while sink rate to thrust is a long term flight path control sensitivity.

Control power is the maximum amplitude of motion available with full control input; for example, rate of climb using maximum available thrust. Again, control power can be applied in both a short term and long term sense.

Response time refers to any of a variety of ways of measuring how fast a particular motion responds to a control. For example, for a unit control step input, the response time could be the time for a response to rise to 50% of its peak excursion. For a first order lag, the rise time to 63.2% (i.e., $1 - \frac{1}{e}$) is the lag time constant.

Response shape is a general term which relates the long term response to the short term response; for example, a response may rise to a peak then decay or completely wash out, or the response shape can be some form of oscillation. This is subject to a variety of definitions.

Stability refers to the tendency to quickly settle to a steady condition. If a system is unstable, the divergence may be aperiodic or oscillatory. Stability is usually considered for controls-fixed or controls-free but may also include the effects of pilot loop closures. A PIO tendency is one variety of closed loop instability.

Finally, linearity is the uniformity of response to a control input over a range of input magnitudes. Non-linearities can cause variations in control sensitivity, response time, or response shape with input magnitude.

Any significant amount of non-linearity is presumed undesirable and, as such, was not generally considered in these simulations.

In the following pages each of the three main longitudinal control functions will be discussed and respective simulation results given. The main point of discussion in each case will be the peculiarities associated with the use of powered lift. The most important of these discussions will be that dealing with vertical flight path control because it is the main piloting task and it differs most from conventional aircraft.

5.2.1 PITCH ATTITUDE CONTROL

The objective of this subsection is to provide a general discussion of pitch attitude control needs for powered-lift aircraft. This is followed by a brief presentation of simulation results which relate to the subject.

Minimal attention was given to the subject of pitch attitude control during this simulation program for two reasons. First, pitch attitude control has been the subject of extensive handling qualities research (notable references for powered-lift aircraft are 4, 8, and 30), thus the factors involved are relatively well understood, and in this program it seemed proper to concentrate on the less understood aspects of flight path and flight reference control. Second, use of powered lift tends to make pitch attitude control less important because in most cases a STOL piloting technique is more appropriate, at least in the approach flight phase. (During the flare, pitch attitude control may be more important, and this will be discussed in Section 6.)

There are two main functions for pitch attitude control. The first is phugoid damping, and the second is use of commanded attitude as either the primary flight path or flight reference control. Phugoid damping is required in both conventional and powered-lift aircraft, and usually requires a relatively loose regulation (i.e., lower crossover frequency) of pitch attitude. In conventional aircraft, pitch attitude is likely to be used as the primary flight path control, at least during the more critical stages of the approach flight phase. This normally requires a tighter control of pitch attitude than that needed simply for phugoid damping. In powered-lift aircraft where pitch attitude is usually used as a flight reference control, only a relatively loose loop is required, and a lower quality of pitch attitude control can be tolerated.

Phugoid damping is a prime benefit of holding pitch attitude and this applies to all flight conditions. The phenomenon of phugoid damping can be illustrated using conventional feedback control methods. Consider the example of a conventional airplane* as shown in Figure 5-2. Note that as a $\theta \longrightarrow \delta_e$ loop is closed the oscillatory phugoid roots tend towards the real θ numerator zeros, $\frac{1}{T_{\theta_1}}$ and $\frac{1}{T_{\theta_2}}$. The short period roots tend toward an oscillatory high frequency mode related to the tightness of the loop closure.

A relatively low gain closure will provide substantial phugoid damping but a higher gain is necessary for good flight path response. The closed loop response of flight path is given by:

$$\frac{\gamma}{\theta_c} = \frac{K_\theta N^{\gamma}_{\delta_e}}{\Delta + K_\theta N^{\theta}_{\delta_e}} = \frac{K_\theta N^{\gamma}_{\delta_e}}{\left(s + \frac{1}{T'_{\theta_1}}\right)\left(s + \frac{1}{T'_{\theta_2}}\right)\left(s^2 + 2\zeta'_{SP}\omega'_{SP}s + \omega'^2_{SP}\right)}$$

The normal effects of the various closed loop roots on the response are:

- The closed loop short period complex pair $(\zeta'_{SP}, \omega'_{SP})$ produces an initial response delay (like an actuator lag)
- The dominant response mode is $\frac{1}{T'_{\theta_2}}$
- A slow decay in γ results from the $\frac{1}{T'_{\theta_1}}$ mode.

Increasing the attitude loop gain slightly increases $\frac{1}{T'_{\theta_2}}$ (drives it closer to $\frac{1}{T_{\theta_2}}$) and thereby quickens the flight path response.

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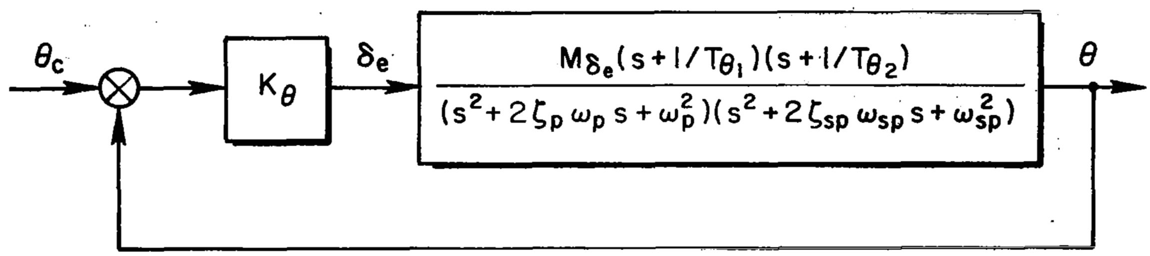
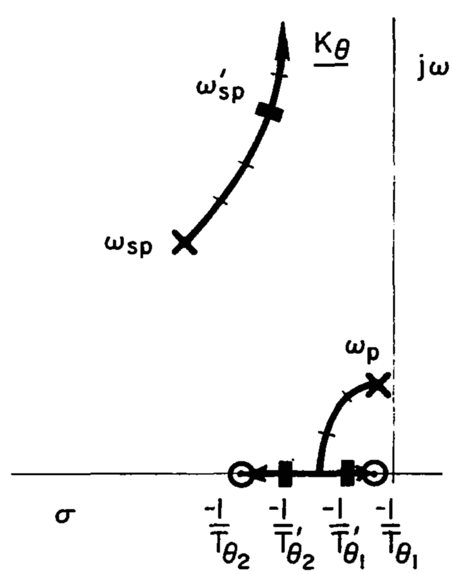
* Although the illustrative example corresponds to a conventional airplane the same principles apply to powered-lift airplanes.



a) Block Diagram



b) Root Locus

Figure 5-2: Pitch Attitude Loop Example

During most of the simulation program the manual task of pitch attitude regulation was eliminated through use of an attitude command SAS. Thus, phugoid damping was provided by the SAS and the pilot was left with the direct modulation of the column to control flight path or flight reference as the case may have been. The prime benefit of using such a stabilization system was to allow concentration on the areas of flight path and flight reference control. In some cases, though, attitude stabilization was not used and the pilot had the additional task of regulating pitch attitude. The simulation results relating to pitch attitude control from these cases are presented below.

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FINDING:

Manual pitch attitude control for a typical powered-lift configuration was generally considered to be a high workload but not the single limiting factor.

DISCUSSION:

A pitch SAS was not used in the first BR 941 simulation (Reference 11) nor for the baseline case of the STOL-X simulation (Reference 14). The 60 kt and 65 kt cases in the former simulation are regarded as being typical of powered-lift vehicles. The pilot comments relating to pitch attitude control workload were that the proportion of scan required in order to maintain pitch attitude was excessive because of little or no apparent stability, and noted that maintaining pitch attitude was a high workload task.

A relatively low short period frequency and high short period damping ratio characterized pitch attitude control in the vehicles examined. For example, the BR 941 operating between 60 and 65 kt had a short period frequency of about 0.9 rad/sec and a short period damping ratio of 0.9. The baseline STOL-X vehicle had a short period frequency of 0.7 rad/sec and a short period damping ratio of about 0.75. The high damping ratio indicates that the Z_w and M_q stability derivatives dominated the short period mode and that the static stability derivative, M_α, had a relatively weak effect. A low value of M_α was, in fact, considered to be what the pilot was complaining about when he referred to little or no apparent stability. The low value of M_α was due at least partially to the relatively low dynamic pressure or high C_L. It is for this reason one can expect powered-lift aircraft to generally have rather low short period frequencies and relatively high short period damping ratios.

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FINDING:

Substantial enhancement of attitude control was obtained through use of an attitude command system even though the system was characterized by a relatively slow response.

DISCUSSION:

The pitch attitude SAS used in both the second BR 941 simulation (Reference 11) and in the AWJSRA simulation (Reference 12), when characterized in terms of a first order lag, had an effective time constant of 0.75 sec. The augmentation system used in the Generic STOL simulations (Reference 13) had an effective time constant of nearly 2 sec. Normally these would be considered as slow attitude response times. In fact, the short period frequency (SAS on) for the Generic STOL was roughly 0.4 to 0.5 rad/sec which is below the Level 1 limit of MIL-F-8785B (Reference 31) and near the Level 2 boundary. There were no complaints, however, that pitch attitude response was, in fact, too slow in the approach flight phase, even for cases flown using a CTOL technique (Generic STOL case 1240). We can surmise, then, that the key effect of the addition of an attitude stabilization system was in damping the phugoid by holding the attitude and that any enhancement in short period response was not really essential.

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5.2.2 VERTICAL PATH CONTROL

This section covers the subject of vertical path control for powered-lift aircraft. It is an important section because it deals with some features previously addressed in a number of research efforts but not in airworthiness standards nor in handling qualities specifications. It represents one of the more extensive thrusts of this program. The section begins by a detailed development of basic pilot/vehicle dynamic relationships. This is followed by the presentation of simulation results broken down in terms of:

- Dynamic response
- Control power
- Cross coupling.

The preliminary discussion of vertical path control will center on a comparison between a typical conventional jet transport and what is presumed to be a typical powered-lift transport of comparable size. The development of this comparison starts with a few key parameters. These key parameters provide the basis for a subsequent formulation of stability derivatives and transfer functions, and finally, the computed motion response for both control and gust inputs. The main objective in this comparison is to provide a background for vertical path control in the approach but a broader use of this will be made. In particular, this material will also be applied to flare and landing.

The essential features of vertical path dynamics can be defined by six fundamental parameters. This is not a unique set of parameters, for they can be combined in many different combinations. Also, it should be noted at the beginning that we are describing the two-dimensional forces in the vertical plane, e.g., lift and drag, which are a direct result of either an applied thrust force, an instantaneous pitch attitude change, or a vertical or horizontal gust. Hence, propulsion system lags and control system lags must be considered separately, and it is convenient to do so.

We shall use the following set of fundamental parameters to define the bare airframe with respect to vertical path control. These parameters are:

- Approach speed, V_{app}
- Rate of descent, $\dot{h}_o$
- Effective thrust inclination, θ_T
- Powered-lift factor, η_p
- Normal acceleration with angle of attack, n_{z_α}
- Tangential acceleration with angle of attack, n_{x_α}.

The first two items in the above list simply represent the trim flight condition. The last four require some explanation. While the following paragraphs provide a brief explanation of each, a more complete discussion is given in Appendix A.

The effective thrust inclination, θ_T, is merely the angle formed by the resultant of the horizontal and vertical force components for an incremental power or thrust change. For conventional aircraft, the effective thrust angle is nearly, but not exactly, equal to the geometrical thrust angle with respect to the flight path. (It would be exact if the effects of ram drag were neglected.) For most powered-lift designs, the effective thrust angle has no direct geometrical relationship. It can have a complex relationship with such things as flap deflection, magnitude of blowing, the basic aircraft drag polar, vectored nozzle deflections, etc. It is, however, convenient to lump the combined effects of all of these into a simple effective thrust inclination.

The powered-lift factor, η_p, is some equivalent measure of the proportion of powered-lift to total lift. The definition used here is based on a key stability derivative in flight path control, Z_u. The derivative Z_u can be interpreted as the specific z-force change resulting from an airspeed change. For a conventional aircraft, Z_u is closely approximated by:

$$Z_u \doteq -\frac{2g}{V}$$

Powered lift tends to reduce Z_u and the powered-lift factor, η_p, is defined as that fractional decrease.

$$Z_u \doteq -\frac{2g}{V}(1 - \eta_p)$$

or

$$\eta_p \triangleq 1 + \frac{VZ_u}{2g}$$

The powered-lift factor can easily be related to basic aerodynamic properties if thrust effects are lumped into lift and drag. If C_J is the non-dimensional thrust coefficient, and if thrust does not vary with airspeed, then

$$\eta_p \doteq \frac{C_J}{C_L}\frac{\partial C_L}{\partial C_J}\bigg|_{\alpha=\text{const}}$$

or

$$\eta_p \doteq \frac{\partial \log C_L}{\partial \log C_J}$$

Thus, the powered-lift factor is the fractional change in the lift coefficient over the fractional change in the blowing coefficient. Hence η_p could be obtained directly from the slope of log C_L versus log C_J at constant α as shown in Figure 5-3. This relationship is derived in Appendix A.

Normal acceleration with respect to angle of attack is a commonly-used derivative and is normally expressed as n_{z_α}. The derivative n_{z_α} is approximately equal to C_{L_α}/C_L. If C_{L_α} is invariant, the derivative n_{z_α} is dependent upon C_L, and n_{z_α} is approximately proportional to the inverse of the speed squared.

The last parameter mentioned in the list above is tangential acceleration with respect to angle of attack. This is the counterpart of n_{z_α} aligned with the x-axis, or, as it is expressed in the following paragraphs, n_{x_α}. The derivative n_{x_α} is itself nearly invariant and depends primarily on wing aspect ratio. For aspect ratios of about 7, n_{x_α} is approximately 0.6. As aspect ratio decreases, n_{x_α} approaches unity as a limit.

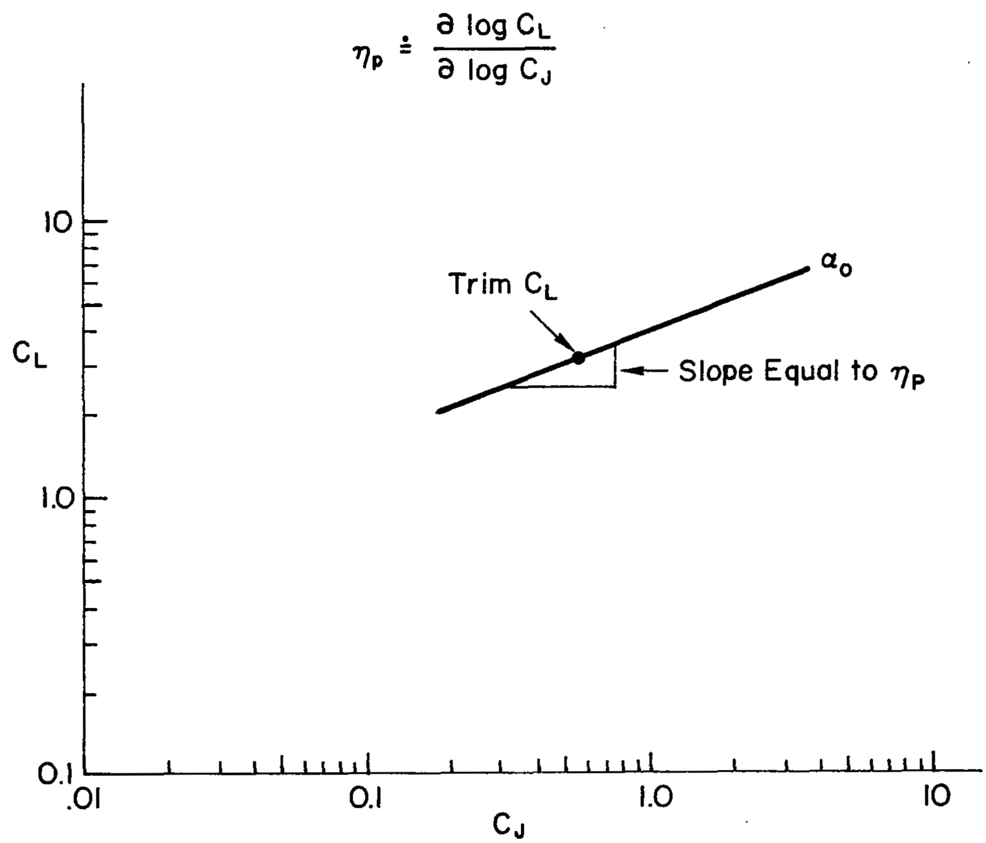

Figure 5-3: Appearance of η_p in Plot of Lift Coefficient Versus Thrust Coefficient

In the powered-lift/conventional aircraft comparison mentioned previously, we defined the two cases in terms of the above six parameters. These are shown in Table 5-1. The numerical values are simplified but are, nevertheless, typical of their respective designs. (Appendix A discusses typical values of these parameters.) A broad range of conventional jet transports are approximated by the numerical values given. They would correspond to a flight condition near 1.3 V_{min}. The list of parameters describing the powered-lift aircraft is also highly representative except that the flight path dynamics are probably more sensitive to the same percentage variation of these parameters. Note that the key differences between these two examples are:

- The approach speed (130 kt versus 75 kt)
- Effective thrust inclination (horizontal versus vertical)
- Effective powered-lift factor (zero versus 40%)
- Normal acceleration with respect to angle of attack (4 g/rad versus 2 g/rad).

If we assume an equal wing loading, then the approach speed difference indicates a difference in lift coefficient of a factor of three. This speed difference combined with the difference in n_{z_α} of a factor of two therefore implies that C_{L_α} has increased by 50% in the powered-lift vehicle, which is reasonable. (Appendix A discusses how C_{L_α} can increase with jet flap effect.)

The parameters can be transformed into a set of dimensional stability derivatives and transfer functions to describe vertical path dynamics. These are summarized in Table 5-2. It is important to note the similarities in the two sets of stability derivatives. The derivatives X_u, Z_u, and Z_w all are approximately the same magnitude even though the parameters composing these derivatives are all different. The major difference to be noted in these derivatives are in the thrust control derivatives X_{δ_T} and Z_{δ_T} which are strictly functions of effective thrust inclination. A modest increase in the derivative X_w is primarily responsible for a degree of coupling in the normally distinct flight path and airspeed response modes.

TABLE 5-1

FLIGHT DYNAMICS COMPARISON
(Basic Assumptions)

	CONVENTIONAL	POWERED-LIFT
Approach Speed	130 kt	75 kt
Rate of Descent	3 m/sec (600 ft/min)	3 m/sec (600 ft/min)
Effective Thrust Inclination	Horizontal	Vertical
Effective Powered Lift	zero	40%
Normal Acceleration with α	4 g/rad	2 g/rad
Axial Deceleration with α	0.6 g/rad	0.6 g/rad

TABLE 5-2

COMPARISON OF STABILITY DERIVATIVES AND TRANSFER FUNCTIONS
(Simplified Path Equations of Motion)

		CONVENTIONAL	POWERED-LIFT
Stability Derivatives:			
	V_{app} (kt)	130	75
$\frac{2g}{V}\left(\tan\gamma_o - \frac{\eta_p}{\tan\theta_T}\right)$	$= X_u\,(\frac{rad}{sec})$	$-.04^*$	$-.04$
$\frac{g}{V}(1 - n_{x_\alpha})$	$= X_w\,(\frac{rad}{sec})$	+.06	0.10
$-\frac{2g}{V}(1 - \eta_p)$	$= Z_u\,(\frac{rad}{sec})$	-.29	-.31
$-\frac{g}{V}n_{z_\alpha}$	$= Z_w\,(\frac{rad}{sec})$	-.59	-.51
$\frac{g}{100}\cos\theta_T$	$= X_{\delta_T}\left(\frac{ft/sec^2}{\%}\right)$	0.32	0
$-\frac{g}{100}\sin\theta_T$	$= Z_{\delta_T}\left(\frac{ft/sec^2}{\%}\right)$	0	-.32
Transfer Functions:			
	Δ	$(.07)(.56)^{**}$	(.12)(.43)
	$N_\theta^{u_a}\,(\frac{kt}{deg})$	-.20(.98)	-.20(.85)
	$N_\theta^{\gamma}\,(\frac{deg}{deg})$	0.59(-.004)	0.51(-.053)
	$N_{\delta_T}^{u_a}\,(\frac{kt}{\%})$	0.19(.59)	-.019
	$N_{\delta_T}^{\gamma}\,(\frac{deg}{\%})$	0.024	0.146(.04)
	$N_{\dot{u}_g}^{u_a}\,(\frac{kt}{kt/sec})$	(.59)	(.51)
	$N_{\dot{u}_g}^{\gamma}\,(\frac{deg}{kt/sec})$	0.13	0.24

* C_D/C_L assumed 7.5 and $X_u = -\frac{2g}{V}\frac{C_D}{C_L}$.

** The following shorthand notation is used for transfer functions:

$$\left(\frac{1}{T}\right) \triangleq s + \frac{1}{T} \quad \text{and} \quad [\zeta,\omega] \triangleq s^2 + 2\zeta\omega s + \omega^2$$

The motion resulting from three different kinds of inputs to the two kinds of aircraft are shown in Figure 5-4. The motions plotted are γ and V and the inputs are pitch attitude, incremental thrust-to-weight, and horizontal wind shear. Each of these inputs are in the form of a unit step. The main features of these time histories are summarized in Table 5-3. Note that the largest difference is in the γ and V motion resulting from a thrust input, i.e., the effect of a horizontal thrust inclination versus a vertical one. This, of course, was plainly visible in the control derivatives as mentioned previously. Also, it is the most influential feature so far as choice of piloting technique is concerned.

The features viewed in the above comparison help to formulate the following scheme for describing the important elements of vertical path control. These elements are lumped into the following groups:

- Dynamic response
- Control power
- Cross coupling.

Dynamic response refers to such features as control sensitivity, response time, response shape, stability, and linearity. It involves a response to disturbances such as gusts as well as the response to control inputs. Dynamic response directly determines flight path tracking precision of the closed loop pilot/vehicle system. Referring back to Figure 5-4, the features of dynamic response of particular importance to powered-lift aircraft are the time for γ to respond to $\Delta T/W$ and how well the initial response is sustained. The case illustrated rises quickly but possesses some decay of flight path which could be unsatisfactory if too extreme.

Vertical path control power describes the maximum path excursion possible, up and down, in both the short and long term. Control power capability ultimately determines the maximum size and duration of disturbance that can be tolerated. Note that some indication of relationships from linear dynamic response and limiting control excursions is evident.

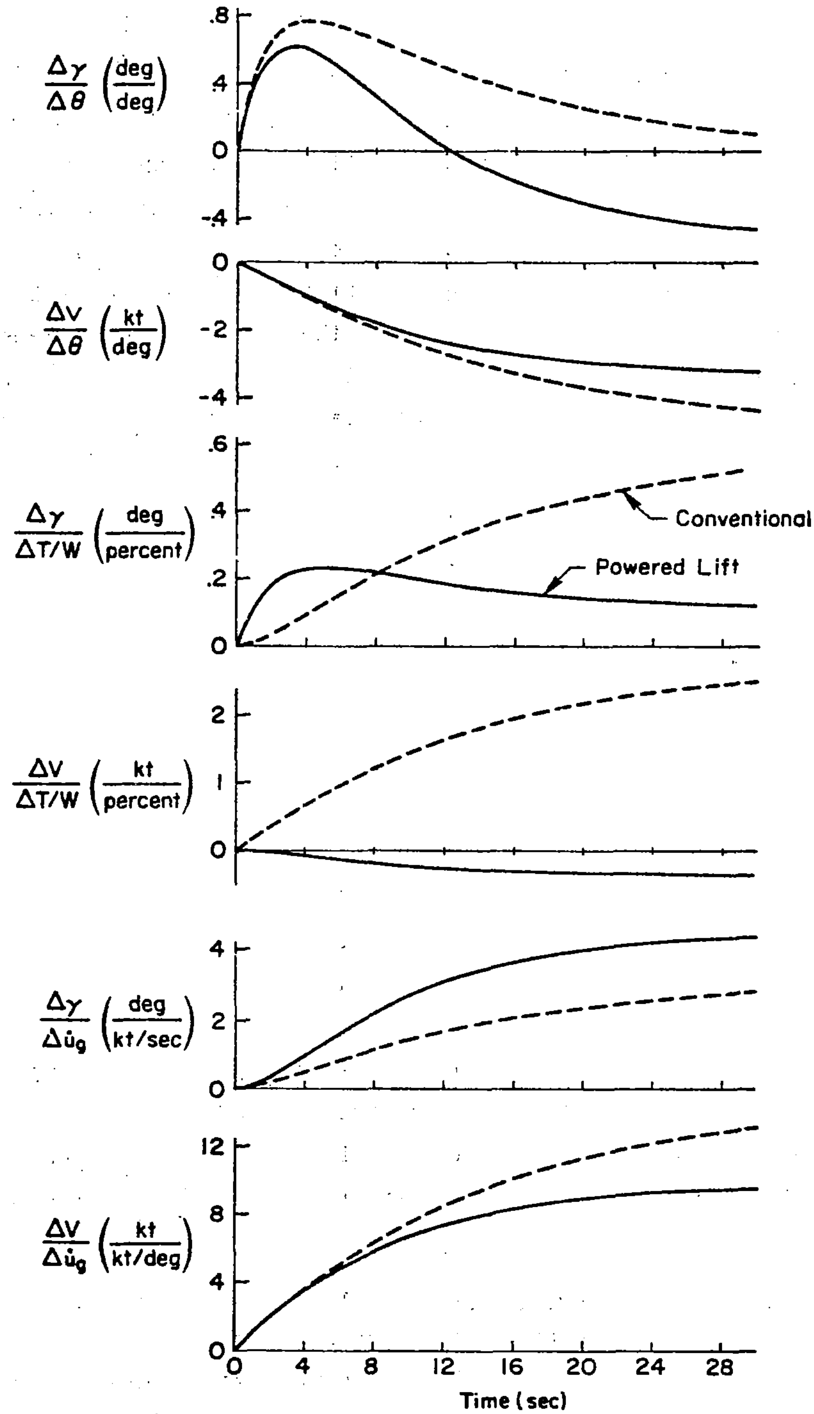

Figure 5-4: Comparison of Time Responses Between a Conventional and Powered-Lift Airplane

TABLE 5-3

DISTINGUISHING FEATURES OF γ AND V RESPONSES FOR POWERED-LIFT AIRCRAFT COMPARED TO CONVENTIONAL.

	CONVENTIONAL	POWERED-LIFT
$\frac{\Delta\gamma}{\Delta\theta}$	Fast rise, then slow washout	About the same rise time but larger and faster washout with long term loss (backside)
$\frac{\Delta V}{\Delta\theta}$	Slow but consistent loss in V	Initial loss comparable but not as sustained (higher effective speed damping as characterized by $1/T_{\theta_1}$)
$\frac{\Delta\gamma}{\Delta T/W}$	Very slow but eventually large magnitude (long term creeping response)	Rapid response which quickly peaks and partially decays
$\frac{\Delta V}{\Delta T/W}$	Sluggish response with sustained long term increase, corresponds to long term flight path increase	No initial response and only small amplitude long term decrease
$\frac{\Delta\gamma}{\dot{u}_g}$	Similar to $\Delta T/W$ response	Same shape as conventional aircraft but larger magnitude, however if $\dot{d}$ response compared then powered-lift and conventional about the same.
$\frac{\Delta V}{\dot{u}_g}$	Unit slope initial response with sustained long term increase	Same initial response as conventional but reaches limit sooner.

Cross coupling, in one sense, refers to the effect that the primary control has on variables other than flight path. The most important of these coupling effects are the effect of primary control on pitch attitude and primary control on flight reference. But, cross coupling can also refer to the effect the secondary control has on flight path. The unfortunate result is that pilot management of cross coupling effects can cause a significant increase in overall pilot workload.

5.2.2.1 DYNAMIC RESPONSE, VERTICAL PATH CONTROL

Vertical path dynamic response is an area in which considerable progress was made in understanding and in developing specific airworthiness criteria. The findings are presented in rough chronological order following a brief account of the nature of the simulation experiments used.

The first BR 941 simulation (Reference 11) served as a starting point. For the most part we simply observed the effects of dynamic response characteristics naturally occurring in the model. No direct variations were made, only indirect changes as the result of varying airspeed and transparency configuration (differential inboard/outboard propeller pitch). During the second BR 941 simulation (also Reference 11), dynamic response was focused on with the help of the pitch attitude command SAS. This relieved the pilot of manual attitude regulation. But, as in the previous experiment, there was no direct variation of dynamic response characteristics.

During the AWJSRA simulation (Reference 12), systematic variation of some dynamic response features was begun. In addition to variation of airspeed, the effective engine lag was varied. During the post-simulation analysis, the mathematical description of dynamic response was formalized and limits proposed.

These proposed limits for short term response were explored during the Generic STOL simulation (Reference 13). Aside from looking at a wide variety of dynamic response cases, there was some systematic variation of response time. As a result of this simulation, the proposed dynamic response criteria were revised and tentative numerical limits set. These were introduced in the first STOL Standards Development Working Group (SSDWG).

The STOL-X model (Reference 14) was developed based on the working group short term response limits. However, during the simulation, some variation was made in dynamic response in order to refine numerical limits. Following the STOL-X simulation, dynamic response criteria were adjusted using all available sources up to that point. These were introduced at the second SSDWG meeting.

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FINDING:

The most important feature of dynamic response regarding powered-lift aircraft was the quickness with which the change in flight path or sink rate follows a change in primary control.

DISCUSSION:

This general finding suggests the need for a direct criterion governing short term response of flight path. This feature of dynamic response has long been recognized as important in the closed-loop flight path control problem. Most forms of conventional aircraft, however, have possessed adequate response and no direct criteria have appeared in military specifications or civil airworthiness standards. Nevertheless, others have conducted research to determine important parameters and numerical limits. Such research was used to formulate our approach at the outset of this program.

During the first BR 941 simulation, short term response was viewed in the powered-lift regime. The variations in approach speed that were run covered a wide range of response, albeit in the presence of other variations such as in safety margins and cross coupling. The subject pilots did comment on problems associated with sluggish path response. Even at the comparatively good condition of 65 kt with transparency the pilots noted that following a power change they had to allow considerable time for the effect to become apparent before making further changes. During the second

BR 941 simulation, a pitch attitude hold SAS was used in order to effectively remove the pitch attitude loop considerations. Thus, longitudinally, only flight path and flight reference tasks remained. Still, flight path control was viewed as very difficult and unresponsive. One way the pilots gauged this lack of responsiveness was by directly observing the flight path response on IVSI.

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FINDING:

Relief from the pitch attitude control task did not necessarily result in improvement in the flight path/flight reference task.

DISCUSSION:

Historically, pitch attitude control has been considered an important factor in ensuring good flight path control. Hence, considerable weight has been given to observing boundaries of minimum short period frequency and damping. Undoubtedly this is important for conventional aircraft since pitch attitude control is frequently used as the primary path control and easy, fast attitude control is significant in the overall control/airframe response. Also, as a rule, conventional jet transport aircraft possess reasonably good heave damping, thus the vertical path response of the basic airframe is fast.

For a powered-lift airplane where a STOL piloting technique is used, the quality of pitch attitude control is relatively unrelated to the flight path control task. If pitch is used for flight reference control then only a low frequency response is required. Improvement of θ frequency response beyond a certain point past the flight reference crossover frequency is unnecessary. During the second BR 941 simulation an attitude command/attitude hold SAS was used in order to remove attitude loop considerations. Thus, in the longitudinal axis, only the flight path and flight reference tasks remained but glide slope control was still viewed as very difficult and unresponsive. The attitude SAS did aid in relieving pilot workload as indicated in a previous finding but it had no significant effect in improving flight path control.

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FINDING:

Short term control problems became more critical with decreasing altitude such that the most critical part of the approach was just prior to the flare.

DISCUSSION:

This finding was based on pilot observation and explained by an analytical argument. It was verified and discussed in detail in Reference 10. The implication relative to airworthiness criteria is that the most critical part of the approach should be considered, and this critical segment most likely involves an outside visual flight path reference rather than a cockpit ILS reference.

Early in the program, pilots were asked to comment and rate vertical path control in terms of "ILS glide slope tracking." Also, performance was measured during ILS tracking, i.e., from 300 m (1000 ft) down to 90 m (300 ft). During later experiments it became clear that glide slope tracking did not represent a particularly critical task. The latter stage visual tracking portion of the task, in fact, presented more of a problem. Therefore, in subsequent experiments, pilots were requested to comment on the entire approach down to flare initiation.

The phenomenon of increasingly critical flight path control is explained by the fact that the pilot is, in general, closing a loop around an angular flight path relation. During the IFR portion of the approach, this angular relation is that displayed by the glide slope needle, $\frac{d}{R}$. During the visual portion, this angular relationship may be that between the instantaneous flight path and the runway aim point, $\frac{d}{R} + \Delta\gamma$. These are illustrated in Figure 5-5. In either case, if the pilot tries to maintain a constant angular excursion, he is thereby forced into trying to control within a smaller linear error as range decreases. Therefore, this

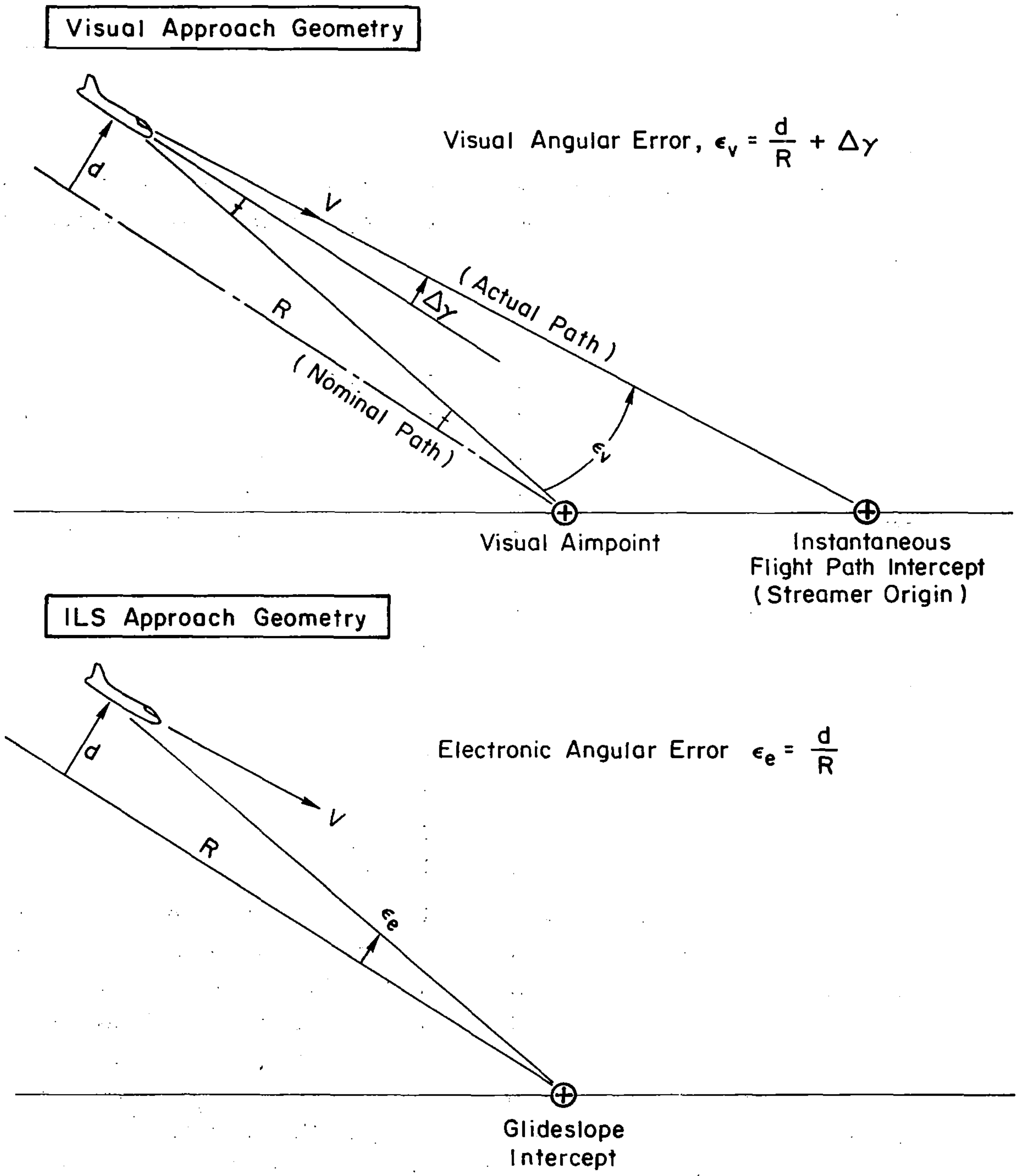

Figure 5-5: Two Examples of Perceived Flight Path Error (Visual and ILS)

demands the highest precision in linear distance from his nominal glide path at the lowest altitudes. The result of this flight path loop closure is that closed-loop damping in the pilot/vehicle system decreases continually as flare is approached. If the damping is too low, the pilot must counter it by generating lead compensation or making use of rate information such as the IVSI. In either case pilot workload is increased.

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FINDING:

The most important disturbance effect on vertical path control is horizontal gust activity, including wind shears.

DISCUSSION:

The implication of this finding is that the airworthiness criteria must address the problem of horizontal gusts and shears as directly as possible. Also, it is important to define the maximum gust environment expected in operation.

The atmospheric disturbance environment used in the various simulations of this program is described in Appendix B. One important feature relative to vertical path control is level of intensity, normally expressed as the RMS horizontal gust, σ_{u_g}. Another important feature is the low frequency spectral content of horizontal gust given by the scale length. The horizontal gust is converted to airplane heave motion primarily through the dimensional stability derivative Z_u. It was noted previously that although Z_u increases with decreasing approach speed, it is reduced by the powered-lift factor, η_p. As a consequence, the value of Z_u for the powered-lift aircraft is comparable to conventional ones.

Vertical gusts such as those simulated do not have a significant effect on vertical path control. At higher altitudes, say 300 m (1000 ft), the intensity of vertical gusts is about equal to horizontal gusts but since the vertical gusts have a higher frequency content, much of the effect is filtered out by the low frequency response of the airplane. Intensity of the vertical gust decreases with altitude and the choppiness further increases. Thus the effect on the airplane diminishes even more. Horizontal gusts, on the other hand, remain at about the same intensity while their frequency content stays well within the response bandwidth of the airplane all the way to the ground. This is described more fully also in Appendix B.

Horizontal gusts and especially sustained shears were always a dominant factor in pilot ratings. In all simulations, runs were made first in calm air then in some substantial level of turbulence, commonly $\sigma_{u_g} = 1.4$ m/s (4.5 ft/s). This consistently increased pilot workload and degraded performance. Specific examples of this will be shown in subsequent findings.

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FINDING:

The factors which were theoretically shown to determine short term response tended to be verified in the piloted simulation experiments. These included:

- Airframe heave damping
- Primary control lag (usually engine)
- Inclination of net force resulting from primary control action
- Airframe cross coupling.

DISCUSSION:

The following is a series of examples of how key short term response parameters were viewed during particular experiments.

In the various powered-lift configurations considered in this program, probably the most frequent determining factor for short term response was the effective inclination of net force due to action of the primary control. To re-cap the effect of this briefly consider the cases in Figure 5-6. Dynamic characteristics aside from θ_T are identical to the previous powered-lift example. If the inclination is slightly forward of vertical, say approximately 70 to 80 deg, then the short term response is almost exclusively

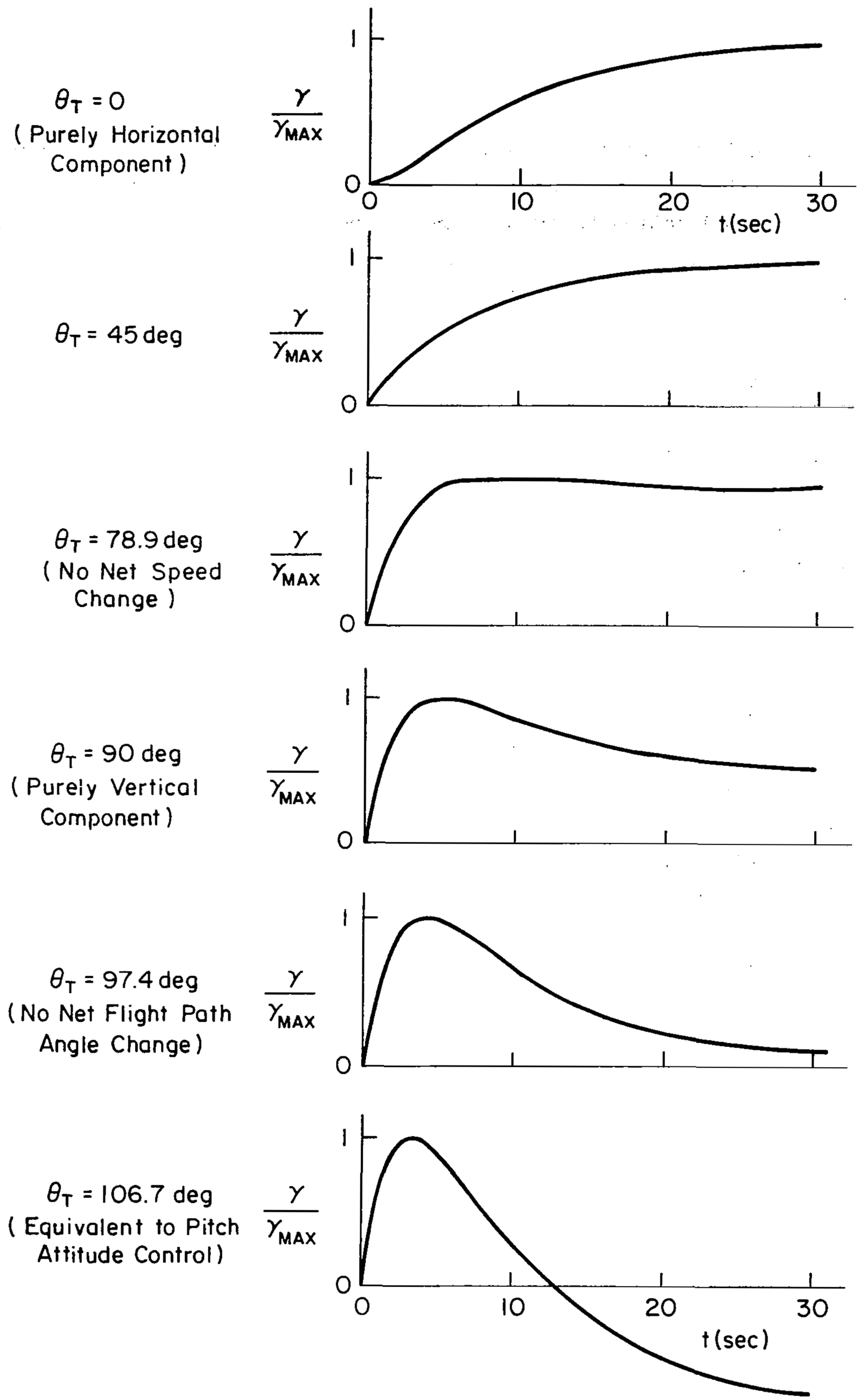

Figure 5-6: Shape of γ Response to Step δ_T for Varying θ_T

determined by airframe heave damping and primary control lag. Also, the shape of the response will be close to that of a first order lag. If the control inclination is tipped forward toward horizontal there is an increase in response time. Conversely, if the effective inclination is tipped aft, the initial response is more rapid, but there is a decay from the peak value.

A difference in thrust inclination was one of the most distinguishing features between the AWJSRA and BR 941 airplane simulations. The AWJSRA simulation, over a speed range of 60 to 65 kt, had an effective thrust angle of 90 deg. For the BR 941 simulation, over the same speed range, the effective thrust angle was about 80 deg. While the two aircraft had nearly the same effective lag due to heave damping and engine lag, the overall rise time (to 50% of the peak) of the AWJSRA was only 1.7 sec compared to around 3 sec for the BR 941. This was reflected in a difference in pilot opinion of about 1 unit when flying in turbulence. A similar situation was observed in the Generic STOL cases 1210 and 1250 (Reference 13). There was also a corresponding difference in pilot opinion. In these simulations it was observed that aircraft having a thrust angle of around 90 deg were more difficult to degrade through increased primary control lags. Also, in cases with the vertical thrust inclination there was significantly less mention of sluggish flight path response. Thus, this is an example of how at least one form of cross coupling can have a favorable effect.

Another observation was that the dominant feature of short term response is the net airframe plus control lag. The way in which airframe lag can combine with engine lag is shown in Figure 5-7. It is not possible to set an absolute limit on control lag alone and thereby guarantee short term response. This was most directly shown in the STOL-X simulation in which, for two configurations, the overall rise time was kept constant but the portion of control lag versus airframe lag was varied. Thus for an increase in engine lag, the airframe lag was reduced through use of an automatic direct lift control. To the pilot the engine lag was reflected by the engine noise and engine RPM indication. Two pilots examined these cases. One pilot could make no distinction at all. The other pilot could

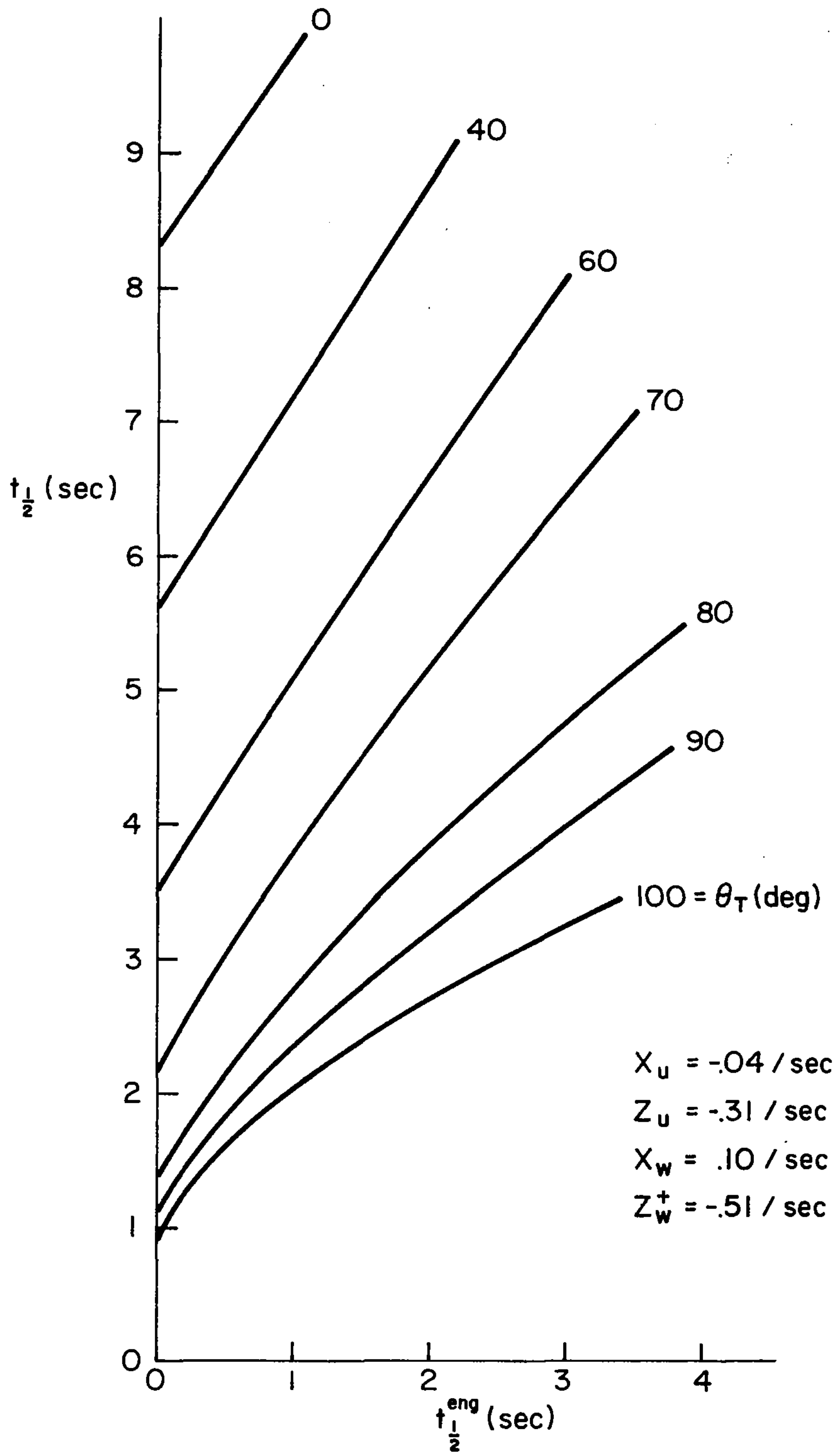

Figure 5-7: Overall Flight Path Rise Time Versus Thrust Rise Time and Thrust Inclination

make a distinction but it was not great. With an additional variation in engine lag of 0.3 sec out of a total of 2, the distinction all but disappeared.

An example of airframe cross coupling acting or influencing short term response was observed in the baseline STOL-X configuration. This case was flown without an attitude stabilization system. It had a favorable pitching moment due to power such that it would tend to pitch to hold the desired flight reference without the pilot initially commanding a new attitude. This resulted in a net reduction in rise time compared to a rigidly constrained pitch attitude situation.

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FINDING:

Pitch attitude regulation requirements are important when considering short term path response.

DISCUSSION:

This idea is not new -- it comes from previous analytical and experimental efforts, but it bears reinforcement in our consideration of airworthiness standards. That is to say, any path response standards should be stated with an appropriate treatment of pitch attitude regulation.

The main idea is that short term path response has no real connection to bare airframe response modes, in particular the phugoid. Instead, path response is a direct result of either varying pitch attitude and leaving other controls fixed, or varying another control and holding pitch attitude fixed. The resulting dominant modes are those which are important to either manual or automatic control of flight path.

If pitch attitude is the primary control, then there is no real problem in applying the assumption of perfectly constrained pitch attitude. In order for the pilot to command pitch attitude it is necessary for him to regulate it. If pitch attitude is a secondary control, then there is some complication depending upon pitching moment due to primary control and automatic pitch stabilization. The nature of pitch attitude regulation can vary with individual pilot technique, and it can be dependent upon altitude.

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FINDING:

No active manual regulation of flight reference should be assumed when considering short term path control.

DISCUSSION:

How the pilot regulates flight reference, just as pitch attitude, affects the short term response. Therefore, the proper requirement must be applied to flight reference control as well as pitch attitude control in determining an appropriate short term response criterion. The most reasonable assumption is that there is no manual regulation of flight reference. The general argument for this is that flight reference control is carried out in a looser or lower frequency loop than is flight path control, and if flight reference is controlled very tightly then it involves an excessive workload. In fact, loose flight reference control relative to flight path control is increasingly truer with decreasing altitude where flight path control becomes most critical.

Some insight into this idea came as a result of the second Generic STOL simulation. The series of cross coupling cases that were examined included ones for which, with the flight reference perfectly regulated, the short term path response would be just barely adequate. It was found that the pilot would not tightly regulate his airspeed flight reference even in cases where the airspeed response to the primary control (throttle) was clearly excessive. In fact, in those cases, the pilot tended to drop all flight reference regulation during the critical low altitude segment of the approach. Under this condition, the attitude-fixed short term path response was so sluggish that a flight path PIO was encountered.

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FINDING:

Rate of descent information is an important aid to the pilot in the vertical path control task.

DISCUSSION:

While sink rate regulation is not an outer loop, it can be and is used as an important supporting loop. A display of rate of descent gives the pilot lead information with respect to glide slope excursions. Use of such lead information helps to reduce the oscillatory tendency in the flight path loop. Most pilots commented that they did, in fact, continually scan the IVSI. Reference 10 reports describing function measurements for the glide slope tracking task. These show an effective pilot delay of several seconds and suggest a rate of descent feedback loop is required; otherwise the large pilot-generated compensation would represent an excessive pilot workload. The implication of this is that rate of descent information may be a requirement for the approach flight phase with a powered-lift vehicle. See Reference 21 for more evidence on this point.

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FINDING:

A rise time criterion is an effective means of specifying adequate short term response.

DISCUSSION:

This is a key result of this program relating to short term dynamic response. Conceptually, a rise time criterion is attractive because it involves a direct measurement of response following a given control input. Some possible risks are over-simplification, application to inappropriate input-output quantities, and sensitivity to measurement error.

The input-output quantities considered most useful for a rise time measurement are flight path angle due to a step primary control input with fixed secondary control. One can claim that if γ closely follows the primary control this gives the pilot a so-called velocity control of glide slope error, an ideal form of controlled element. The rise time feature would thus provide a direct connection to the potential glide slope control bandwidth (the frequency out to which the airplane behaves as an ideal velocity control).

The rise time definition tentatively chosen is the time from initiation of input step to $\frac{1}{2}$ the first flight path peak, $t_{1/2}$. This definition is somewhat arbitrary -- rise time could be the time to 63%, from 10% to 90%, etc. The scatter in the available data was large enough to obscure any relative evaluation of the various rise time definitions. $t_{1/2}$ was chosen mainly for its simplicity.

The upper limit on rise time has been determined from a relatively large number of cases. These cases are tabulated in Table 5-4. Relatively consistent results were obtained for constant levels of turbulence. Figure 5-8 shows a plot of pilot rating versus $t_{1/2}$. Based on the 1.4 m/s (4.5 ft/s) σ_{u_g} used in this simulation program, a $t_{1/2}$ less than 3 sec appears adequate.

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FINDING:

A bandwidth or phase lag related criterion is an alternative to a rise time criterion but not as satisfactory.

DISCUSSION:

This concept was tentatively adopted at an early stage of the program but suffered from abstractness as well as a failure to correlate results as well as rise time did. Bandwidth refers to the potential a system has for being rapidly controlled in a closed loop sense. More specifically, bandwidth as used in this program is the frequency at which the open loop phase lag becomes excessive for comfortable closed loop control. If the system is controlled with a crossover frequency which exceeds the system bandwidth it is presumed that either lead compensation must be generated or that the resulting closed loop response will be too oscillatory. For

TABLE 5-4

DATA CONSIDERED FOR APPROACH RESPONSE RISE TIME

BASIC VEHICLE	$t_{1/2}$(sec)	AVERAGED PR*	SAMPLE*	REMARKS
AWJSRA	1.1	4	1	DLC used, $\tau_{eng} \doteq 0$
AWJSRA	1.7	4.6	8	
1250	2.4	3.5(2.5)	3(2)	
1240	2.3	4.5(4)	3(2)	
BR 941	2.7	5.8(3.3)	2(2)	65 kt, T out
1210	2.8	4.2(1.8)	5(3)	CTOL Technique, attitude control lag 2 sec
BR 941	2.9	5.5(3.3)	2(2)	60 kt, T in
BR 941	3.2	5.3(3)	2(2)	65 kt, T in
STOL-X	3.0	5.5(3.5)	5(3)	$\Delta\ \tau_{eng} = 2.5$ sec
1250	3.2	5	1	
1210	4.7	5.8	2	$\Delta\ \tau_{eng} = 2.5$ sec
1220	4.9	7(4.5)	2(2)	
F13	5	(3.1)	(2)	Reference 34
F14	7.4	(3.5)	(2)	Reference 34
1240	7.5	8(5)	1	STOL Technique
AWJSRA	8.9	6(3.6)	2(3)	δ_γ Primary Control
AWJSRA	8.9	5(2.5)	1(1)	DDC Primary Control

* Data in parentheses for calm air, otherwise for $\sigma_{u_g} = 1.4$ m/sec (4.5 ft/sec)

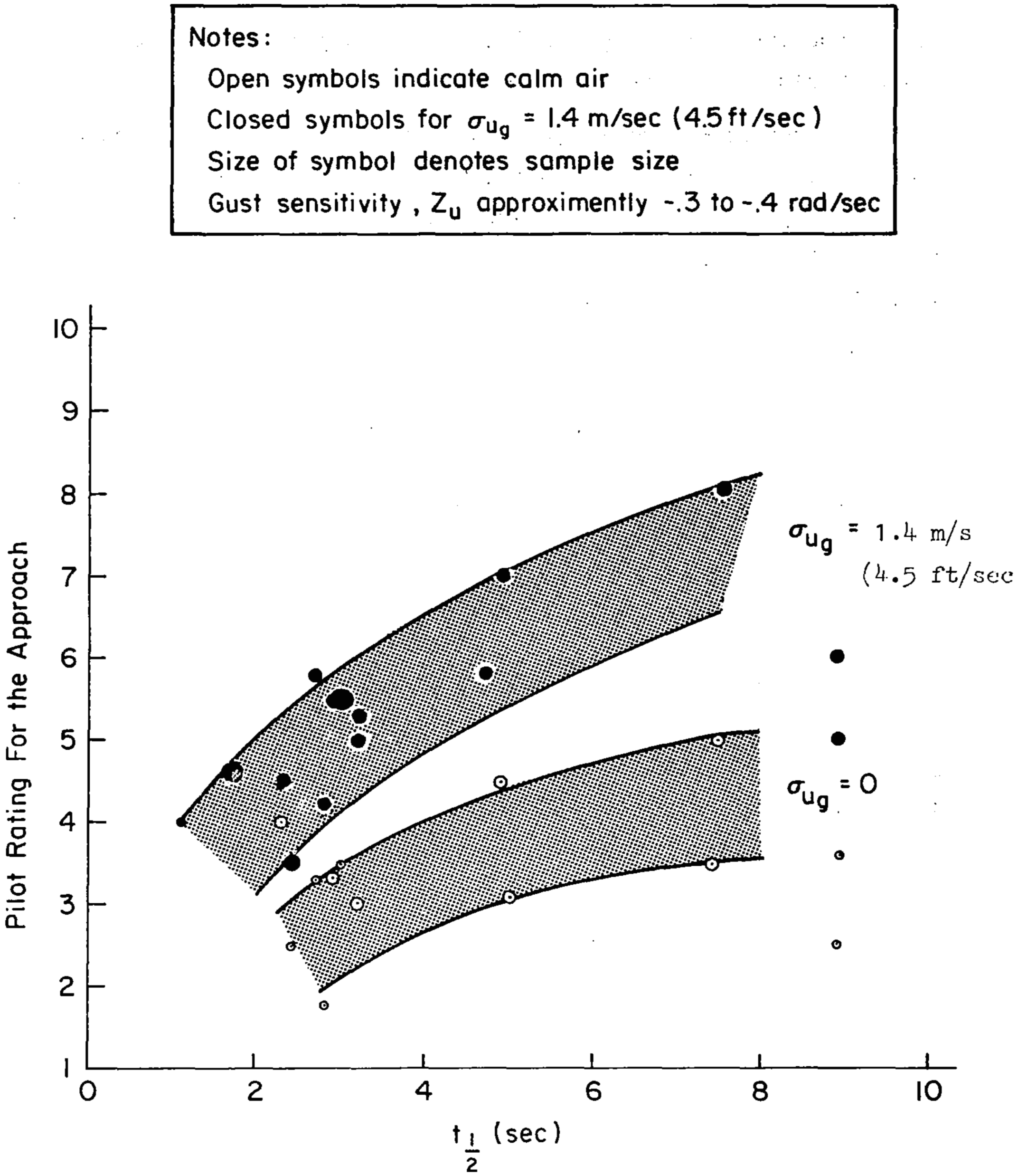

Figure 5-8: Averaged Pilot Opinion Trends Versus Flight Path Rise Time (Based on Table 5-4)

manual control, lead compensation increases pilot workload. In the other case, an oscillatory tendency is a PIO.

As a result of the BR 941 and AWJSRA simulations, a bandwidth criterion was proposed for the approach flight phase. This requirement was placed on the glide slope deviation response to the primary flight path control. The bandwidth limit proposed was a phase lag less than 135 deg at 0.25 rad/sec. It should be noted that a 135 deg phase lag for glide slope error corresponds to a 45 deg phase lag for flight path angle. One additional feature of the proposed criterion was that airspeed could be regulated to a reasonable degree in order to demonstrate the bandwidth criterion. This was further specified as an attitude or secondary control change proportional to the power change so as to minimize speed variation.

Following the Generic STOL simulations, the short term response criterion was revised. It was still expressed in terms of a phase lag limit but it did not allow for modulation of the secondary control. At the same time a rise time alternative was considered. The criteria suggested were that (a) the phase lag between the primary control and flight path angle should not exceed 60 deg at a frequency of 0.5 rad/sec, or (b) for a step input of primary control the change of flight path angle should reach 50% of its peak value within 3 seconds. At the stage these were introduced there was more emphasis on the form of expressing the criteria rather than on the validity of the numerical values themselves.

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FINDING:

Long term flight path response is an important factor, but no satisfactory criterion has been developed.

DISCUSSION:

In a conventional airplane the long term flight path response following an attitude change is a slow decay. This decay is associated with a change of airspeed. The amount of decay reflects how far on the frontside or backside of the drag curve the airplane is operating. Intuitively, such a decay should not be excessively large nor should it take place too quickly. In addition, some reasonable use of secondary control should be capable of arresting the decay.

Originally, the shape of flight path response was associated with flight path/airspeed cross coupling. While airspeed usually has something to do with the decay in flight path, it is not always so. One case in the STOL-X experiment demonstrated how a decay in flight path could take place without flight path/airspeed cross coupling. This involved use of a powerful direct lift control blended with the throttle. The flight path response was quickened in the short term but as the DLC washed out, the decay in flight path resulted. This occurred without any appreciable change in flight reference. Nevertheless, the pilot objected to the flight path decay tendency itself. Therefore, it seems reasonable that the long term flight path response be suitably limited.

No criterion was considered which directly addressed flight path response shape although a proposed cross coupling criterion indirectly did so. This will be mentioned shortly. As suggested previously, the main features to be addressed regarding flight path response shape are: how much and how rapidly flight path decay takes place following an input of primary flight path control. (Note that the so-called "creeper" condition in which flight path continually but slowly changes, is taken care of effectively by a rise time criterion.) Criteria such as time to decay to a given percentage of peak were considered but without any particular success. The availability of an easy and effective secondary control appears to be a factor. For example, a large and rapid flight path decay would not be nearly so troublesome if the secondary control were effective in countering that decay. A conditional flight path decay criterion was therefore considered at the SSDWG meeting prior to the STOL-X simulation. The proposed criterion limited flight path decay to 1/3 the peak while permitting application of secondary control not earlier than 6 sec following primary control input. The amount of secondary control was limited to that required to restore flight reference. This criterion, while addressing the important factors, could not be adequately validated with existing data. Therefore, further study in this area is needed.

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5.2.2.2 VERTICAL PATH CONTROL POWER

Levels of flight path control power initially considered were those of the basic BR 941 and AWJSRA models. This actually involved a relatively large variation in control power and led to an initial postulation of control power definition and respective limits. During the Generic STOL simulation, there was a systematic variation of 1) long term or steady state vertical path control power capabilities, and 2) short term control power capabilities. These experiments further refined limits and definitions. The STOL-X simulation was a final look at tentative control power requirements. One especially interesting finding in the post-simulation analysis involved relating steady-state control power requirements to horizontal shear protection.

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FINDING:

Vertical path control power involves both short term and long term characteristics.

DISCUSSION:

This implies that both short term and long term control power should be included in airworthiness criteria. Long term control power requirements are intuitively obvious in that they must provide for adequate flight path correction for likely variations in headwind and for sustained vertical drafts. For these sustained corrections, the pilot has time to use his secondary control to regulate his flight reference. On the other hand, for a relatively short duration gust, he will make the correction without using the secondary control. This leads to the requirement for short term control power. A short term control power requirement is not really new because past criteria involving load factor have really been a form of short term control power requirement.

The fundamental questions to be answered are:

- What definitions are appropriate for short term and long term vertical path control power?

- What constitutes short term and long term?

- What magnitude of vertical path control power is required?

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FINDING:

Long-term control power can be stated adequately in terms of a plus and minus incremental flight path angle at a constant flight reference, i.e., the vertical flight path excursion which is available while maintaining flight reference and configuration.

DISCUSSION:

Long-term flight path control power is essentially a measure of how much the pilot can maneuver about to his nominal flight path. There are a number of ways possible to describe such a characteristic. Some of the possibilities include specification of an incremental flight path angle, an incremental closure rate with flight path (i.e., velocity perpendicular to the nominal glide slope), or an incremental altitude rate. The latter two possibilities are very nearly identical for glide slope angles of interest here.

The choice of incremental flight path angle rather than incremental sink rate is somewhat arbitrary. Over the limited range of speeds investigated these quantities are essentially equivalent. Only if we were to consider either a much slower or a much faster approach speed must we worry about making a distinction between flight path angle and sink rate. During the course of this program, use of incremental flight path angle was a convenient measure of long term control power and was found to be widely used in related literature.

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FINDING:

Steady state control power should consist of a basic incremental flight path angle capability of $\pm$ 4 deg.

DISCUSSION:

Determination of incremental flight path angle requirements was made during the Generic STOL experiments and consisted of simply varying maximum and minimum power to limit the incremental flight path angle. The experimental matrix was based on observations made during the previous experiments, i.e., BR 941 and AWJSRA, and on recommendations from other sources.

Initially it was established that the range from $\pm$ 2 deg to $\pm$ 6 deg was most interesting, but further resolution was difficult. The minimum control power capability appeared to depend upon the individual pilot, how precisely he tracked the glide slope, and finally the level of atmospheric disturbance encountered.

The data shown in Figure 5-9 were used to infer the level of long term flight path control power required. The probability distribution of maximum throttle excursion for each approach run in a series of several runs in the BR 941 and AWJSRA simulations is shown. The maximum throttle excursion from each run was converted to an equivalent flight path angle change at constant airspeed from a γ - V curve. For the data shown, a 4 deg $\Delta\gamma$ capability appeared to be a reasonable choice. Therefore, it was used for planning purposes in the Generic STOL simulation, one part of which was study of long term flight path control power requirements.

For the Generic STOL simulation the overall $\Delta\gamma$ capability of $\pm$ 4 deg was considered marginal, although there was not complete agreement among the subject pilots. They did, however, agree that acceptability depends strongly upon the atmospheric turbulence level, and that evaluations in calm air can be misleading. Lack of turbulence was one problem associated with certain flight tests of powered-lift aircraft in which the flight path control power was considered. The main problem connected with severe turbulance was having sufficient control power to permit arriving at a reasonable flare window. During the Generic STOL simulation there was more concern expressed about a limited upward capability than a limited downward capability.

The final selection of a $\pm$ 4 deg requirement was reasonably well confirmed in the STOL-X simulation. On nearly all the approaches made during

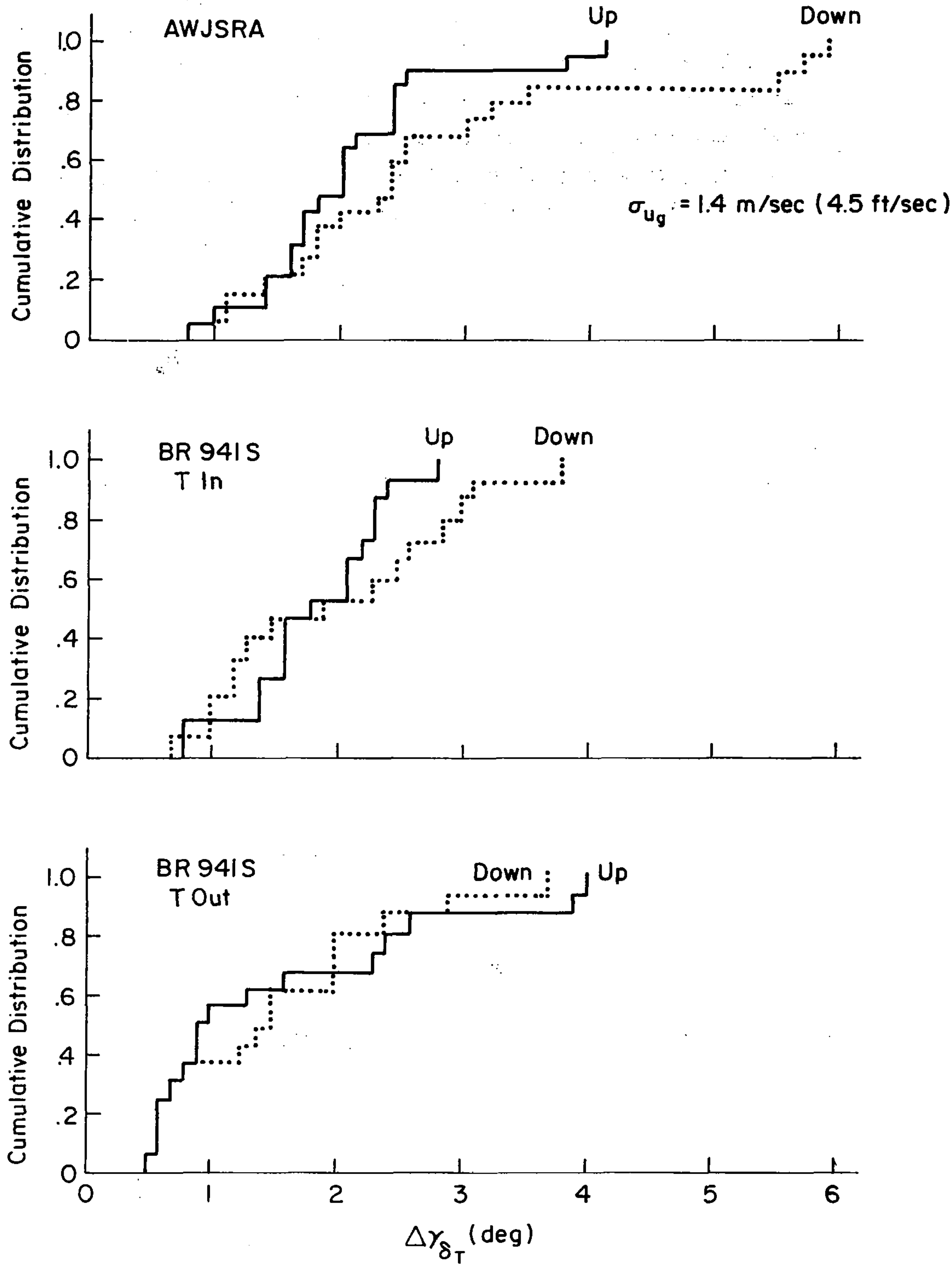

Figure 5-9: Cumulative Distribution of Maximum
Flight Path Corrections

the STOL-X experiment there was an incremental flight path limit of 4 deg on either the upside or the downside depending upon whether it was a steep or a shallow approach. Over a wide range of atmospheric disturbance levels (up to a 1% probability of exceedence) and for the large number of subject pilots used, the 4 deg value appeared adequate. On a few approaches, the combination of glide slope and headwinds restricted the pilot's up capability to something less than 4 deg. In some of these cases the pilots detected this lessened capability.

In flight tests using the TRANSALL V1 airplane flying 6 deg STOL approaches (Reference 29), the availability of approximately ± 3 deg of incremental flight path angle was judged inadequate. A value of 4 deg was felt to be desirable (and additionally, level flight as mentioned shortly).

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FINDING:

Steady state control power should include a level flight capability in addition to a basic incremental flight path angle capability.

DISCUSSION:

During this simulation program, subject pilots expressed the desire to be able to arrest sink rate during the approach without a change in configuration, i.e., that the incremental flight path angle in the upward direction be sufficient to attain level flight. This is reflected in the recommended criteria of Reference 15. In view of the desire to utilize glide slopes of 6 to 8 deg, such a level flight requirement obviously goes beyond the 4 deg incremental requirement. While the incremental 4 deg requirement is meant to provide some ability to make some corrections relative to the nominal flight path angle, a level flight requirement would appear to be more in the category of a safety margin.

Level flight capability was a feature tested in the STOL-X simulation, and was felt to be acceptable except where there was not the basic incremental 4 deg capability. Results of the TRANSALL V1 flight tests concluded that level flight capability is desirable although it was not available in the TRANSALL V1 configuration tested.

Currently, for conventional aircraft, FAR Part 25 requires flight path control power in the approach configuration in excess of level flight, i.e., a climb gradient of 3.2% (1.8 deg). In practice this requirement is normally exceeded, but it does form the basis for at least one means of comparison with a level flight requirement for powered-lift. This is shown next.

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FINDING:

The long term flight path control power can be interpreted in terms of the ability to counter a sustained horizontal wind shear.

DISCUSSION:

In order for an airplane to maintain a constant inertial flight path angle and constant airspeed in the presence of a sustained horizontal wind shear, it must have the ability to accelerate as rapidly as the wind is changing. This acceleration is equivalent to an incremental flight path angle given by:

$$\Delta\gamma = \frac{\dot{u}_g}{g} \quad \text{or} \quad \frac{\dot{u}_g}{\Delta\gamma} \doteq \frac{1}{3}\,\frac{\text{kt/sec}}{\text{deg}}$$

This is a valuable relationship to connect the flight path angle requirement to one of its primary reasons for existence, to cope with atmospheric disturbances. It also permits a direct comparison to current conventional aircraft capabilities. Finally, it gives more credence to the use of a $\Delta\gamma$ requirement rather than a $\Delta\dot{h}$ requirement since the relationship between $\Delta\gamma$ and horizontal shear is independent of airspeed.

Figure 5-10 shows the ability to counter a sustained horizontal shear for both conventional aircraft operation under FAR Part 25 and powered-lift aircraft operating under the proposed level flight requirement. Note that the comparison differs somewhat depending upon rate of descent and whether the horizontal shear is characterized as varying with altitude or varying with time. In all cases, however, there is approximately equal

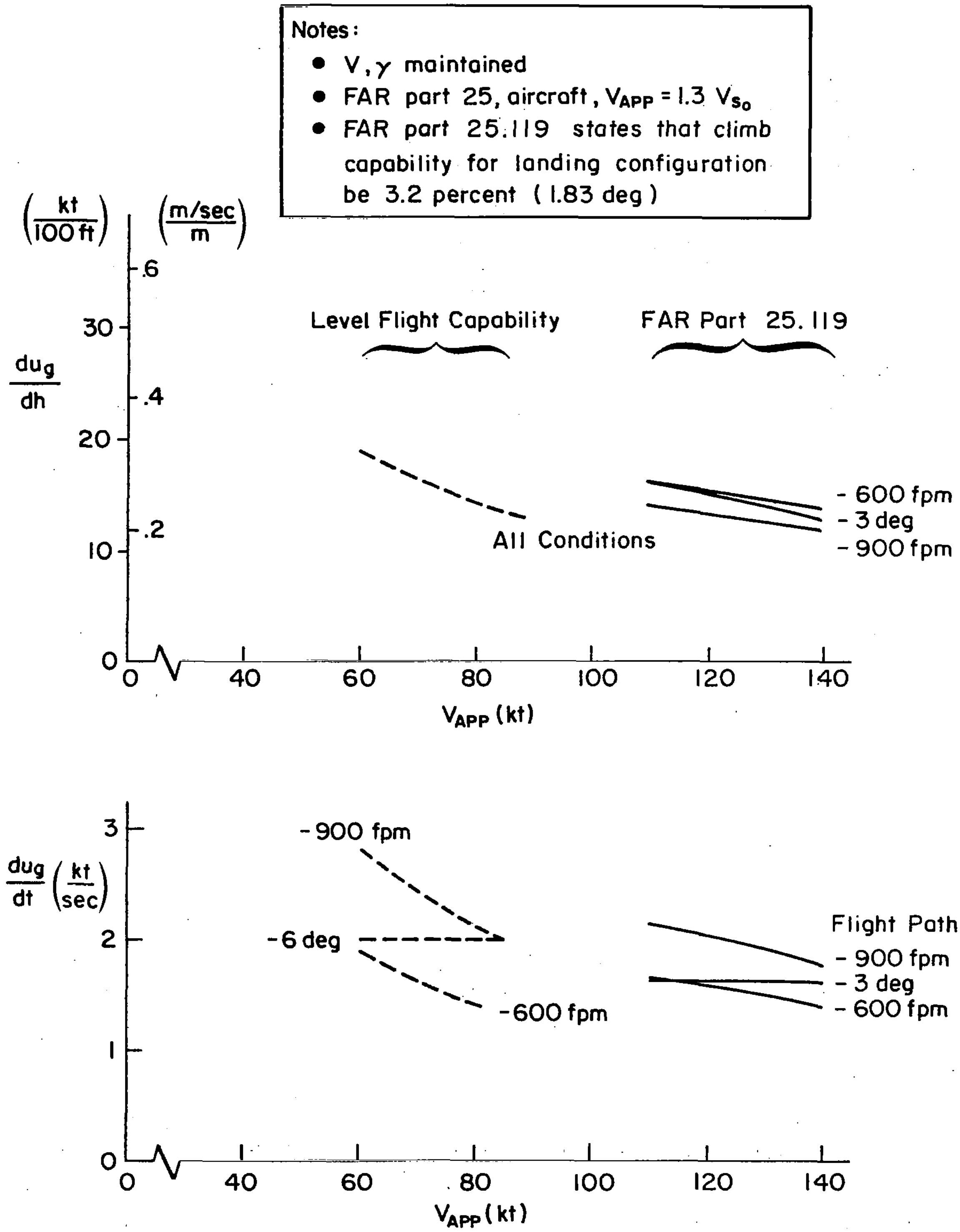

Figure 5-10: Comparative Ability to Counter Horizontal Shears in the Long Term

shear protection between powered-lift aircraft and conventional jet transports at their typical approach speeds given that the respective requirements are just met and that <u>airspeed</u> is <u>well regulated</u>. The latter assumption is not necessarily good for powered-lift aircraft as indicated in Reference 21.

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FINDING:

The short term control power can be appropriately defined in terms of a $\Delta\gamma$ within a given period following application of primary flight path control while maintaining secondary control.

DISCUSSION:

Short term control power is a measure of how much and how quickly the flight path can be changed. Quantification, therefore, must include the elements of magnitude and time. Definition of the magnitude would carry the same considerations as previously discussed for long term flight path control power. Thus it could be expressed in terms of an incremental flight path angle or an incremental sink rate. In keeping with the foregoing discussion, use of an incremental flight path angle seems reasonable.

The time element to be associated with short term control power should be consistent with the rise time requirement previously discussed. Since the required rise time, $t_{1/2}$, is the maximum time allowed for a significant <u>percentage</u> of peak response, it makes sense to tie the required <u>absolute</u> response to the same time frame.

The matter of what to do with the secondary control must also be fit into the proper time frame. In view of the fact that (i) secondary control use is associated with flight reference regulation, and (ii) flight reference regulation is carried out at a lower frequency than flight path regulation; then short term flight path change capability should not involve any significant secondary control use. Further, the most convenient assumption is that secondary control be simply held fixed.

It may be argued that incremental load factor is an alternative way of defining short term control power. This is not true, in general, because it neglects the time between control initiation and peak incremental load factor. An airplane could possess a relatively large incremental load factor capability. If, however, the time required to obtain that capability were excessive through a long lag in the primary control, then any given flight path change would be correspondingly long. In addition, the really important parameters such as Δh, $\Delta\dot{h}$, or $\Delta\gamma$ are integrals of load factor. Hence, a large peak a_z followed by a rapid washout would be ineffective.

The requirement for short term control power is really intended to plug a technical loophole and may have no effect on most powered-lift designs. This requirement is specifically intended to prohibit the following type of situations:

- Primary control response is rapid, but effectiveness is very small.
- Secondary control effect on flight path is large enough to help meet the steady state requirement, but response is very sluggish.
- The pilot cannot adequately regulate flight path in turbulent air.

These situations can be avoided by a requirement that the primary control alone provide a significant portion of the steady state control power, at least on a short term basis.

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FINDING:

The level of short term control power required is considered to be approximately 2 deg incremental flight path angle in 3 sec following application of primary control.

DISCUSSION:

Experimental determination of short term control power requirements was difficult. First, it was not possible to independently vary short term control power while holding all other flight path characteristics constant, such as steady state control power. Next, it was difficult for the pilot to determine precisely when control power was inadequate because of the random nature of the simulated atmospheric disturbances. Occasions when maximum flight path control power was needed tended to be fairly infrequent even in relatively strong gusts. Finally, there was a good deal of variation among the subject pilots in how little control power they would accept.

The cases which can be used to infer some minimum level of short term control power are given in Table 5-5. These are not only those from direct investigation of short term flight path control power, but include cases from other experiments. These were picked because they have no deficiencies in rise time or long term control power according to the criteria previously suggested. A clear pilot opinion trend with short term flight path control power is not evident. It is suspected that part of the reason is that a short term flight path control power sufficiently low to significantly degrade pilot opinion was not evaluated. It may be that a given rise time and long term control power, in practice, always provide adequate short term control power.

The flight path control features mentioned above combined to limit short term control power to at least a value of approximately 2 deg in 3 sec. The results show this level is adequate but not that a lesser amount would be so. Also, it is not known whether rise time and long term flight path control power will always limit short term control power to the level seen here. Therefore it is suggested that the level demonstrated (i.e., 2 deg in 3 sec) be adopted as a tentative requirement. Further experiments should be conducted to more precisely define the limits of this criterion.

5.2.2.3 VERTICAL PATH CROSS COUPLING

Cross coupling is an area in which progress was made in understanding but no satisfactory means of quantification was developed. One reason for this is the complex nature of coupling. Coupling can arise directly from a control input. For example, a primary control input of power can affect pitch attitude which, in turn, affects flight path and/or flight reference.

TABLE 5-5

SUMMARY OF DATA USED TO INFER REQUIRED SHORT TERM CONTROL POWER

BASIC VEHICLE	$t_{1/2}$ (sec)	$\Delta\gamma_\infty$ (deg)	$\Delta\gamma$ (3 sec) (deg)	PILOT RATING
1250	3.2	16.6	5.4	5
1210	2.8	10.8	5.4	3, 5
STOL-X	3.0	6.6	4.1	Acceptable
306	2.3	4	2.7	4, 5
309	2.3	4	3.1	4
309	3.1	4	2.3	4, 4, 5, 6
306	3.1	4	2.0	5
1210	2.8	4	2.1	4, 5
1250	3.2	4	1.7	5

Coupling can also arise in the natural response of the airplane. With pitch attitude constrained a conventional airplane has relatively easily identified airspeed and heave modes. Both are first-order modes with substantially different time constants. A powered-lift aircraft generally has more airspeed/heave coupling, especially at speeds near V_{min}. The two first-order modes can couple into one second-order mode in which case any control input will involve both airspeed and flight path responses.

In the BR 941 and AWJSRA simulation, we observed coupling as primarily a flight reference control problem. One major characteristic which was observed was the tendency to slow down as power was increased, the result of a near vertical thrust angle. Also noted along with this characteristic was the flight path overshoot tendency in which a flight path change peaked then decayed as airspeed changed. In the analysis of these initial simulation experiments we developed the μ^{STOL} concept (which is defined shortly) in an attempt to quantify cross coupling effects. Following this, in the Generic STOL simulation, we systematically varied cross coupling but found that the μ^{STOL} scheme was, in fact, based on a faulty assumption. As a result, we fell back on a simplistic time-response view of cross coupling. This also failed to adequately quantify the level of cross coupling problems. The simulation findings which are presented below will point up the highlights of the cross coupling investigations made and will elaborate on problems encountered.

The form of cross coupling of most interest in this study was that coupling between flight path and airspeed. This is an area where powered-lift aircraft tend to be fundamentally different from conventional aircraft.

Coupling between flight path and airspeed is almost entirely a function of the airframe (including any lift and drag augmentation). It can be described by considering transfer relationships between γ and V and the primary and secondary controls. A complete absence of coupling would correspond to no effect of primary control on airspeed and no effect of secondary control on flight path. Such a condition is impossible for a bare airframe and can be approached only with extensive lift and drag stability and control augmentation. Because of the likelihood of coupling and its potential impact, it does need to be considered.

Flight path-airspeed cross coupling occurs in two important ways which can easily overlap. First, coupling can take place through primary control orientation (i.e., θ_T if thrust is primary). This can cause a flight path decay problem. It corresponds to the primary control acting directly to increase flight path while decreasing airspeed. This is most obvious if $\theta_T > 90$ deg.

The other way in which coupling can occur is independent of thrust angle and only a function of the cross derivatives X_w and Z_u which are defined in Appendix A. This is where an airspeed perturbation produces a vertical force and a flight path perturbation a horizontal force. In the extreme, this condition forces airspeed and flight path to respond at the same frequency. The resulting attitude constrained motion is oscillatory rather than the usual well separated exponentially decaying flight path and airspeed modes.

Both of the above forms of coupling combine to degrade manual flight path control. The flight path response shape is a manifestation of this coupling and addresses the overshoot or decay but not the oscillatory aspect. This latter item is discussed further in the next finding.

Up to this point we have been describing flight path/airspeed coupling. Now consider flight path/flight reference coupling. There is no distinction, of course, if the flight reference happens to be airspeed. This, however, cannot be assumed in general.

Flight path/flight reference coupling depends on how the flight reference is mechanized as well as the airframe dynamics. Where the impact of flight path/airspeed coupling is mainly on flight path control, the impact of flight path/flight reference coupling is on both flight path and flight reference. In Section 5.2.3 we will discuss this aspect of cross coupling in detail.

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FINDING:

The term "cross-coupling" can refer to a number of different effects which are visible to the pilot and influence his workload and performance.

DISCUSSION:

Cross coupling is, in itself, an imprecise term. In viewing the powered-lift configurations of this program, however, only two prominent forms of cross coupling were identifiable by the subject pilots.

The first of these was the effect of a throttle change on pitch attitude. This is simply the result of effective thrust vector being offset from the c.g., but is not easily generalized in terms of sign or magnitude for powered-lift airplanes nor is it a function of geometry as in the case of conventional aircraft. There was no intentional variation of this effect in the configurations considered. In general, such coupling was relatively weak, especially in those cases involving a pitch attitude hold SAS. Nevertheless, even small amounts of throttle-to-attitude coupling were readily perceived by the subject pilots. The largest amount of throttle-to-attitude coupling was experienced in the STOL-X baseline configuration. This case did not turn out to be objectionable and, in fact, it was desirable because it was a favorable form of coupling. That is, when the throttle was varied, the airplane pitched in such a way as to help maintain the desired flight reference. In fact when an attitude hold SAS was added the pilot opinion actually worsened somewhat because this favorable coupling was not present. Reference 32 describes the effects of this kind of coupling for an AWJSRA simulator experiment.

The second form of cross coupling obvious to the subject pilots was the unusual effect of throttle on airspeed. Because of the vertical thrust orientation there was a distinct tendency for airspeed to decrease for an increase in throttle. Notable examples were the AWJSRA and Generic STOL configuration 1250.

This form of coupling is normally associated with a flight path overshoot tendency. That is, a step change in throttle results in a corresponding change in flight path followed by a decay. (Recall the set of responses shown in Figure 5-5.) Since this form of coupling was viewed as a direct consequence of powered lift, it received considerable attention. One analytical approach is described next.

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FINDING:

The theoretical cross coupling parameter termed μ^{STOL} was not an acceptable metric of cross coupling between flight path and flight reference.

DISCUSSION:

The μ^{STOL} parameter was adapted from a mathematical approach introduced to the control system field by E. H. Bristol in the mid-1960's (Reference 33). This parameter provided a relatively simple measure of how airspeed regulation affected flight path control and vice versa. It was defined as the ratio of flight path response with throttle at constant attitude to the flight path response with throttle at constant airspeed, and was a function of frequency.

$$\mu^{STOL} \triangleq \frac{\left.\dfrac{\partial\gamma}{\partial\delta_T}\right|_{\theta=\text{const}}}{\left.\dfrac{\partial\gamma}{\partial\delta_T}\right|_{V=\text{const}}}$$

The optimum value was unity which meant that airspeed regulation would not alter the flight path response. It also meant the airspeed to throttle response was zero. (See Reference 33 for additional details and a thorough discussion of the implications of μ^{STOL}.)

The μ^{STOL} concept was investigated in a systematic manner during the second Generic STOL simulation. A series of airplane configurations were considered whose high frequency and low frequency values for μ^{STOL} were varied according to Table 5-6. Accompanying characteristics, including time response for step flight path changes and γ - V contours for the two controls are also shown. In general, there was simply not a clear correlation between pilot opinion of flight path and flight reference response and the μ^{STOL} values. For example, configuration 1270 should have been the most severely coupled case in terms of its μ^{STOL} description. However, it was judged as good or better than the supposedly ideal case, 1210. A detailed account of these results is included in Reference 13.

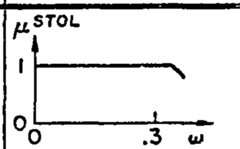
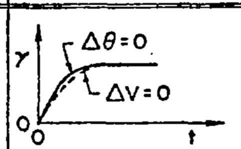
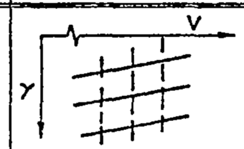
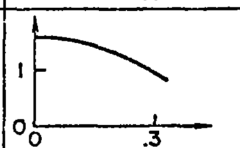
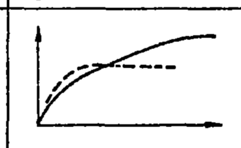
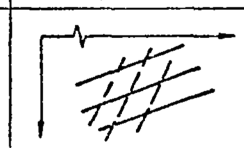
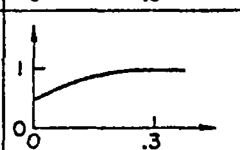
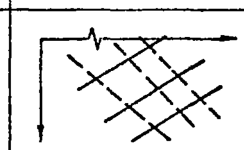
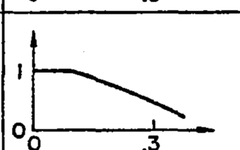
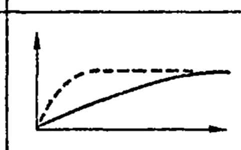
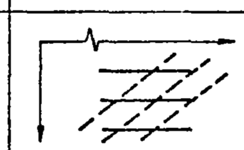
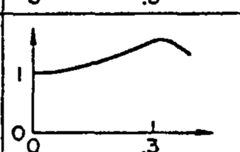
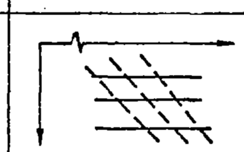
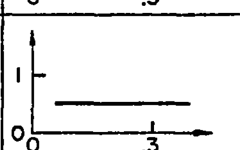
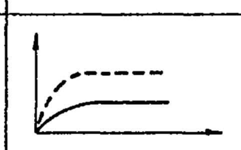
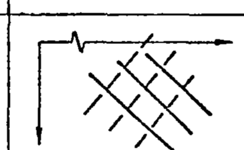
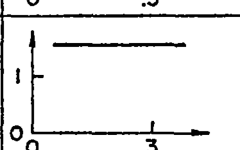
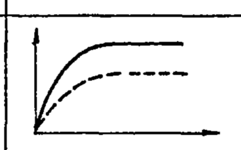
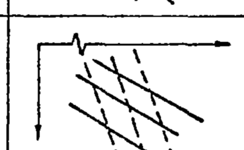
CONFIGURATION	μ^{STOL} vs ω	γ RESPONSE TO STEP δ_T	γ vs V — Constant δ_T --- Constant θ	PILOT RATINGS (CALM AIR/WITH TURBULENCE)
1210				1-2/3-5
1220				4-5/6-8
1230				3-4/5
1240				4/5 (CTOL Technique) 5/8 (STOL Technique)
1250				2-3/3-4
1260				1.5/2
1270				/3

TABLE 5-6

CONTROL CROSS COUPLING TEST CASES

There are three main problems with the μ^{STOL} parameter. First, the pilot is not likely to regulate airspeed tightly. Airspeed does not necessarily need to be the flight reference, but even if it were, the regulation would be at a lower bandwidth than flight path. Hence the basic assumption of perfect speed control is faulty.

The second problem is that μ^{STOL} may be an overly simple or inappropriate measure of coupling. For example, it does not directly measure airspeed excursion due to flight path control or problems in regulating flight path or airspeed. This is the risk in relying on a single parameter to describe a complex relationship.

Finally, μ^{STOL} would be difficult to measure, or at least its non-steady aspect. The steady state value is a simple function of $\frac{\partial \gamma}{\partial V}$ for constant attitude and throttle. The value for a given frequency would probably require mathematical manipulation of time response data.

For these reasons along with the results of the Generic STOL simulation, the μ^{STOL} parameter was dropped from consideration.

It should be noted that flight path/airspeed coupling was addressed in the AWJSRA simulation program reported in Reference 34. While the results involved only calm air conditions, a strong pilot opinion effect is nevertheless shown for the steady state metric:

$$\left.\frac{\partial u}{\partial \gamma}\right|_{\theta}$$

This will be further studied in forthcoming flight tests using the AWJSRA and may provide a key to effectively defining flight path/airspeed coupling limits.

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FINDING:

A limit may be required for coupling of flight path/airspeed modes.

DISCUSSION:

As mentioned earlier in this subsection, attitude constrained dynamics can appear as a coupled oscillatory mode rather than the usual two separated exponentially decaying modes. The amount of such coupling can be expressed in terms of the damping ratio of that oscillatory mode.

It is suspected that a lower limit on this damping may be necessary to insure acceptable closed loop flight path control. Based on pilot/vehicle analyses one can show that too low a damping ratio prevents tight flight path control. On the other hand, this feature may be effectively governed by other requirements such as short term response rise time.

It is suggested that damping of a coupled flight path/airspeed mode be specifically limited until such a limitation is found to be redundant. Based on results from this program and those of Reference 10 a tentative limit would be approximately 0.6 to 0.8 critical damping. Cases having a value less than 0.6 were never acceptable. A value of 0.8 was the lowest value connected with an acceptable vehicle.

5.2.3 FLIGHT REFERENCE CONTROL

As mentioned in earlier discussions, flight reference control consists of a relatively loose outer loop the objective of which is to keep the flight condition within safe margins.

In this section we will discuss the relatively qualitative results concerning flight reference control. This will consist of a chronological account of flight reference related experiments and how they were viewed over the course of the program.

During the BR 941 simulations indicated airspeed was used mainly as the flight reference quantity but there was also some limited use of angle of attack. During these simulations we noted comparative features such as a PIO tendency if angle of attack were tracked too tightly. Also we gave the pilot both indicated airspeed and angle of attack simultaneously. During the AWJSRA simulation indicated airspeed was used as flight reference throughout. During the Generic STOL simulation we again assumed the indicated airspeed as flight reference but found that the pilot was not always willing to closely track this flight reference if it was not necessary to preserve adequate margins. Finally, during the STOL-X simulation, a qualitative exploration of several facets of flight reference control was made.

The following are the findings related to flight reference control.

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FINDING:

Flight reference can be a multi-variable function.

DISCUSSION:

This is an important idea because it implies that airworthiness standards must be formulated in terms general enough to handle such a definition of flight reference. During the early simulations in this program, flight reference was generally assumed to be airspeed but angle of attack was used occasionally. To a limited extent either airspeed or angle of attack was appropriate to the BR 941 and AWJSRA configurations but this was not so with some of the more unusual cases considered during the Generic STOL simulation. In some cases it was found that use of a constant airspeed flight reference imposed an unnecessarily difficult workload on the pilot, and that a flight reference such as constant pitch attitude was a better alternative.

The concept of flight reference was then further expanded to include any of a number of generalized parameters. One such flight reference was used extensively in the STOL-X simulations. In this case, a generalized flight reference was used which preserved a constant speed margin. This flight reference was a linear combination of angle of attack and throttle position.

In general, the flight reference, displayed on a single gauge, could consist of a function of any number of variables contrived to help the pilot maintain a safe and effective operating point.

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FINDING:

There must be only a single flight reference quantity for the pilot to regulate.

DISCUSSION:

The primary supporting data for this finding came from the original BR 941 simulation. During some runs both indicated airspeed and angle of attack were available as a flight reference, that is, the pilot was given both a nominal airspeed and a nominal angle of attack. This resulted in confusion and increased workload because of frequent conflicts. The pilots noted that they had to choose one or the other to regulate. They frequently found simultaneous airspeed and angle of attack excursions which were indicating pitch corrections of opposing signs.

This is not to discourage or prohibit display of status information other than flight reference. When flying an airspeed flight reference it was found that pilots did, in fact, like to monitor angle of attack for the purpose of margin indication, but would make angle of attack corrections only if the excursions became excessive.

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FINDING:

Several different factors must be considered in selecting a flight reference mechanization from among the numerous possibilities.

DISCUSSION:

Flight reference can be mechanized several different ways, even for the same safety margin criteria. Consider the usual γ - V plots with contours of constant power, constant pitch attitude, and constant angle of attack. Any trim point in the plot can be specified in terms of any two of the five variables:

- Flight path angle, γ
- Airspeed, V

- Power, δ_T
- Pitch attitude, θ
- Angle of attack, α.

A family of desired trim conditions (subscript c) could be uniquely defined by any one of 10 plots, such as V_c versus δ_{T_c}, θ_c versus V_c, or γ_c versus α_c. Each of these curves could then be used for two different flight reference mechanizations. For example, the curve of V_c versus α_c could provide a flight reference of $V - V_c\ (\alpha)$ or $\alpha - \alpha_c\ (V)$.

All of these flight reference schemes provide the same trim conditions but differ in other important considerations, including:

- Dynamic (or short term) response to control inputs
- Sensitivity to atmospheric turbulence
- Sensor and computational complexity
- Effects of accelerated flight.

While all of these are important considerations in selecting a flight reference mechanization, the last one may be the most critical.

Even conventional aircraft have a flight reference problem in accelerated flight. Student pilots are carefully instructed about the effects of a steady turn on stall speed. This is one reason some recommend using angle of attack instead of airspeed as the flight reference. The problem is more severe in a powered-lift aircraft because even the maximum angle of attack may not be constant.

For powered-lift aircraft another important accelerated flight condition is operation in a wind shear. To maintain constant airspeed in a wind shear, the aircraft must have an inertial acceleration equal to the wind acceleration. Because the aircraft is accelerating, the usual γ - V plot is no longer valid. However, the effect can be approximated by using an effective flight path angle which is defined by:

$$\gamma_{eff} = \gamma_a + \frac{\dot{u}}{g}$$

where γ_a = flight path angle relative to the air mass

This allows one to use the γ - V plot to establish approximate trim conditions in a wind shear. The contours of constant α and δ_T are still valid but the constant θ contours are shifted. Pitch attitude is always given by $\theta = \gamma_a + \alpha$ not $\gamma_{eff} + \alpha$; therefore, a wind shear shifts the θ contours by $\dot{u}/g$. The net result is that pitch attitude may not be an appropriate flight reference in a wind shear. Holding the desired pitch attitude in a wind shear could substantially reduce safety margins.

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FINDING:

Flight reference dynamics are visible to the pilot and affect his workload.

DISCUSSION:

Some effects of differing flight reference mechanizations were demonstrated during a brief experiment with the STOL-X simulation. For the baseline airplane the flight reference was mechanized using angle of attack and throttle setting*. So far as the pilot could tell, the flight reference appeared to be a typical angle of attack indication. Movement of the throttle had a weak but linear effect on the flight reference indication. For this particular airplane configuration, this flight reference mechanization was acceptable, even though the flight reference display was somewhat noisy when flying in turbulence.

As an alternative, a mechanization was employed which used pitch attitude in combination with throttle position**. If the flight reference needle were zeroed, the resulting trim condition would be identical to the baseline case. The fact that pitch attitude and throttle were used, however, eliminated the noise characteristic of an angle of attack indication. The flight reference indication itself was very steady and, in effect, provided the pilot with a pitch attitude command via the flight reference gauge. Hence, if the pilot increased his throttle setting to go up, the flight reference gauge immediately moved to an indication which could be interpreted

* Flight reference was $\alpha - \alpha_c$ where $\alpha_c = f(\delta_T)$

** Flight reference was $\theta - \theta_c$ where $\theta_c = f(\delta_T)$

directly as the pitch attitude change required to maintain the desired flight reference in the long term. It was easier for the pilot to fly this simply because the short term dynamics of the displayed flight reference were somewhat improved over the baseline flight reference.

Unfortunately, the effects of wind shear were not fully appreciated at the time of the experiment. No tests with a definite shear were performed but subsequent analysis indicated that safety margins could be significantly reduced with the second (pitch attitude) flight reference scheme.

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FINDING:

Manual flight reference regulation is looser than flight path regulation and cannot be used to enhance the short term flight path response.

DISCUSSION:

One feature of the cross coupling experiment during the Generic STOL simulation was that each of the configurations had equivalent flight path response potential provided that airspeed was well regulated. For example, in case 1220 which had approximately a 45 deg thrust inclination, flight path response to a throttle input was very slow. If the pitch attitude was immediately increased to offset the speed increase when power was added, then the flight path response was much faster.

It was found, however, that when the pilot had to depend upon this rapid and simultaneous regulation of flight path and flight reference in rough air and near the end of his approach, the workload simply became too high. In this case he reduced or dropped flight reference control and regulated flight path using only the throttle. Both of the pilots who flew this configuration used this control technique and got into a flight path PIO. Knowledge that tight airspeed regulation would relieve this was of no help. The workload involved in doing so was simply too high. This, then, is an illustration of the need for basic flight path control standards through use of primary control only.

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SECTION 6

LONGITUDINAL STABILITY, CONTROL, and PERFORMANCE; LANDING

This section covers longitudinal stability, control, and performance for the landing flight phase, i.e., that part of flight beginning with flare initiation and ending with touchdown. Landing is handled in a similar way to approach. In particular, the landing is analyzed in terms of piloting technique and control functions. The landing, like the approach, is primarily a vertical path control problem. In fact, it is more so since active flight reference control is not involved. We begin by discussing key differences in the landing task between conventional and powered-lift aircraft. This is followed by presentation of simulator results.

In the section dealing with piloting technique, we see that, just as in the approach flight phase, piloting technique can be classified in terms of the primary control. In the landing, though, there is much less emphasis on the secondary control. Additionally, the landing task is defined in terms of certain performance features, such as flare height, target touchdown sink rate, and target touchdown point. It is important to note that flare can include the range from full flare (nearly zero touchdown sink rate) to no-flare. One additional degree of freedom in landing technique offered by powered lift is use of power to flare. This discussion involves development of the means to treat power-to-flare as well as the conventional attitude-to-flare technique.

The control functions to be considered are pitch attitude and flight path. Flight reference regulation is not really involved in landing even though the initial flight reference value is important. The impact of powered lift on pitch attitude and flight path control is generally the same in the landing flight phase as in the approach. The main effect is primarily that of high lift coefficient and low airspeed. The vertical thrust inclination feature of powered lift is also important in the case of flaring with power.

Landing experiments which were conducted in this simulator program followed the same general outline described in the approach section. The program was begun using detailed models of actual aircraft. This led to more specially designed experiments in which critical landing parameters were varied. Finally, tentative criteria were tried on a specific aircraft design, STOL-X. In general, the landing was conducted as a part of the overall approach and landing task. This meant that the pilot would arrive at the flare initiation point having flown a representative approach task involving various adversities. The pilot was always given a prescribed flare technique along with some indication of target touchdown objectives. Landing rollout was frequently accomplished, although it was not really a part of this simulation program.

6.1 PILOTING TECHNIQUE; LANDING

The following is a presentation of ideas related to the role of piloting technique in longitudinal stability, control, and performance for the landing flight phase. As for the approach, this will be in preparation for the subsections on longitudinal control functions. The points to be covered here are pilot objectives and the means to describe piloting technique in the landing. This will be followed by a few of the more general findings relating to piloting technique.

As in the approach, the primary piloting task during the landing is that of vertical path control. It is, however, more of a terminal control problem since the goal almost totally is to arrive at a set of desired touchdown conditions. Most likely the main objectives are obtaining a target touchdown sink rate and a target point on the runway. In a conventional aircraft the emphasis is frequently on the former. In a powered-lift aircraft the emphasis is more on the latter if a STOL environment is involved. There are, of course, other concerns at the point of touchdown, such as maintenance of reasonable safety margins, landing gear geometric and structural constraints, etc.

In the approach or landing, pitch attitude control is the longitudinal task subordinate to flight path control.

The term "flare technique" refers to the specific way in which the cockpit controllers are used to accomplish the landing task. Flare using pitch attitude is analogous to a conventional piloting technique and flare using power analogous to a STOL technique.

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FINDING:

The likely techniques for landing powered-lift aircraft include both flare using pitch attitude and flare with application of throttle.

DISCUSSION:

The choice of a particular flare technique depends greatly on vertical path control characteristics available. Given suitable characteristics, the subject pilots in this and the Reference 10 simulation programs demonstrated the ability to perform flared landings with either pitch attitude or throttle as the primary flare control.

Flared landings using pitch attitude as the primary control appeared identical to those performed in conventional aircraft so far as general piloting technique is concerned. In general, pitch attitude varied nearly linearly with altitude after flare initiation. The subject pilots characterized this as a closed loop task to the extent allowed by the visual display and controllability of the aircraft.

Flares using throttle were initially tried during the BR 941 simulation (Reference 11). Two pilots were involved in this short experiment, one had considerable helicopter background and the other virtually none. The pilot with helicopter experience quickly adjusted to flares with throttle. The similarity to a flared helicopter autorotation maneuver seemed to be a factor. The pilot not having helicopter experience was reluctant to endorse this technique. After trying this again with a significant reduction in engine lag, the second pilot then agreed that the technique might be feasible after a suitable training period.

During the Generic STOL and STOL-X simulations (References 13 and 14) flare using throttle was investigated further and, in fact, was used as the normal prescribed technique for several cases including the STOL-X model. It was found that all the subject pilots could adopt use of this technique provided vertical path dynamics were adequate.

It was noted that in flight tests with the NC-130B (Reference 24) flare using throttle was tried but was found to be an unsatisfactory technique. This appeared to the pilots to be due to long engine lags. Based on what was learned in this simulation program, however, it is felt that the problems were mainly in the path dynamics related to the effective thrust angle, and not in the inadequacy of the technique itself.

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FINDING:

It is reasonable to permit a flare technique involving open loop application of the secondary control.

DISCUSSION:

Initial simulation experiments (References 11 and 12) involved the use of only one flare control at a time, either pitch attitude or throttle. During the first Generic STOL simulation (Reference 13) a series of experiments was run in which the pilot was allowed to use an open loop application of the secondary control. Some of the results are important with regard to piloting technique in the flare. It is important to point out, however, that the results may not have completely general applicability. The nature of the airplane dynamics plays an important role as mentioned previously.

It was found that flare using attitude could be aided by an open loop application of throttle. The most direct benefit was in reducing the maximum pitch attitude excursion by initially breaking the sink rate with throttle. This can be important for powered-lift aircraft because low n_{z_α} combined with a large glide slope angle requires relatively large pitch excursions.

It is difficult to add a precise increment of throttle because there are no direct cues other than engine noise. The difficulty in adding thrust to aid in flare is its likely inconsistency of application. It should be

Noted that this difficulty is present to some degree in flares using pitch attitude alone owing to the variation in throttle during the approach, which in turn creates some dispersion at flare initiation thereby altering the initial conditions of the flare.

Flares using throttle were not, in general, aided by an open loop application of pitch attitude except where it was necessary to establish a level attitude for touchdown (i.e., where the approach attitude was less than nose level). The main difficulty was similar to that mentioned previously in that application of pitch attitude introduced some variability in amount of throttle required to flare.

In all cases where the pilot was allowed to use an application of secondary control he did so prior to initiation of the flare with primary control. That is, the main segment of the flare maneuver was a single control operation. This was also noted in Reference 10. There is good reason for this being the natural inclination. It allows some of the transient effects of the secondary control application to settle before the pilot begins closed loop application of the primary control. This is the same reason that a precise measure of secondary control input is desirable.

This aspect was further explored in the STOL-X simulation. For this model the nominal flare technique was prescribed as use of throttle as primary flare control with an open loop change in pitch attitude sufficient to clear the nosewheel at touchdown. This technique was suited to the path dynamics of the model and was found acceptable by the eight subject pilots.

6.2 LONGITUDINAL CONTROL FUNCTIONS; LANDING

The two landing control functions considered here are pitch attitude and vertical path. Flight reference control is not included because it is a low frequency function which is not significant in the comparatively short term flare maneuver.

6.2.1 PITCH ATTITUDE CONTROL; LANDING

This section presents ideas associated with pitch attitude control in the landing flight phase and closely parallels the corresponding section for the approach.

The importance of pitch attitude control in the landing is almost entirely dependent upon the flare technique. If the flare is performed using pitch attitude as the primary control then pitch control characteristics need to be considered. If flaring with power, pitch attitude control is considerably less important.

As mentioned previously, pitch attitude control was not a specific topic of investigation. Variations in pitch attitude control characteristics were strictly an indirect result of variations in other control functions.

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FINDING:

When pitch is the primary flare control, pitch attitude control involves the same factors as for conventional aircraft; it differs only in the magnitude of the pitch attitude change.

DISCUSSION:

When pitch attitude was used as the primary flare control, the nature of the pitch maneuver itself was essentially identical to that observed in conventional aircraft with one notable exception, the magnitude of the pitch change was noticeably larger. This is illustrated in Figure 6-1 in which profiles of θ versus h are presented for various types of aircraft. The flare profiles for a simulated powered-lift vehicle are typical of most of the configurations involved in this simulation.

The common feature of all the plots is that pitch attitude varies approximately linearly with altitude after flare initiation. Only minor fluctuations from the linear trend occur. Also, each of the cases are characterized by a reasonably well defined flare initiation height. This is a convenient marker for the beginning of the maneuver. Correspondingly the

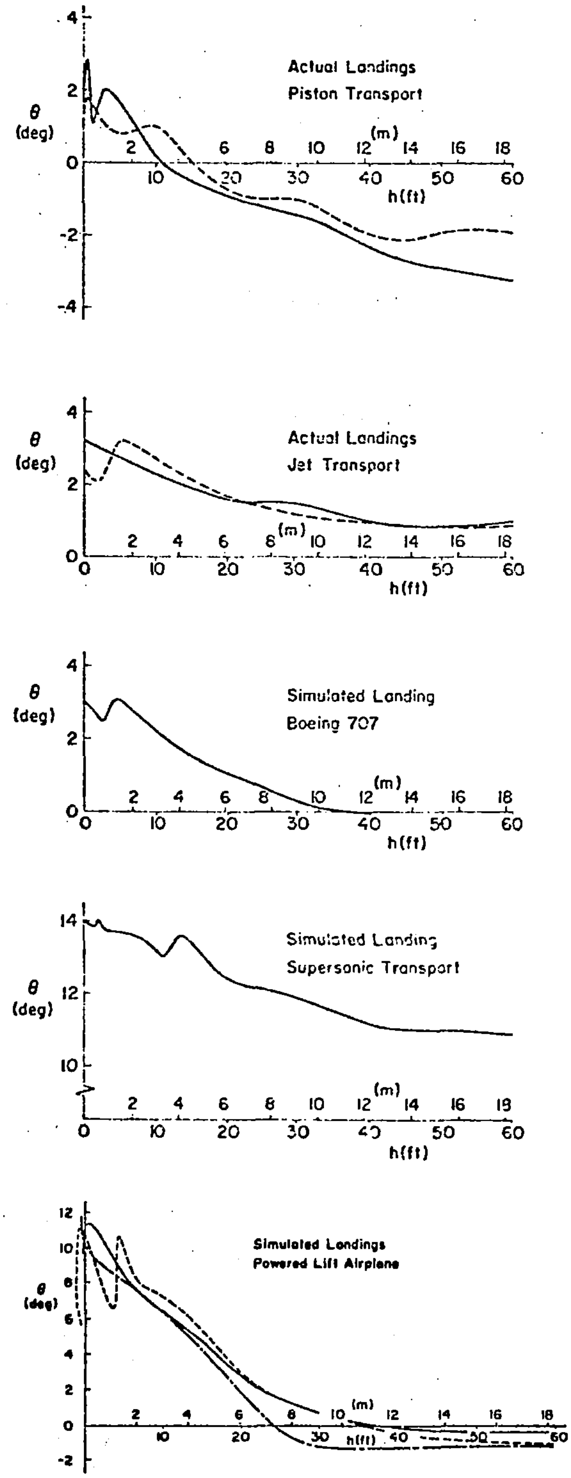

Figure 6-1: Comparison of Flare Maneuvers for Several Aircraft

pitch attitude increment from the approach to touchdown is another convenient metric. These features will be important in the subsequent section dealing with vertical path control.

The large pitch attitude required for powered-lift aircraft is mainly the result of a lower n_{z_α}. Since approach sink rates for conventional and powered-lift aircraft are comparable, then the level of normal acceleration required to flare should be about the same. Hence, for a lower value of n_{z_α} a correspondingly larger pitch attitude is required to obtain a given normal acceleration.

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6.2.2 VERTICAL PATH CONTROL; LANDING

This subsection treats the main longitudinal task in the landing, the vertical path aspects of the flare maneuver. The vertical path control characteristics of powered-lift aircraft are described and compared to a conventional aircraft in the approach section. The same characteristics apply to the landing, therefore, this will not be repeated.

The following is a description of the manual flare maneuver. This description helps in identifying the key features in the airplane which are related to the flare. This is a mathematical model which seems to bear a strong resemblance to the nominal flare maneuver as observed in simulations as well as actual flight situations. The main objectives for this report are to develop a comparison of features of powered-lift and conventional aircraft. For background and details of this mathematical device the reader is referred to Reference 12. We consider first a flare maneuver using pitch attitude control, then extend the same ideas to flare using throttle.

The key features of the manual flare maneuver are described in the previous subsections with regard to piloting technique and attitude control. To recap, below the flare height, h_{FL}, pitch attitude varies nearly linearly with altitude. This implies the flare maneuver can be modeled by a simple linear feedback of altitude to commanded pitch attitude. Thus, given initial conditions on the trimmed approach, a flare height, and the pitch increment between approach and touchdown, $\Delta\theta$, it is possible to compute a nominal flare maneuver. This includes solving for touchdown conditions such as sink rate, position on the runway, airpseed, and angle of attack. The details of these calculations are given in Reference 12.

The following example shows the various features of the flare maneuver computed for a typical powered-lift airplane compared to a conventional jet transport. The two cases are the same ones used in the example of approach flight path dynamics of Section 5. The parameters picked to describe the flare maneuver in each case are typical of those observed in actual flares. There was an attempt, however, to match the abruptness and duration of the two cases to provide a more direct comparison. The respective maneuvers are defined in Figure 6-2.

A comparison of the main features of flares between powered-lift and conventional airplanes (using an attitude control) is shown in Figure 6-3. This figure shows time histories for altitude, rate of descent, airspeed change, angle of attack change, and horizontal distance relative to the approach aim point (with no flare, the aircraft would land at the aim point).

The similarity of the vertical path changes is shown in the time histories of altitude and rate of descent. The resemblance of the two examples is great and is typical of what one observes from piloted flare maneuvers.

One minor but nevertheless realistic difference should be noted in the sink rate time histories. Near the end of the flare, sink rate for the conventional airplane tends toward zero while for the powered-lift case it levels off and starts to bend slightly upward. This is indicative of the tendency for the conventional airplane to float while the powered-lift airplane continues to settle thus touching down more firmly. A more pronounced difference would occur if the powered-lift airplane were characterized by an even lower n_{z_α} or lower heaving damping. This is discussed in more detail shortly.

The time history differences are not nearly so subtle for the remaining three variables, airspeed, angle of attack, and horizontal distance. The

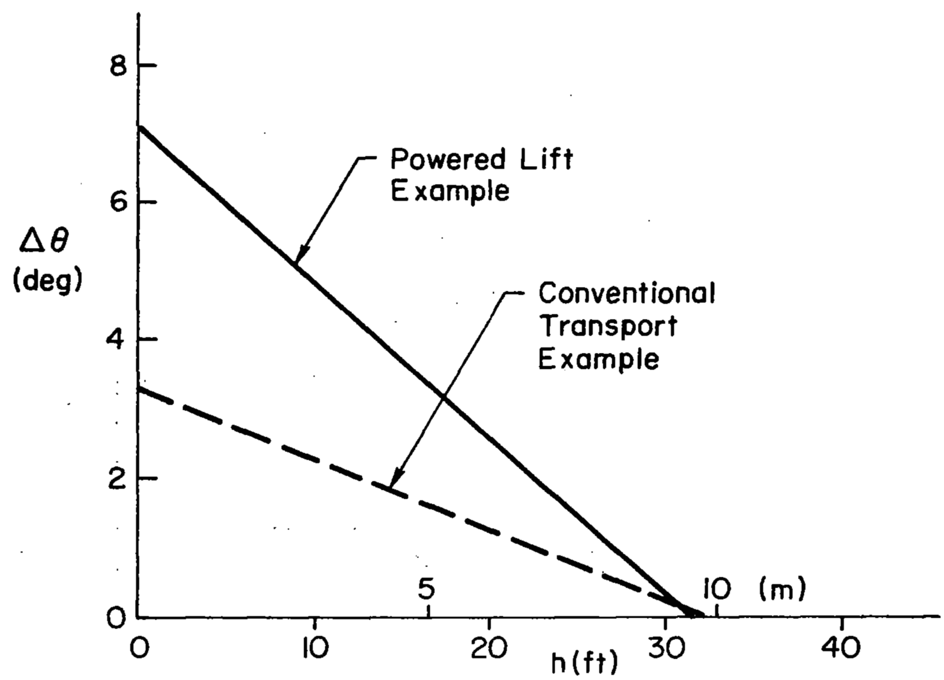

Figure 6-2: Definition of Flare Maneuvers used in Flare Comparison

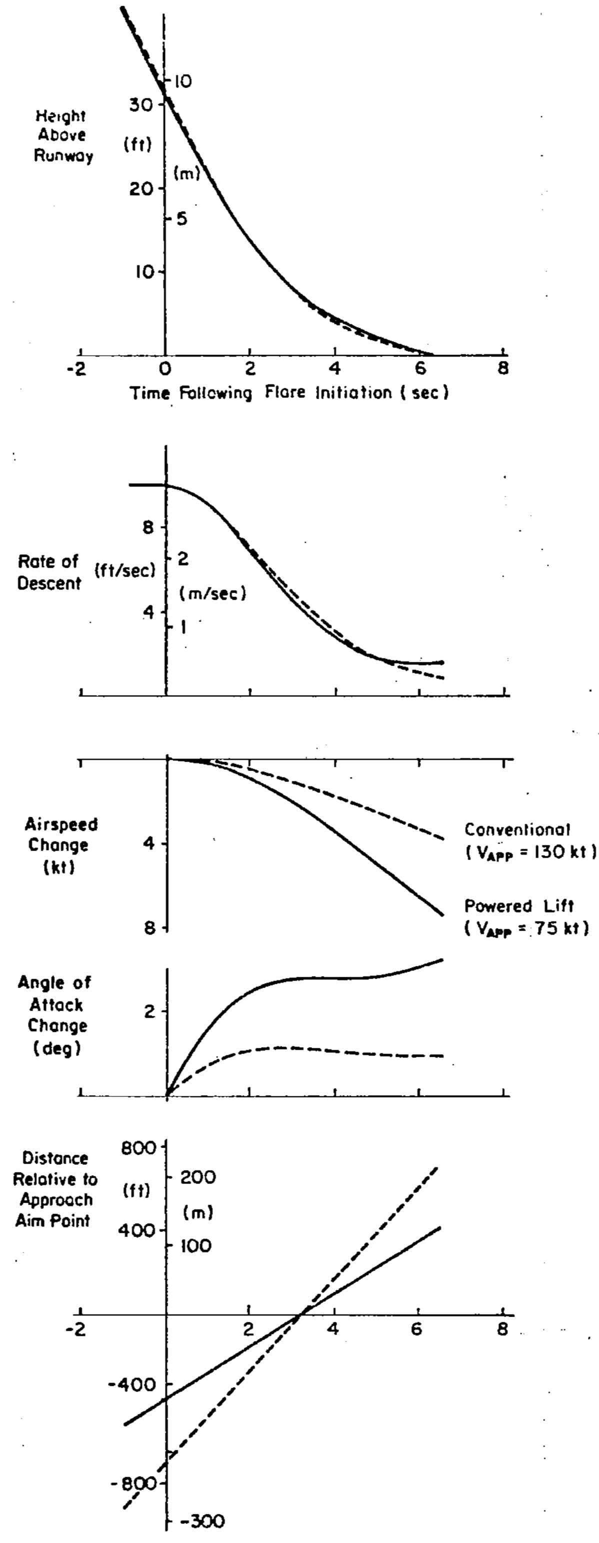

Figure 6-3: Comparison of Aircraft Catagories with Regard to Flare

airspeed change for the powered-lift airplane is nearly twice as great while the original approach speed was about one half that of the conventional airplane. The main factor governing this relationship is simply the ratio of n_{x_α} to n_{z_α}. Since n_{x_α} tends to be relatively constant between these two types of airplanes the main determining factor is therefore simply n_{z_α}.

The angle of attack change is also significantly greater for the powered-lift airplane. This is really the result of the lower n_{z_α} which requires a larger angle of attack change to get the same vertical acceleration. Note, though, that the angle of attack excursion is significantly less than the attitude excursion shown in the previous figure, i.e., the flare is not a high angle of attack maneuver.

Finally, the horizontal distance traveled during the flare is greater for the conventional airplane even though, in this example, the flare duration is the same. The difference here is solely due to the difference in approach speeds since the speed, even at touchdown, is not much less than the initial speed.

Thus, the above set of examples provides an overview of the flare maneuver using attitude as the primary flare control. Airspeed, angle of attack, and distance relationships are shown within the context of a realistic control of flight path.

Now we will consider the aspects of flight path control more closely. In a flare where attitude is the primary control the key to describing vehicle dynamics is simply the $\dot{h}/\theta$ response. This is most easily described in terms of a transfer function and is discussed in detail in both Section 5 and in Appendix A.

The dominant features of flight path due to attitude change can be summarized as the following:

- Sensitivity of vertical acceleration to attitude change, n_{z_α}
- Lag in initial response as influenced by heave damping, $Z_w^\dagger$
- Flight path decay as influenced by speed damping and the degree of backsidedness.

One should expect that by suitably limiting each of these three elements, the flight path dynamics with regard to flare would also be suitably constrained.

It can be shown using the flare analysis method of Reference 12 that the closed loop flare dynamics are really a function of the first two items (n_{z_α} and $Z_w^\dagger$) and that the third (backsidedness) is considerably less important except in extreme cases. Further, the first two items are related by a simple relation:

$$Z_w^\dagger \doteq -\frac{g}{V} n_{z_\alpha}$$

That is, for a given airspeed there is a naturally occurring relationship between flight path control sensitivity and flight path response. Hence, the main determining factor for flight path control during flare for powered-lift airplanes should be either n_{z_α} or $Z_w^\dagger$.

It is not clear which of the two factors is more universal or which should be directly limited. Both likely have limits. Under certain conditions the sensitivity limit can be more easily reached, while for others the response lag will be critical. A more quantitative treatment of this is given in the first finding soon to be discussed.

Meantime, let us compare the condition of flaring using throttle to the use of attitude and discuss some of the features. The important aspects of using throttle to flare can be shown using an analytical model of the flare maneuver similar to that shown above.

One might hypothesize that throttle is used in the same manner as attitude, i.e., an approximately linear feedback of altitude. Based on simulator data this seems to be a valid model of the flare maneuver, at least after the pilot has undergone sufficient familiarization and when the aircraft itself has adequate vertical path control potential. Figure 6-4 shows some actual simulator landings where throttle was used to flare.

An analytically developed example is shown in Figure 6-5. This is a direct comparison of flare with throttle to flare with attitude. The same basic powered-lift airframe is considered. It should be again noted that in this case the effective thrust angle is vertical and engine lag is zero.

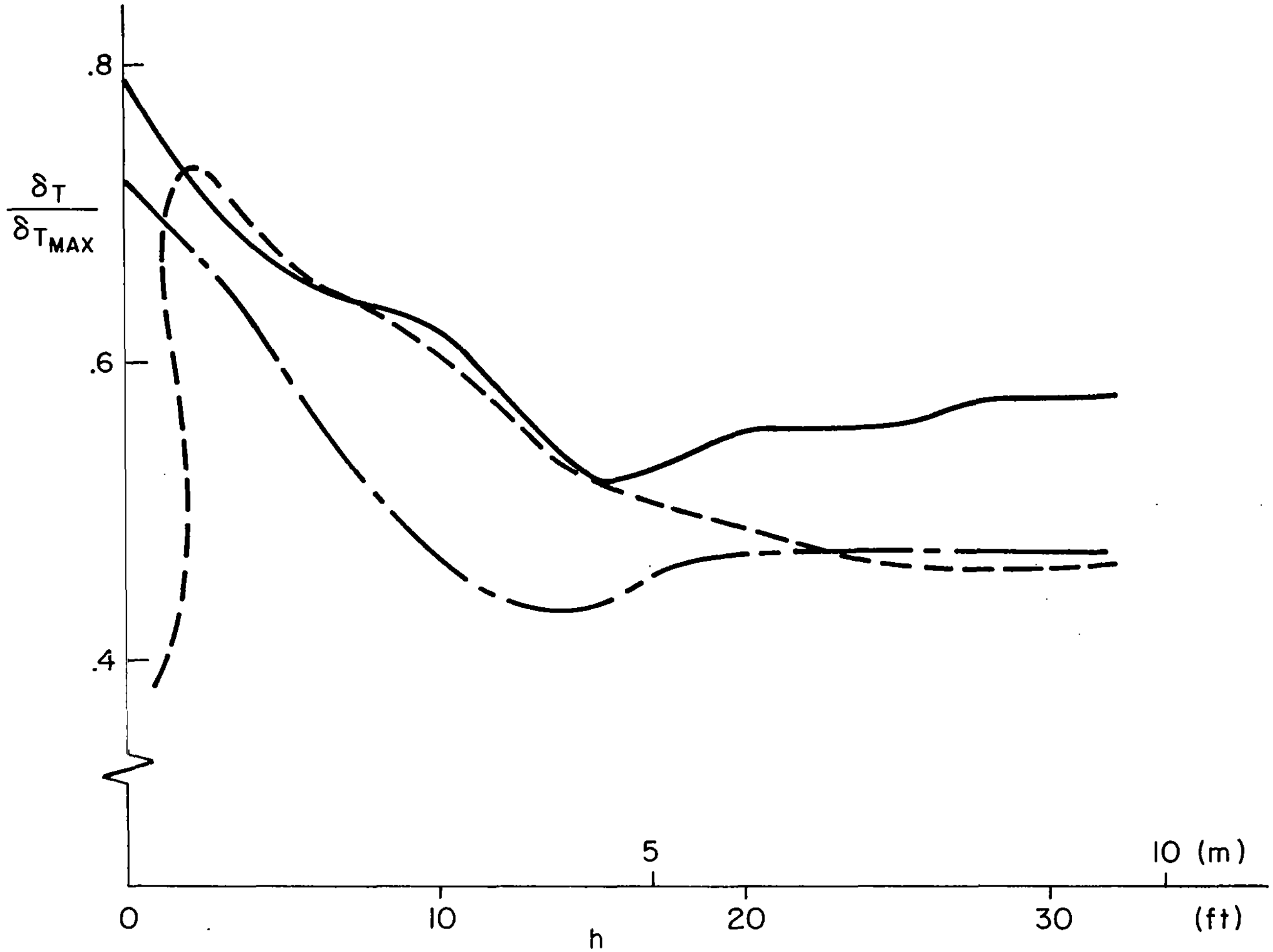

Figure 6-4: Flares Using Throttle

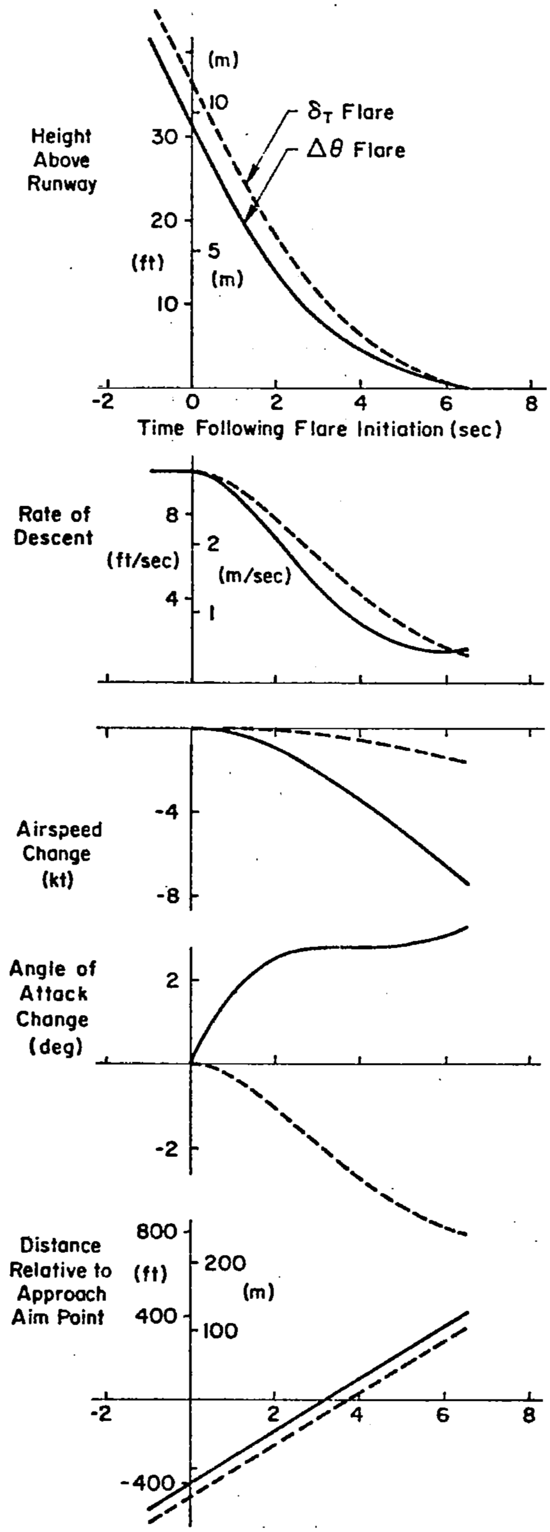

Figure 6-5: Comparison of Flare Technique

The flare height and throttle increment used to describe the flare are somewhat arbitrary, but are meant to reflect typical cases observed on the simulator. Variation could be expected depending on pilot and specific vehicle characteristics. The considerations in this example were to use a reasonably common flare height, abruptness of flare, and touchdown sink rate.

Note that the flight path change itself in terms of altitude and rate of descent is generally comparable using both techniques. This simply indicates the potential for the same kind of maneuver with either control. According to this example there is a tendency for flare using throttle to produce a float ($\dot{h}$ continues toward zero) compared to positive settling using attitude. This general effect is visible to the pilot.

The most notable characteristics are the comparative effects on airspeed and angle of attack. Flare with throttle tends to affect airspeed less. The exact effect depends mainly on effective thrust angle as noted in Section 5. While the airspeed change tends to be small, the angle of attack change is large but in a favorable sense with respect to safety margins. The latter is simply a result of holding attitude while increasing flight path angle.

There is a slight difference in the horizontal distance, but this is only due to the small difference in flare height in these examples.

The important aspects of using throttle to flare as shown above are net increases in speed and angle of attack safety margins during the flare. This contrasts sharply with the clear decrease in margins when flaring with pitch attitude.

Now consider the vertical path control aspects of flaring with throttle. The vertical path dynamics are really the same as described for the approach phase (Section 5). The distinguishing factor from pitch attitude control is the absence of a large flight path decay tendency. Presuming an adequate limitation on this for the approach phase, then the main features to be addressed in flare are simply vertical path response time and control power. It is natural to expect that the need for increased precision in the flare will require a correspondingly faster response time and increased short term control power.

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FINDING:

Reasonable limits on n_{z_α} and heave damping, $Z_w^\dagger$, for flaring with pitch attitude have been determined, but application of the limits to approach speeds significantly different from those simulated is questionable.

DISCUSSION:

The main unsolved problem in this respect is determining whether the limiting factor is control sensitivity, n_{z_α}, or control response, $Z_w^\dagger$. Arguments can be made to support either or both. The following describes some of the data obtained from this program as well as pertinent information from other sources.

First, consider a selected portion of data obtained from this program. Figure 6-6 shows comparison plots of pilot rating versus n_{z_α} and $Z_w^\dagger$. These data are relatively clear of other factors because of the following:

- They represent the ratings of a single pilot
- None of the configurations are excessively frontside or backside
- The standard turbulence level was used in all cases
- The same method of evaluation was used
- A significant speed range was spanned (55 kt to 75 kt).

For small values of either n_{z_α} or $Z_w^\dagger$ a significant degradation in pilot rating occurs. The parameter, n_{z_α}, however, appears to provide better correlation. The importance of n_{z_α} is further supported by pilot comments which specifically mention excessive attitude change. In some extreme cases there was a loss of view of the runway near touchdown, but these cases are not included in the plot because this was not a control related problem.

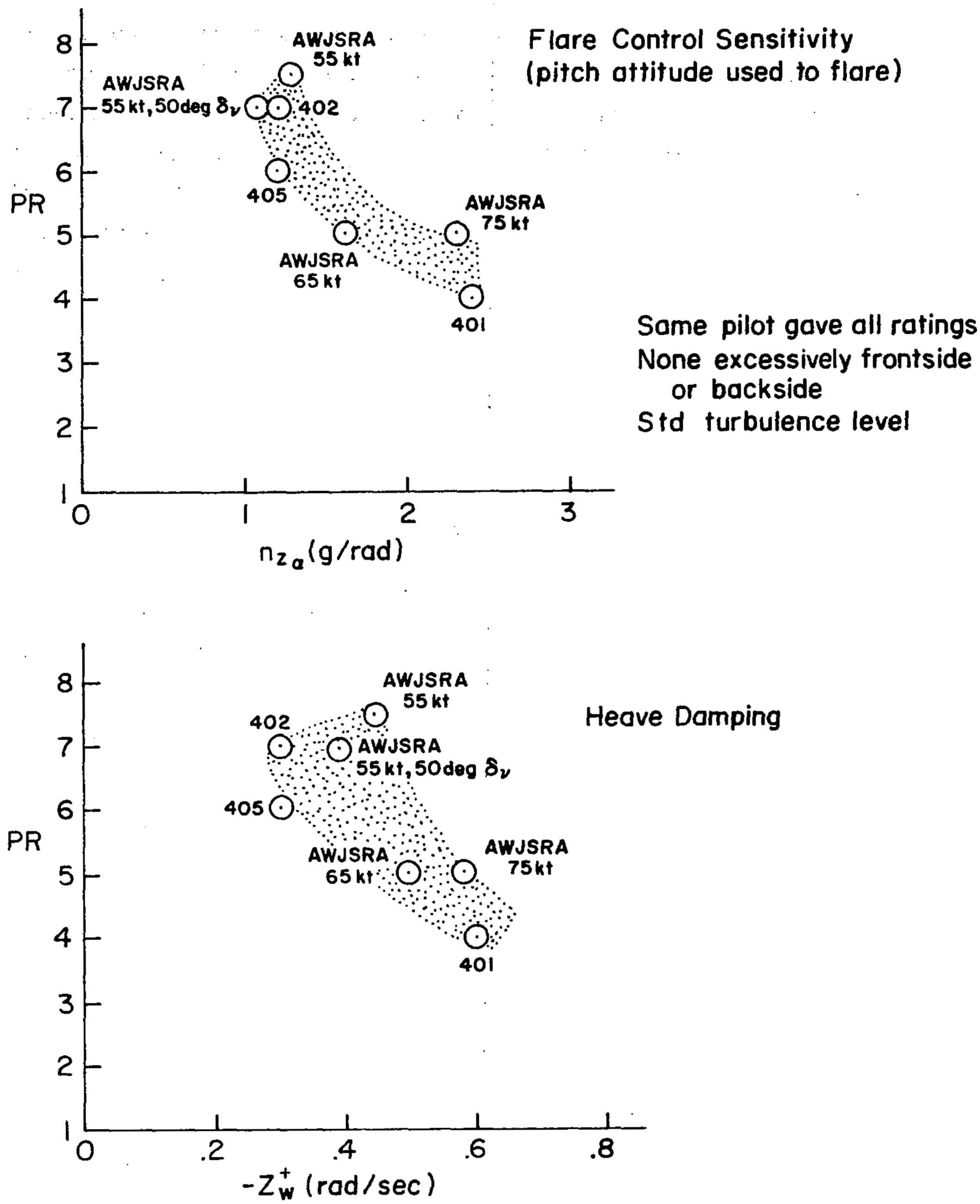

Figure 6-6: Attitude Flare Parameter Comparison

Nevertheless, it is not possible to rule out heave damping or path response as the critical factor. The difference in data dispersion in the two plots is not that great. Furthermore, the same levels of heave damping in a much faster aircraft (hence larger n_{z_α}) give similar pilot ratings. Figure 6-7 shows the previously plotted data compared to pilot rating trends for a simulated space shuttle vehicle landing at about 180 kt (Reference 35). On the basis of this, one is tempted to assume heave damping is the more critical parameter.

While we are still left not knowing what was really deficient in the cases studied, an opinion is nevertheless ventured here. It is felt that the degradation of pilot opinion shown in Figure 6-6 is tied to both low sensitivity and low heave damping. The pilot complaints of excessive attitude excursion must be considered as well as the space shuttle data. It seems reasonable to consider tentative limits of $n_{z_\alpha} \doteq 1.6$ g/rad and $-Z_w^{\dagger} \doteq 0.45$ rad/sec. Similar limits are proposed in Reference 10 from another simulator investigation of path control requirements for powered-lift aircraft.

The above limits on n_{z_α} and $Z_w^{\dagger}$ are exactly equivalent at a speed of 68 kt. For speeds close to this it is not really important which parameter is more critical. The question does become important for speeds which are significantly higher or lower. Unfortunately, there is insufficient data to determine which parameter would be more critical in either case. Therefore the limits should be considered valid only for approach speeds in the range of roughly 60 - 80 kt.

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FINDING:

The suggested short term response criterion for flare with throttle is a rise time to $\frac{1}{2}$ amplitude of approximately 2 sec.

DISCUSSION:

This criterion is based on a large number of simulator cases in which the results were reasonably consistent. The data considered for this flare

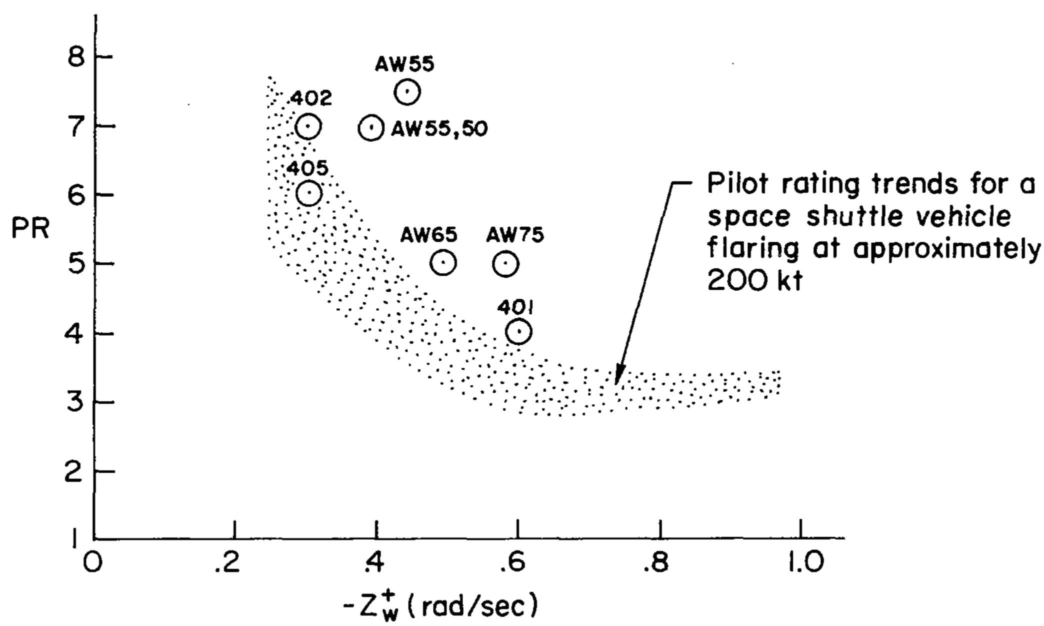

Figure 6-7: Pilot Rating Versus Heave Damping
Powered Lift Compared to Space Shuttle Vehicle

response criterion are tabulated in Table 6-1 and plotted in Figure 6-8. The trend of averaged pilot ratings versus rise time is well defined. The main question is what pilot rating to use as a cut-off in determining maximum rise time. This was done primarily by relying on the results of the STOL-X simulation. It was considered that this, the last simulation of this program, produced the most consistent and well-defined results. There was a clear concensus that a 2 sec rise time in $\Delta\gamma$ produced an adequate level of response for the power-to-flare maneuver in the 1.4 m/s (4.5 ft/sec) turbulence level used. This criterion should apply not only to throttle, but to any choice of primary control for flare.

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FINDING:

If the nominal landing is characterized as non-flared then the minimum short term response rise time need be only that for the approach segment.

DISCUSSION:

A special experiment run during the STOL-X simulation showed that if a so-called carrier landing were performed, the short term response requirement was less stringent than for a flared landing. In fact, the minimum level of short term response needed to fly the approach was adequate all the way to touchdown.

This seems reasonable since the non-flared landing is no more than an approach continued to the ground with no real change in technique or visual guidance information. The flared landing, on the other hand, involves a departure from the approach mode of operation and some loosely defined increase in vertical path precision.

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FINDING:

A calm air demonstration of the landing flight phase using a set of the appropriate abuses can give an indication of landing characteristics in turbulence.

TABLE 6-1

DATA CONSIDERED FOR FLARE RESPONSE RISE TIME

CASE	$t_{1/2}$ (sec)	RATINGS	AVERAGE	REMARKS
AWJSRA, DLC	1.1	4	4	
405	1.6	3	3	
STOL-X, DLC	1.7	Acceptable (1 pilot)	Acceptable	
STOL-X, $\Delta\ \tau_E = -1$	2.0	Acceptable (2)	Acceptable	
STOL-X, DLC	2.0	Acceptable (1), Unacceptable (1)	Marginal	
BR 941, 65 T in $\tau_E = 0.5$	2.0	Acceptable (2)	Acceptable	
1250	2.3	3	3	
STOL-X, $\Delta\tau_E = -.5$	2.5	Marginal (1)	Marginal	
402	2.6	4	4	
1210	2.8	6, 6	6	
STOL-X	3.0	6, 6, $6\frac{1}{2}$ to 7	6.2	
BR 941, 65 T in	3.2	Acceptable (1), Unacceptable (1)	Marginal	
1250, $\Delta\tau_E = 2.5$	3.7	7	7	
STOL-X, $\Delta\tau_E = +1$	4.0	6, 7, $7\frac{1}{2}$	6.8	
BSL-1 (NAV)	4.1	7, $6\frac{1}{2}$ to 10	~7	Involved initial use of pitch then modulation with power, slight effect of SAS gain
401	4.3		Unacceptable	Written comments indicate unacceptability
1210, $\Delta\tau_E = 2.5$	4.6	6, 8	7	

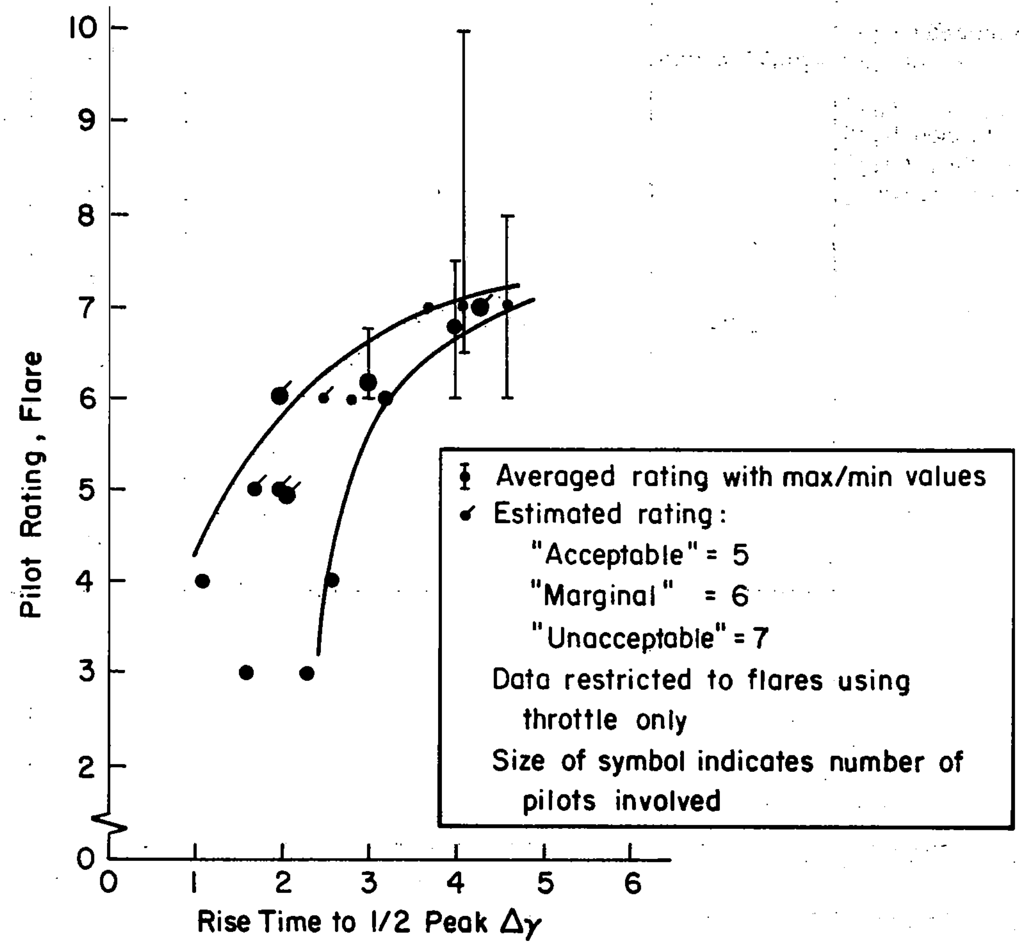

Figure 6-8: Pilot Opinion of Flare Versus Flight Path Response Rise Time
(From Table 6-1)

DISCUSSION:

This is a key finding of this program, which points the way to determination of airworthiness relative to the landing without requiring direct demonstration in a given level of wind or turbulence. It was recognized that landing under adverse atmospheric conditions was probably the most critical part of powered-lift operations. Ideally one would like to demonstrate landings in the worst expected atmospheric conditions. Such a demonstration, however, would be impractical from the flight test standpoint. It would be considerably more difficult than demonstrating in a given mean wind condition since one would want a variety of RMS gusts, shears, etc. Therefore, we investigated what might be done under the calm air conditions to somehow simulate problems encountered in significant levels of turbulence.

The basic idea behind the calm air demonstration is that one main effect of turbulence is to increase the dispersion of conditions at flare initiation, such as sink rate, airspeed, and control settings. It is presumed that if the pilot/vehicle in calm air is capable of a safe landing from off-nominal initial conditions and has an adequate level of short term response, then it should be able to handle a given level of atmospheric turbulence. In general, then, we wanted to look at a variety of abuse conditions that would be the likely result of turbulence and show that the pilot could successfully flare without a major change in flare technique. Also, this would presumably look after aspects of flare control that are difficult to measure directly. This could include such things as vertical path control power or cross coupling which may be dependent upon ground effect.

The idea of a calm air flare demonstration was a result of the BR 941 and AWJSRA simulations. It was developed conceptually in the first working group meeting, then explored in the STOL-X simulation. Feasibility was examined for a variety of flare techniques including full flare with power, full flare with attitude, and no flare (i.e., continuation of the nominal approach flight path and piloting technique to touchdown). Minor adjustments in the baseline STOL-X configuration were made to accommodate each of these flare techniques. Each of the cases will be discussed in the following paragraphs.

Flare using pitch attitude as the primary control is the most interesting case in that all abuses are relatively important. A high sink rate abuse indicates the basic requirement for vertical path control power. In addition, it shows the ability to counter low power and high sink rate at flare initiation due to a late correction or an off-nominal power setting. A fast abuse of flight reference can reveal a variation in flight dynamics which may lead to over-rotation and resulting long touchdown. A slow abuse of flight reference can result in a variation of flight dynamics which may reduce flight path sensitivity and result in an inadequate break in sink rate and inadequate safety margins. The most serious abuse is likely to be an inadvertent throttle cut at flare initiation which produces a serious loss of vertical path control power available with pitch attitude. This is considered an important abuse becuase it represents an action that is frequently taken in conventional aircraft in which there is no correspondingly adverse effect.

A flare in which throttle is the primary control is, in general, not as critical relative to abuses as when flaring with attitude. A fast/slow abuse does not produce the same results because the flare control is not directly affected by speed, i.e., Z_{δ_T} tends to remain constant while n_{z_α} does not. A secondary control abuse, pitch attitude, is not likely in the same sense that a throttle abuse is. Thus a nose-down attitude abuse corresponding to a throttle cut for the opposite technique seems unnecessary. The most significant abuse for throttle-to-flare is an off-nominal sink rate. Demonstration of such an abuse would directly check for vertical path control power.

The final flare technique case considered was use of throttle as the primary control with no break in sink rate, i.e., the approach is continued all the way to touchdown. This is really just an extension of the previous case. Therefore, the speed and secondary control abuses would be expected to have little importance, however, a sink rate abuse would remain important in order to assure flight path control power in the presence of any ground effect.

The results of calm air flare demonstrations for the STOL-X simulation are summarized in Table 6-2. The numerical flight path and flight reference abuses shown were considered by the pilot to be representative of conditions encountered in the simulated 10% turbulence conditions. Note that for an attitude flare the pilot selected the fast abuse larger than the slow abuse. He did not try to make this distinction in the power-to-flare case, though, because speed abuses were not considered as important. In the no-flare case a steeper sink rate abuse was suggested by the pilot and a distinction was made between the size of fast and slow flight reference abuses.

Based on the limited examination during the STOL-X simulation, the calm air demonstration of landings appears to be a feasible way of determining airworthiness. The concept, however, needs more thorough investigation to determine numerical abuses which would apply to a wider range of airplane configurations.

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FINDING:

An approach-type short term/long term vertical path control power requirement is unsatisfactory for the flare due to complicating factors such as ground effect and flare technique.

DISCUSSION:

Vertical path control power is an important requirement in the flare maneuver but it defies the simple step input criterion suggested for the approach flight phase. The long term vertical path control power requirement does not really apply to the flare maneuver because of its relatively short duration. Short term control power is, however, important but it depends upon the degree of abruptness of the flare maneuver and the amount of positive or negative ground effect present in a specific vehicle. Therefore, the control power requirement is very much design dependent.

TABLE 6-2

SUMMARY OF APPROPRIATE CALM AIR DEMONSTRATION ABUSES

STOL-X

FLARE TECHNIQUE	FLIGHT PATH ABUSE	FLIGHT REFERENCE ABUSE	REMARKS
Pitch Attitude Primary Control	2 deg steeper (than nominal)	2 units* slower (3 kt) 4 units faster (7 kt)	Secondary control abuse consisting of idle power during flare was most critical condition. Flight reference abuses critical because of strong speed influence on attitude flare dynamics.
Throttle Primary Control (3 1/2 deg attitude change required to clear nosewheel)	2 deg steeper	4 units slower (7 kt) 4 units faster (7 kt)	It was not indicated if these flight reference abuses were representative of turbulence conditions. Not critical, however, since speed does not have a big effect on throttle flare dynamics.
No-Flare, Throttle remains primary control to touchdown	3 deg steeper	2 units slower (3 kt) 4 units faster (7 kt)	Fast flight reference abuse critical for attitude re nosewheel, not for speed per se. Increased flight path abuse probably an indication of a less constrained sink rate flare window.

* Units refers to the flight reference used with this specific model. Numbers in parentheses indicate the corresponding steady state airspeed excursions.

The most direct way of guaranteeing an adequate level of vertical path control power is through an appropriate calm air flare demonstration. In particular, a demonstration involving a sink rate abuse would produce a direct measure of the same things looked after in the approach control power criteria. Ground effect or flare technique peculiarities would be directly taken care of.

SECTION 7

LATERAL-DIRECTIONAL STABILITY AND CONTROL; APPROACH AND LANDING

Lateral-directional stability and control was not a subject of formal investigation in this program. The decision was made to concentrate on the longitudinal axis because powered-lift aircraft have fundamentally different longitudinal characteristics and airworthiness problems than do conventional aircraft. Inherent lateral-directional differences between powered-lift and conventional aircraft are a matter of degree, not fundamental character. The basic piloting techniques are the same and the same handling qualities criteria should apply.

The above is not meant to imply that powered-lift aircraft have no lateral-directional problems. On the contrary, they seem to generally have worse characteristics than conventional aircraft. The following problems appear to be common in many powered-lift designs:

- Poor turn coordination
- Relatively rapid spiral divergence
- Low roll damping
- Low frequency and damping of the dutch roll mode.

As a result, stability augmentation may be more of a necessity for powered-lift aircraft. There is, however, a large background of pertinent research and many handling qualities criteria have been proposed, e.g., References 1, 2, 3, 4, 6, 7, 36, 37, and 38.

Since a formal investigation of lateral-directional characteristics was not conducted, there are no firm quantitative results to report. The objective of this section is to record a few qualitative observations which were made during the program.

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FINDING:

Turn coordination or heading control problems were the major complaints during the few simulation runs made without lateral-directional SAS.

DISCUSSION:

As a rule, the simulation experiments of this program were run with a lateral-directional SAS for the specific purpose of concentrating on longitudinal aspects. The instances in which no lateral-directional SAS was used were:

- All runs during the first BR 941 simulation (October/November 1972)
- One series of runs for the AWJSRA simulation
- Some runs with the Generic STOL model.

In these cases the pilots found it difficult to track the localizer because of problems in making precise heading corrections. This was due to large adverse yaw characteristics. Furthermore, it was difficult for the pilots to compensate for the adverse yaw by using the rudder to coordinate the turns.

The key factors in turn coordination and heading control are generally:

- Dynamic adverse/proverse yaw, $(N'_p - g/V)$
- Aileron adverse/proverse yaw, $(N'_{\delta a}/L'_{\delta a})$
- Dutch roll frequency and damping.

High directional stability (high dutch roll frequency) tends to reduce sideslip and thereby improve turn coordination. Unfortunately, the simulated aircraft had low dutch roll frequency and damping and were therefore more sensitive to the other two parameters.

The yaw directly due to aileron $(N'_{\delta a}/L'_{\delta a})$ was not a serious problem as it was relatively small (the ideal value is zero). The main culprit was dynamic adverse/proverse yaw, $(N'_p - g/V)$ which should be zero for turn coordination. The simulated aircraft had relatively large, negative values which resulted in adverse yaw (aircraft yaws out of the turn). The main reason was the slow speed which increased the g/V term. The values of N'_p were also relatively large and negative.

Because the adverse yaw was due to $(N'_p - g/V)$ rather than $(N'_{\delta_a}/L'_{\delta_a})$, turn coordination with the rudders was more difficult. One can compensate for a $(N'_{\delta_a}/L'_{\delta_a})$ effect by a simple crossfeed from wheel to rudder, i.e., the pilot need only apply rudder proportional to his wheel input. To compensate for an $(N'_p - g/V)$ effect requires a lagged crossfeed (lag equal to the roll mode). This is more difficult, if not impossible, for the pilot to do accurately. It should result in at least a high workload and probably poor tracking performance.

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FINDING:

Pilots initially had some difficulty adjusting to the higher turn rate/roll sensitivity resulting from the low approach speed.

DISCUSSION:

In a steady turn, $\dot{\psi} \doteq g/V \cdot \varphi$; thus, a low approach speed means a higher sensitivity of turn rate to roll. Some of the pilots more accustomed to higher speed aircraft found this adjustment troublesome. They had to regulate roll attitude more precisely and use small roll corrections. After sufficient training, though, the problem seemed to disappear.

While this may have been only a training problem it could also indicate the need for more precise roll control for slower aircraft. To our knowledge no handling qualities study has ever considered the effects of approach speed on roll control requirements.

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FINDING:

All of the powered-lift configurations considered in the course of this program had acceptable crosswind landing characteristics, at least in the modest crosswinds evaluated.

DISCUSSION:

It should be recalled that only three basic lateral-directional configurations were involved in this program —— the BR 941, the AWJSRA, and the Generic STOL (including STOL-X). This small sampling is really further reduced by the fact that a similar lateral-directional stability augmentation system was used in each case. To the extent that this SAS masked the basic aircraft characteristics, one could say that all lateral-directional configurations were really the same regardless of the simulator model.

One minor problem did arise during the development of the SAS. The turn coordination feature would fight the pilot during the decrab maneuver. The result was that excessive pedal forces were required to decrab. The problem was cured by a simple electrical feed-forward command from the pedals to the rudder to offset the turn coordination signal.

An important factor regarding the crosswind landings made during this program was the magnitude of the crosswinds themselves. Early in the program it was found that the visual display severely limited the magnitude of the crosswind which could be considered. The restricted lateral field-of-view of the visual display in combination with a low approach speed meant that only a modest crosswind component could be used without the pilot losing sight of the runway. For this reason, crosswind components of no more than 10 kt were evaluated. Because of the low crosswinds evaluated, the total impact of crosswind landings may not have been fully appreciated. This coupled with the weak variation in airplane control characteristics suggests cautiously interpreting the magnitude of the crosswind landing problem.

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SECTION 8

PROPULSION SYSTEM FAILURE; APPROACH AND LANDING

This section considers conditions associated with the failure of a propulsion system unit during the approach and landing flight phase. It contains not only the results obtained during this simulation program but also a substantial number of results from other similar investigations.

This section is subdivided into two main parts:

- The failure transient itself, including the establishment of a new trim condition
- The continued approach with a propulsion unit failed.

One particularly important task of this section is to describe those features involved in propulsion system failure which are likely to be inherent in powered-lift airplane designs. These features are then considered in describing and intepreting simulation results.

One likely characteristic of a powered-lift design is that the propulsion system is more complex than in conventional aircraft. For example, it could include ducting of hot or cold gases, movable nozzles, or propeller cross-shafting. When considering failures one must include failures of each of these elements as well as the engines themselves. For this reason we use the term, <u>propulsion system</u> failure, rather than just <u>engine</u> failure.

8.1 PROPULSION SYSTEM FAILURE TRANSIENTS

The following treats the transient condition immediately following a propulsion system failure and up through restoration of a reasonably steady state condition. We begin with a description of the transient condition and follow it with the simulation results and their implications on development of airworthiness criteria.

It is important to consider the effects of failure transients in powered-lift aircraft because the transients themselves are significantly different from those occurring in conventional aircraft. The most obvious difference between powered-lift and conventional aircraft is the loss of lift that occurs from the failure. This loss of lift results from the lost engine thrust which was actually generating a portion of the lift force supporting the aircraft. Figure 8-1 illustrates these and other major differences from conventional aircraft.

The first apparent motion resulting from a propulsion failure is a marked increase in sink rate which is simply the direct result of a decrease in powered lift. Also, with thrust acting primarily in the vertical direction there is little tendency for the aircraft to slow down as a conventional aircraft does following an engine failure. In fact, some powered-lift aircraft could tend to increase speed. Recalling the various flight dynamics parameters discussed in Section 5, one can relate the powered-lift loss directly to the powered-lift factor, η_p, and the initial tendency to change airspeed to the effective thrust angle, θ_T.

The failure of a propulsion system unit produces a set of lateral-directional upsetting moments which are also illustrated in Figure 8-1. For a powered-lift airplane in approach configuration, the lift on the wing supplied by the failed engine can be substantially less than the lift on the opposite wing. The net difference in lift produces a rolling moment and the drag difference produces a yawing moment. The yawing moment for a powered-lift airplane is much less than for a conventional airplane if the effective thrust angle is nearly vertical.

The pilot in the propulsion failure situation must first recognize the failure. Next he must cope with the motion transients described above and reattain a reasonably well trimmed flight condition which permits either (i) the successful continuation of the approach or (ii) initiation of a missed approach. The findings relating to this process are broken down in the following manner:

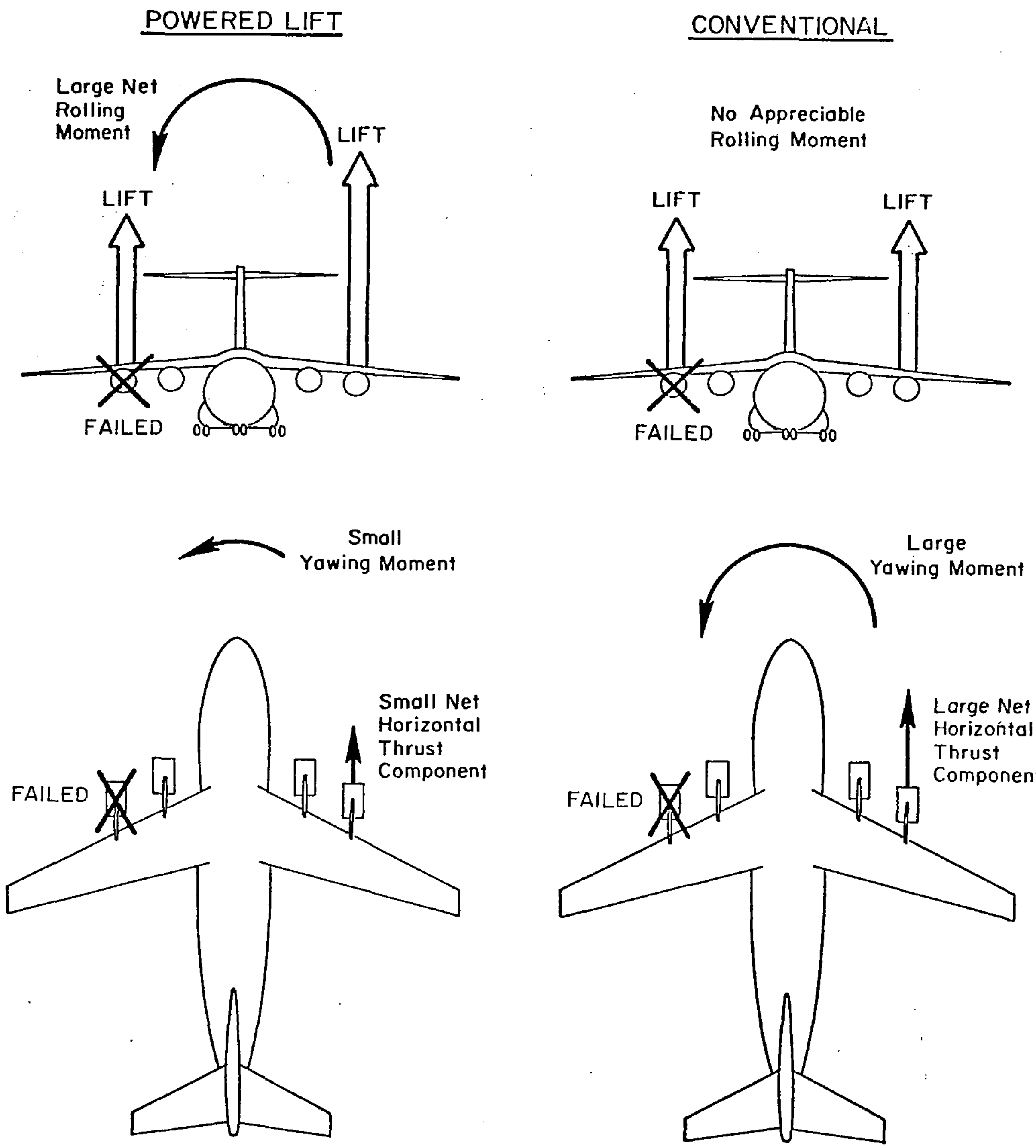

Figure 8-1: General Propulsion System Failure Effects on Forces and Moments

- Recognition of the propulsion system failure
- Piloting technique during the failure transient
- Lateral-directional control requirements
- Longitudinal control requirements.

The findings in each of the above areas are accompanied by a discussion of the problems encountered and their implications for regulatory standards.

8.1.1 RECOGNITION OF PROPULSION SYSTEM FAILURE

Delay in recognition of a propulsion system failure represents a time lag in dealing with a potentially hazardous situation. The following findings relate to failure recognition and reveal some of the aspects involved in operation of powered-lift aircraft.

In this program propulsion system failure was studied in conjunction with three simulator models: BR 941, AWJSRA, and STOL-X. These provided an interesting variety of characteristics but did not systematically cover the broad range possible for all powered-lift designs. In some tests failures were introduced randomly during a series of runs. In other tests there was a series of runs devoted solely to the failure problem. It was never possible, however, to realistically duplicate the element of surprise of a failure during normal operations.

In these simulation experiments a propulsion system failure was usually signaled by an audible tone in the cockpit. The warning was activated at the same time engine RPM started to decay. The tone provided an unmistakable cue that a failure had occurred, but it was necessary to wait until a motion cue became apparent or to check the engine instruments in order to determine which engine had failed. In some cases an audible tone was not used and the pilot had to rely solely on motion or instrument indications to detect and diagnose a failure.

The following findings reveal some of the important aspects related to recognition of a propulsion system failure.

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FINDING:

An artificial warning of propulsion system failure may be necessary for some powered-lift airplanes.

DISCUSSION:

Dependence upon motion cues or engine instruments to warn of propulsion failure does not appear really adequate. The following are some of the factors involved in a number of simulation experiments which show this and suggest the need for artificial failure cues.

The STOL-X simulation (Reference 14) embodied a number of factors which could be considered typical of powered-lift aircraft. Therefore we will begin the discussion with this example.

The initiation of an engine failure was indicated by a 15 sec duration audible tone in the cockpit. Perceivable motions following a failure built up slowly while the sink rate increased rapidly to about 5 m/s (1000 ft/min). It appeared that recognition of failures, at least in the STOL-X design, was more difficult than in conventional aircraft because of the time for perceivable motion to build.

It should be noted that testing of propulsion system failures in this particular experiment lacked the element of surprise. Since the pilot was briefed on the task, he was generally "spring loaded" awaiting the engine failure. Consequently, reaction times were probably shorter and the reactions which ensued were better than might be expected during actual operating conditions. The few failures in which pilots were not informed in advance were recoverable, but the failures did not occur at critical altitudes.

In a simulator experiment with an EBF design (Reference 39), it was found that the quickest reaction times under ideal conditions were on the order of 1.2 to 1.5 sec. The most readily detectable cue of engine failure to which the pilot could respond was the bank angle excursion induced by the roll asymmetry when the engine failed. Vertical acceleration cues from the simulator were not of sufficient magnitude to be detected. The increase in vertical velocity did not become apparent visually until a sizable sink rate had already built. Engine instruments were located on the center instrument panel and were not included in the pilots' continuous pattern. The lateral SAS limited the rolling and yawing excursions to about 6 deg and consequently limited their effectiveness as a cue to a failure. Reference 39 concluded that it is likely that artificial warning will be required.

Not all simulation experience has involved low levels of motion following a propulsion system failure. In Reference 40 it was found that a sudden failure in an engine produced a very noticeable roll and yaw for certain powered-lift designs. It appeared that the pilot would have little trouble in identifying an engine failure in those cases.

The use of cross ducting can produce motion cues that are somewhat confusing when an engine fails. In the simulation of the AWJSRA (Reference 12) it was noted that the aircraft rolled in a direction opposite to that expected (i.e., loss of a right engine produced a net loss of lift on the left wing because of cross ducting), yet the nose yawed to the right which was normal. The addition of thrust, in this case, only aggravated the peculiar combination of lateral-directional asymmetries.

Propulsion system failures in the BR 941 simulation (Reference 14) were difficult for the subject pilot to detect because of the lack of asymmetry due to propeller cross shafting. Also, while a failure of one engine did produce a 25% loss of power this resulted in only a 15% loss of net thrust. The governor changed propeller pitch to maintain propeller RPM which resulted in a net increase in propeller efficiency. Therefore, thrust loss was not as great as power loss. Aside from the audible warning, the only other warning of propulsion system failure in the simulated Breguet airplane was a relatively mild increase in sink rate.

In conventional transport aircraft, the pilot generally experiences substantial cockpit side accelerations due to the asymmetric yawing moment produced by an engine failure. In powered-lift aircraft, a rolling moment may be produced following a failure. Since the pilot is located close to the roll axis of rotation, the cockpit accelerations produced by the asymmetric rolling moment are low. Thus, the acceleration cues provided to the pilot of a powered-lift aircraft are not, in general, as effective as those in conventional aircraft.

In summary, the elapsed time between a propulsion system failure and the pilots' identification of that failure will vary depending on the particular characteristics of that aircraft. Generally, the reaction times for failure recognition will be longer for powered-lift aircraft than for conventional aircraft. Therefore it may be necessary to require some type of artificial failure warning system. At the same time it should be noted that any real failure warning system will have some inherent delay although it might be insignificant.

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FINDING:

An audible propulsion system failure warning is only partially effective.

DISCUSSION:

Most of the experiments conducted in this program employed the audible warning described previously. While it was clearly effective in alerting the pilot of a propulsion system failure, certain qualifications should be noted. First, it was an ideal device in that there was no time delay between initiation of the failure and the warning. Next, it warned only of a failure, it did not tell the pilot which unit failed nor did it indicate what kind of control manipulation was required. For this sort of information he had to rely on engine instruments and detection of motion through feel, vision outside, or cockpit instruments.

One simulation experiment described in Reference 41 revealed that warning lights were also beneficial in warning of engine failure. The relative value of lights and aural devices is a human factors problem which is outside the scope of this program. An important consideration in this regard is the requirement for other warnings, such as SAS failures or overspeed.

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8.1.2 PILOTING TECHNIQUE DURING THE FAILURE TRANSIENT

In dealing with a propulsion system failure, it may be necessary for the pilot to quickly manipulate several controls in an effort to compensate for the upsetting forces and moments. It is highly likely that powered-lift, just as conventional aircraft, will require that the engine power control be advanced following a failure. It is also likely that substantial roll control will be involved and, for some aircraft, it may be necessary to change the pitch attitude. Aircraft such as the AWJSRA require adjustment of the nozzle vector control. Finally, in some cases application of directional control may be required. In consideration of the number of different controls which may be required to counteract the failure, there are a large number of possible sequences of control application. This falls into the category of defining a piloting technique.

The following are a number of findings based on several simulation experiments involving propulsion system failure. For this reason, the findings are somewhat unique to the configurations observed. These are of value from a general point of view, however, because they reveal problems associated with a variety of possible piloting procedures or technique.

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FINDING:

The sequence of corrective actions is a key aspect of piloting technique following propulsion system failure.

DISCUSSION:

The choice of control activation sequence can depend on a number of things. One could choose to pitch over in order to accelerate or regain safety margins, or the choice could be to immediately add power if arrestment of excessive sink rate were important. Another possibility could be the immediate application of roll control to counteract a serious lateral asymmetry. Still another choice of initial reaction could be the selection of a configuration change which takes a long time to effect. For example, if it were important to eventually raise flaps but the flap rate were slow, then the first move might best be initiating the flap change.

During the STOL-X simulation (Reference 14) the choice of initial control application was limited primarily to two possibilities. The first was to immediately pitch down in order to accelerate, and the second choice was to initially add power. In the former case the technique consisted of pitching over to regain the flight reference (essentially a decrease in angle of attack), dealing with rolling moment, and finally adding power to regain flight path. The acceleration to a higher speed took a substantial period of time. By pitching over first, the elapsed time to achieve the higher speed was kept to a minimum. In this case, the higher speed was desirable since it provided increased lateral control and greater flight path capability. While this technique was satisfactory at higher altitudes, there were problems with failures at low altitude which will be discussed shortly.

Some pilots in the STOL-X experiment favored the technique of first applying power, then countering roll motion, and finally, pitching over. This sequence allowed the pilot to track the glide slope more closely because it more rapidly countered the substantial loss of lift from the engine failure. The problem associated with this technique was that without attaining the desired flight reference, i.e., pitching over, the aircraft could barely sustain its nominal approach flight path angle. (This aspect will be developed in the next subsection.)

For the powered-lift design evaluated in Reference 42, a different control sequence was used. The most satisfactory sequence of control application was found to be (1) regain bank angle control, (2) correct pitch attitude and airspeed, (3) initiate heading correction, (4) initiate throttle changes, (5) complete the heading correction, and (6) complete the throttle adjustment. This resulted in 2 to 4 sec of other control manipulations before the pilot was ready to deal with the throttle control. When simultaneous control inputs were attempted the recovery time was actually increased due to a tendency toward PIO.

In another simulation (Reference 41) the lateral-directional SAS was found to be an aid to dealing with a propulsion system transient. The simulated airplane was an EBF and an engine failure caused a large rolling moment. Since the SAS would oppose the roll disturbance, the pilot was able to begin advancing the throttle immediately after detecting a failure. He did not have to be concerned with countering the rolling moment resulting first from the thrust loss and next from the addition of thrust.

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FINDING:

The proper sequence of control application following a propulsion system failure is likely to have a strong dependence on the altitude of the failure.

DISCUSSION:

In the various cases considered, the sequence of application of controls did not seem to be particularly critical at high altitudes; but it was usually highly critical at low altitudes. The problem was one of the time between the failure and landing. The pilot needed time to stop the initial transients and return to an appropriate flight path. The AWJSRA and STOL-X are two cases which reveal a number of interesting features related to the altitude of the failure.

The AWJSRA simulator model employed an additional control (the nozzle angle) which complicated the problem of controlling the vehicle following a failure. For engine failures at a reasonably high altitude, say well above 60 m (200 ft), one technique used was to abruptly increase the pitch attitude, apply thrust, and rotate nozzles from their nominal angle of 75 deg to 40 deg. This increased airspeed about 10 kt to the desired OPUI (one propulsion unit inoperative) approach speed and minimized flight path losses. The reconfigured aircraft was easy to fly on the approach, but it was difficult to land. Touchdowns were generally long and reducing throttle to stop the float only resulted in hard landings.

If the failure occurred below about 60 m (200 ft) the subject pilot found there was insufficient time for any change in configuration. Reasonable landings were possible by avoiding the temptation to make a large power increase and accepting a landing short of the touchdown zone. A large thrust increase excited lateral-directional problems and generally resulted in an unavoidable lateral drift. There appeared to be a gray area for a successful landing following failures between 30 and 60 m (100 and 200 ft). In that region, correct pilot actions were most critical.

As mentioned earlier, the normal OPUI recovery technique for the STOL-X airplane was to pitch nose-down to accelerate to the desired OPUI airspeed. At lower altitudes, however, there were conflicting requirements. The requirement that the pitch attitude be decreased to maintain flight reference resulted in a loss of lift until speed increased. This push-over maneuver temporarily increased the sink rate and, at low altitudes, could result in a higher sink rate at flare initiation. In addition, there was the conflicting requirement of raising the pitch attitude to level to avoid landing nose wheel first during the flare with power.

On the nominal 6 deg glide slope with a failure at 60 m (200 ft) the pilot had less than 15 sec to push over, gain speed, and pull the nose back to level for the landing. The ability of the pilot to reduce his sink rate with a propulsion unit inoperative was directly related to his airspeed at the time of flare and also whether or not the lateral-directional axis had been stabilized.

In Reference 42 it was found that the pilots would reverse the sequence of their normal OPUI technique when failures occurred very near to touchdown. References 39 and 42 also found that low altitude failures resulted in very hard landings primarily because the pilot and/or engines did not have time to respond.

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FINDING:

Instinct is an important factor in piloting technique connected with propulsion system failure.

DISCUSSION:

An illustration of the role instinct played in reacting to an engine failure transient occurred in the case of the STOL-X simulation study. Opinions of the subject pilots were varied. While it was agreed that, at any altitude, the application of thrust following failure was natural, there was disagreement regarding the act of pitching down to accelerate to a more adequate airspeed. One opinion was that pitching down at higher altitudes was, in fact, an instinctive maneuver following an engine failure but that at altitudes below 60 m (200 ft) it was a most unnatural action. Another opinion was that it was simply never natural to pitch down and such an action would have to be developed through a training process.

It should be noted that the requirement for a pitch down maneuver following a propulsion system failure was somewhat peculiar to the STOL-X case, but that any possible requirement to perform such a maneuver which is not instinctive should be given careful consideration for two reasons. First, there is greater chance for pilot error arising from an incorrect action. Second, the time required for a pilot to perform the correct action is likely to be longer if it is not instinctive.

There is another complication which could not be realistically considered in the simulation programs. For a propulsion system failure in a powered-lift aircraft, the transient and appropriate recovery procedure may be entirely different for the approach than for all other configurations. The differences would be due to the different amounts of powered lift being used. In the takeoff and cruise configurations the aircraft would probably behave like any conventional aircraft. It could be difficult to develop the correct instinctive pilot reactions if they were to apply only to the approach configuration.

The problem becomes even worse if one hypothesizes aircraft operations which include alternating powered-lift and conventional landings. The crew might make a non-powered-lift landing (for increased payload) at a conventional airport. The next stop could be a powered-lift landing at a STOL-port. If the proper reactions to a failure during either of those landings are substantially different, the probability of an incorrect or delayed pilot reaction is increased.

8.1.3 LATERAL-DIRECTIONAL CONTROL REQUIREMENTS

This subsection addresses possible lateral-directional control requirements for dealing with the propulsion system failure transient. This is done by describing some of the control problems encountered during this and other programs. As before, we rely on use of specific cases viewed during simulation experiments to develop a more general overview.

It should be noted that many of the control problems related to the transient condition carry over to the steady state condition. These are further discussed in Subsection 8.2.

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FINDING:

Roll control during the propulsion system failure transient appears to be the dominant lateral-directional problem for powered-lift aircraft.

DISCUSSION:

In the STOL-X simulation (Reference 14) and in various investigations connected with the STAI (STOL Tactical Aircraft Investigation) program (References 42 and 43), roll was the primary axis concerning the pilot immediately following an engine failure. This contrasts with conventional aircraft for which the yaw axis is the main concern.

The degree of dominance of roll control problems is, of course, configuration dependent. As described in the beginning of this section, the influential factors are proportion of powered lift, effective thrust angle, and the effective lateral position of the net loss in powered lift (the asymmetry effect).

Since a rolling moment appears to be a major characteristic of the propulsion system failure transient, airworthiness standards for powered-lift aircraft should specifically address the need for reasonably low lateral control forces, rapid and easy to use means of lateral trim, and possibly an indication of the amount of correction required. These standards should also consider the use of automatic power and roll compensation systems such as considered in Reference 41.

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FINDING:

High lateral control forces required to counter the propulsion system failure transient can be a particularly serious problem.

DISCUSSION:

The high wheel forces required to deal with the lateral transient and the problems associated with trimming out those forces were troublesome to a number of the subject pilots in the STOL-X evaluation (Reference 14). Immediately following propulsion system failure and prior to increasing airspeed or developing sideslip in a helpful sense, the wheel force required to counter the upsetting rolling moment was 105 N (23.5 lb). In this particular case, the high wheel forces were a result of the lateral SAS saturating and disabling the command augmentation loop. Had the SAS not saturated, only 62 N (14 lb) would have been required. It should be noted that after the STOL-X was accelerated to 72 kt (its nominal operating point, OPUI), the steady state wheel force was reduced to 49 N (11 lb).

Some of the comments made by subject pilots with regard to the STOL-X case include:

- The wheel forces involved were hard to hold
- It was difficult to retrim while making small corrections against the large forces
- It was difficult to trim two axes (pitch and roll) simultaneously as was desired in this case.

One technique exercised was to simply apply a large amount of lateral trim in an open loop manner at the time of failure recognition.

The relatively slow rate of wheel trim added to the difficulties involved in the STOL-X simulator model. The wheel deflection necessary to neutralize rolling moment immediately following propulsion system failure was 44 deg, and the wheel trim rate was 6.5 deg/sec. It therefore required nearly 7 sec to completely trim out wheel forces. This was considered excessive.

One pilot was making what amounted to open-loop trim corrections, but felt that his open-loop trim technique might not be a feasible procedure for a pilot who had not recently practiced OPUI approaches.

In a number of other simulations in which the propulsion system failure transient was explored, high lateral control forces were not involved (References 41 and 42). Less lateral control was needed to cancel the upsetting roll moments. The STOL-X simulation simply revealed some of the problems that can emerge. This applies to other control problems cited in the following pages.

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FINDING:

In powered-lift aircraft, there can be a tendency toward incorrect use of rudder following a propulsion system failure.

DISCUSSION:

Recall from previous discussions that yawing moments following a propulsion system failure can act in either direction and with varying intensity depending upon the effective thrust angle and the degree of powered-lift asymmetry due to an engine failure. Therefore it is not possible to generalize control tendencies nearly so well as in the roll axis.

In the STOL-X simulation a common problem, particularly during the learning process, was the addition of too much rudder following an engine failure. As a result the pilots were causing lateral flight path problems through the generation of excessive sideslip. In many cases a lateral path divergence resulted which the pilot was not able to sort out. This will be discussed further in Section 8.2.

The results reported in Reference 42 indicate that for the EBF configuration evaluated there, the subject pilots felt the rudder control power was inadequate to deal with an engine failure. It was not possible to determine whether there were also problems with excessive sideslip as in the case of STOL-X. In Reference 40 there are specific indications that rudder requirements can differ significantly among powered-lift concepts. While the specific IBF (internally blown flap) and MF/VT (mechanical flap/vectored thrust) designs were not rudder control power limited, the EBF design was.

The AWJSRA simulation (Reference 12) provided an example of considerable rudder control being required to overcome a secondary transient as opposed to the primary failure transient. Immediately following the engine failure relatively little yaw asymmetry existed. It was necessary in this aircraft, though, to immediately vector the nozzles to a more horizontal angle. This vectoring resulted in an increased yaw asymmetry and required application of a significant amount of rudder control. It was clear that, in this case, an increase in thrust would further aggravate control of the yaw axis.

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FINDING:

Specific configurations of lateral-directional SCAS can create unanticipated side effects during a propulsion system failure transient.

DISCUSSION:

The STOL-X simulation revealed some problems which were unique to the airplane/SCAS configuration considered. This unique set of circumstances is cited here to signal the need for a cautious attitude and awareness of the possibility for such effects in other designs.

The directional SCAS utilized in the STOL-X simulation model included a wheel-to-rudder crossfeed which was intended to assist the pilot in performing coordinated turns. Unexpectedly, this crossfeed aided the pilot in dealing with the transient condition. The wheel deflection required to counter the roll transient resulted in a large enough rudder command via the wheel-to-rudder crossfeed to neutralize the yawing moment. Two subject pilots found that they achieved better success in handling the transient situation by avoiding any rudder pedal input. However, a different mechanization of powered lift might have required opposite rudder to counter the failure yawing moment. The SCAS crossfeed would then make it more difficult to overcome the failure transient. The same SCAS also introduced lateral-directional control problems in the longer term. These are described in Section 8.2.

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8.1.4 LONGITUDINAL CONTROL REQUIREMENTS

In the longitudinal plane, the two main control functions are regulation of flight path and flight reference. The aspects of pitch attitude control are adequately described in Sections 5 and 6 since the impact of propulsion system failure on pitch attitude control is not considered significant.

As described in the beginning of Section 8.1, for airplanes having a large powered-lift effect, a propulsion unit failure has a strong and immediate effect on flight path. Failures call for prompt and immediate action, especially near the ground. Airspeed, per se, is not likely to be immediately affected if the thrust angle is near vertical, but this does not mean that flight reference is correspondingly free from being disturbed during the transient.

In general, there is a longer time frame associated with the longitudinal control functions than with the lateral-directional ones. This is because the latter mainly involves roll and yaw attitude control and their effective time constants are relatively short. A change in flight path and especially in airspeed is usually a slower process though.

Again, we rely on findings based on specific examples to derive general insights.

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FINDING:

It is of prime importance to have an adequate level of flight path control power available very shortly after a propulsion system failure.

DISCUSSION:

For the broad class of powered-lift aircraft any power failure will result in some immediate increase in rate of descent forcing the airplane below its nominal glide path. It is necessary to provide sufficient incremental flight path control power so the pilot can quickly reverse the sinking trend, regain the nominal glide slope, and stabilize on it. The most critical constraints are clearance of obstacles beneath the approach path and proximity to the runway.

In general, the subject of flight path control power could be approached in the same way as for the normal approach and landing conditions (Sections 5 and 6). The main added element in the propulsion system failure situation is the degree of initial flight path error build-up prior to recognition and application of the appropriate piloting technique. This suggests that the flight path control power capability be commensurate with the degree of flight path upset as a result of the failure. While recommended numerical values to address this need have not been determined, the following cases help to point out important situations.

Of those cases considered in this simulation program, the one most clearly lacking a suitable level of flight path control power following a failure transient was STOL-X (Reference 14). The inability of this airplane model to maintain even its nominal flight path after a failure produced substantial problems. The OPUI γ - V characteristics are shown in Figure 8-2. Immediately after suffering an engine failure, maximum throttle would not quite maintain the initial flight path angle of -6 deg. Positive long-term flight path control power was available only after airspeed was increased. Thus, recapture of the glide slope was contingent on the ability to build up speed rapidly as well as relying on the basic short-term flight path response.

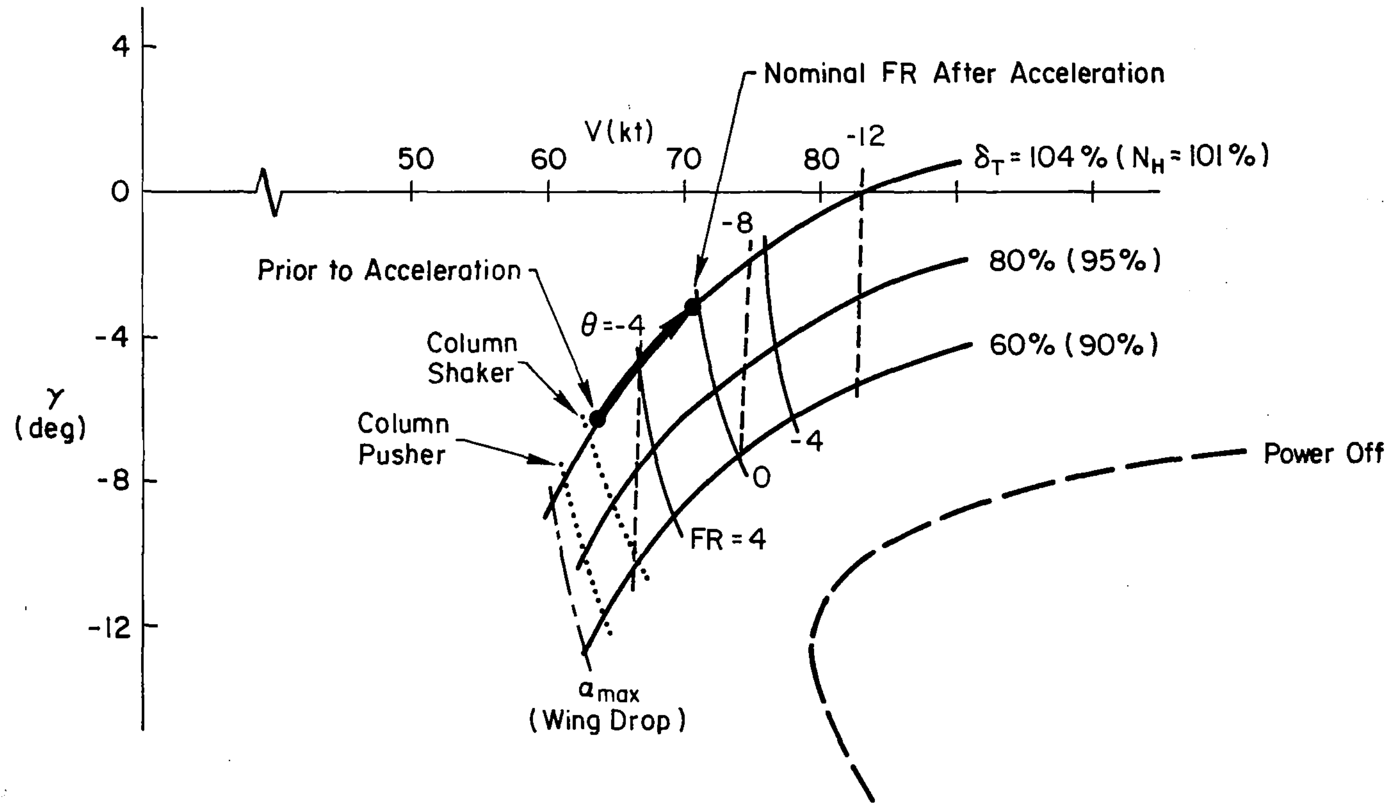

Figure 8-2: Trim γ-V Contours, OPUI, β=0

The key consideration is time. If the failure overly degrades flight path control power, there must be a way to restore an adequate amount quickly. Retracting spoilers or speed brakes would probably be an acceptable solution.

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FINDING:

There is a tendency to overshoot the glide slope following recovery from a propulsion system failure at low altitude.

DISCUSSION

During the STOL-X experiment (Reference 14) two pilots commented that they were having difficulty avoiding going above the glide slope following a failure and the application of full power. This overshooting tendency was a direct effect of fairly slow, but normally acceptable, flight path response. To avoid overshooting the glide slope one pilot purposely flew beneath it after a low altitude failure.

Reference 39 also noted that pilots tended to "duck under" the glide slope to avoid overshooting after a failure. Overshooting the glide slope could upset the pilot's timing during landing and could cause excessively long landings.

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FINDING:

Airworthiness standards should include a limit on the altitude deviation from the nominal glide path which could result from a propulsion system failure.

DISCUSSION:

One of the hazards of a propulsion system failure is the resulting uncontrolled descent below the nominal glide path. This is especially critical for powered-lift aircraft because the direct loss of lift results in immediate settling. The distance which the aircraft settles below the nominal glide path is one direct measure which addresses the likelihood of hitting an obstacle or landing short of the runway.

One possible means of defining the limit on path excursion follows. As an example, consider a 6 deg glide slope intercepting a runway 76 m (250 ft) beyond the threshold. In this case, the centerline of the glide slope passes over the threshold at an altitude of 8 m (26 ft). Assuming the lower extremity of an aircraft nominally follows 2 m (6 ft) below the glide slope, then a path excursion less than 6 m (20 ft) would assure a touchdown at or beyond the runway threshold. Naturally this dimension would vary depending upon the specific geometry for a given airplane, runway, and approach path.

Path excursions for three simulator models were obtained. Of these, the STOL-X model received the most analysis. For the 19 runs made by two pilots, the maximum path excursion following an engine failure transient averaged 8 m (26 ft) with a standard deviation of 3.7 m (12 ft). The largest excursion was 14 m (47 ft). Recall that the STOL-X was a four engine model and required a pitch down maneuver to increase airspeed. In view of the runway geometry constraints approximated above, one might consider the STOL-X flight path excursions to have been excessive.

The simulator model evaluated in Reference 44 had fewer adverse features. Like the STOL-X it was a four-engine EBF configuration but did not have the lateral control problems nor require an increase in airspeed. Descents below the glide slope following failures were typically 4 m (13 ft).

The AWJSRA simulator model exhibited the most severe altitude excursions following a propulsion system failure. The amount of available data is rather limited but it shows that the maximum deviation below the glide slope averaged roughly 15 m (50 ft). It is felt that the main factor in producing such large deviations was that it was a twin-engine aircraft (50% thrust loss) requiring a relatively complex reconfiguration procedure following a propulsion unit failure.

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FINDING:

The most critical propulsion system failures occur within a fairly narrow altitude band.

DISCUSSION:

A number of simulator experiments involving propulsion system failure have seemed to indicate the existence of a critical altitude band for the occurrence of a failure. This band is limited on the upper end by the pilot's ability to handle the failure transient and attain a reasonable state from which to flare the airplane to make a well-controlled landing. The lower extreme of this critical altitude band seems to be that altitude below which the pilot need take minimal action to counter the transient and achieve an acceptable compromise in safety margins and touchdown conditions. Between these two altitudes neither action is entirely satisfactory.

The critical band for the AWJSRA simulation (Reference 12) was considered to be between 60 m and 30 m (200 ft and 100 ft). In this band, the subject pilot felt he had little control over the outcome of the landing. In Reference 39 a less critical model was evaluated. The critical altitude band in this case was narrower and was closer to the ground, i.e., 23 to 12 m (75 to 40 ft).

It should be noted that no precise definition of this critical band of altitudes has been developed. Therefore, the values given are approximate. They are likely to vary depending on the pilot, the specific piloting technique, and the aircraft systems.

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FINDING

Following a propulsion system failure, no change in airspeed should be required in order to continue the approach with adequate safety margins and performance.

DISCUSSION:

During the first SSDWG meeting it was proposed that a continued approach with a propulsion system failure be permitted without a change in flight reference provided that other relevant requirements are met (e.g., safety margins, performance, etc.). The results of the STOL-X simulation which followed this meeting revealed certain factors which would affect such a proposal.

The STOL-X vehicle was an example involving a large change in airspeed (although not flight reference) following an engine failure. When an engine failed the flight reference calculation was changed so that a higher airspeed was required to maintain the nominal flight reference. There was no particular objection to the speed change for restoration of flight reference. The problem was the time necessary to obtain acceptable aircraft performance.

Specifically, the airspeed change was necessary to obtain positive flight path control power and lateral control power. In executing the airspeed change a pitch down was required which temporarily increased the flight path error. Finally, a substantial time lag was involved in attaining the desired airspeed.

The important point of this example was that all the conditions of the above proposed requirement were met, i.e., the same flight reference was flown following the failure and this flight reference did insure reasonable performance and safety margins; but the situation was unsatisfactory because of the time required to reach the steady condition. This time was, in turn, mainly due to the airspeed change required.

Unfortunately, the time required for airspeed changes is strongly related to the basic airframe speed damping. As indicated in Section 5 (and Appendix A), the speed damping time constant is likely to be on the order of 10 sec or longer for either powered-lift or conventional aircraft.

The implication is that in the case of a propulsion system failure no airspeed change should be required in order to continue the approach with adequate safety margins and performance because of the inherent time delay.

8.2 STEADY STATE CONTINUED APPROACH, OPUI

The following are the results related to approach and landing following a propulsion system failure and its associated transients. These results are concerned primarily with the steady state OPUI operation of the aircraft along the glide path and during the flare and landing after transient effects have been overcome.

Strictly speaking, the material contained here could be organized according to a scheme such as that implied by Sections 3, 4, 5, 6, and 7. The OPUI continued approach condition could be considered in terms of limiting flight conditions, safety margins, longitudinal stability, control, and performance, etc. Instead of such a rigorous classification we have chosen to deal only with those features which are especially interesting or significant to this failure condition.

This subsection begins with a background discussion of the pilot/vehicle characteristics which play an important role in the OPUI continued approach. Following this background discussion, we present the simulation results.

The physical characteristics which are important to this situation arise from asymmetric powered lift as described in Subsection 8.1. In particular, recall the diagrams shown in Figure 8-1. These ideas are developed in further detail for the steady state continued approach in the diagram of Figure 8-3. Each of the elements of this figure is expanded in the following discussion.

The objective of Figure 8-3 is to show the cause and effect relationships resulting from a propulsion system failure. The top diagram represents a conventional airplane with an asymmetric horizontal thrust loss. The other two diagrams represent powered-lift aircraft; one involving an asymmetric lift loss, the other, a symmetric lift loss. The direct effects shown are those resulting from the failure itself and the secondary effects are those stemming from the compensating actions taken by the pilot.

One general feature which Figure 8-3 shows is that there are significant differences in the characteristics between a conventional aircraft and a powered-lift vehicle regarding an OPUI continued approach. The

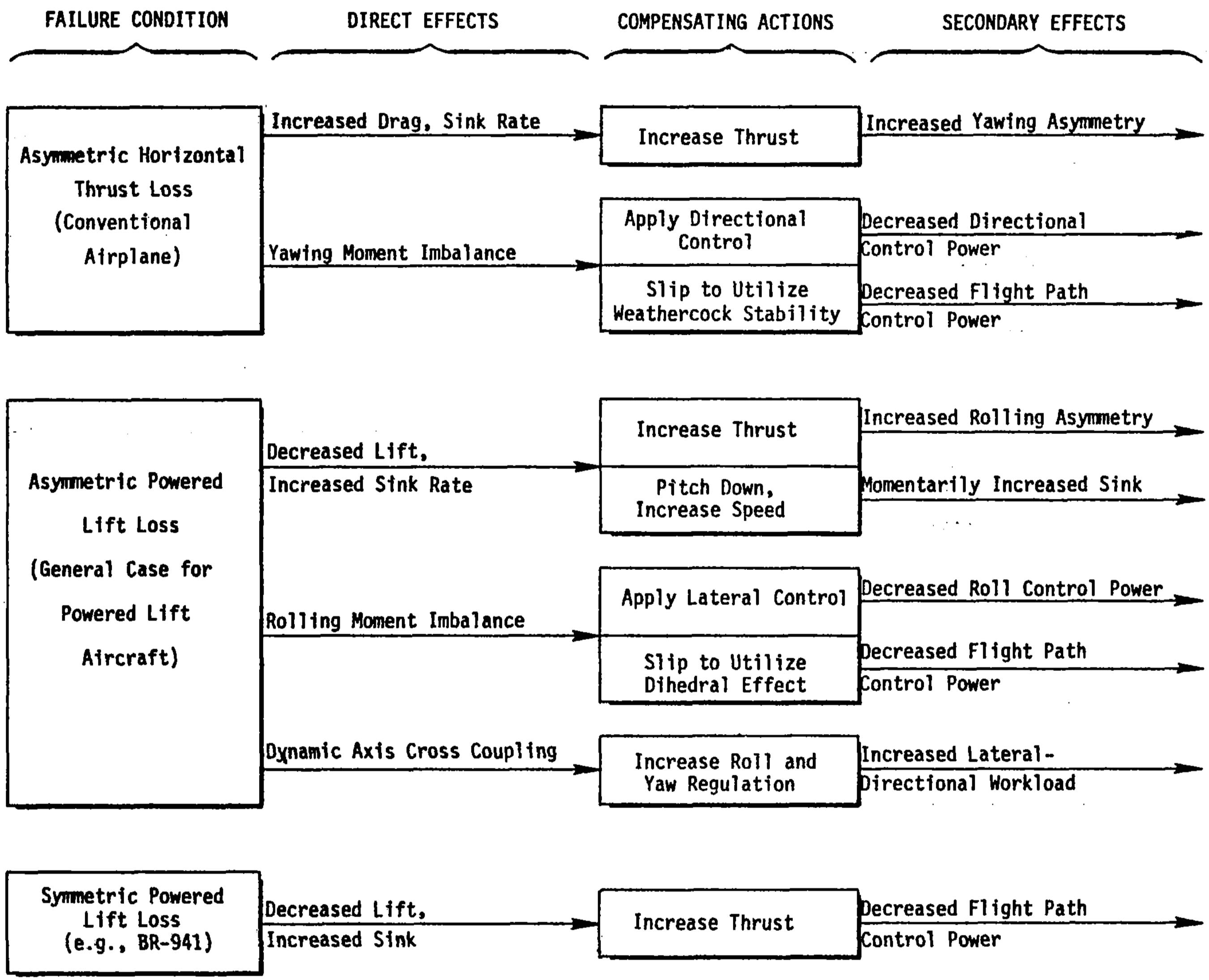

Figure 8-3: Propulsion System Failure Cause and Effect Relationships

fundamental difference, again, is the loss in vertical force versus a loss in horizontal thrust. This difference propagates through the direct effects, compensating actions taken, and resulting secondary effects.

Engine failure effects for powered-lift vehicles are configuration-dependent. Two extremes are shown in Figure 8-3. These consist of the clearly asymmetric powered-lift loss cases versus simple symmetric lift loss cases. In the simulation experiments conducted during this program a clearly asymmetric powered-lift loss was represented by the STOL-X model and the symmetric lift loss was represented by the BR 941. In the case of STOL-X an engine failure produced loss of lift on that side and a resulting rolling motion. This could be considered a normal failure configuration for powered-lift aircraft. The BR 941 used cross-shafted propellers; loss of power in any one engine was completely equivalent to a simple reduction in throttle setting. No lateral-directional upset was experienced, only an increased rate of descent.

The AWJSRA represented an interesting configuration variation aside from the two extremes described above. This aircraft involved a combination of vectored hot thrust and an augmentor wing jet flap using the fan air flow cross-ducted to the opposite wing. Loss of thrust in one engine resulted in an unusual combination of lateral-directional moments as mentioned previously.

One important distinction between thrust loss in a conventional airplane and a powered-lift airplane is the change in the critical lateral-directional control. For a conventional airplane where a yawing moment is produced, then the rudder is most likely to be the critical control. In contrast, the powered-lift airplane experiencing an asymmetric lift loss is likely to be critically limited in roll control. In both cases the critical lateral-directional control is subject to some relief or aggravation through use of sideslip.

The yawing moment in a conventional airplane can be offset somewhat by sideslip through the mechanism of positive weathercock stability. To the extent this is possible, it reduces the amount of rudder needed. If there were zero directional stability, then the entire upsetting yawing moment would have to be countered entirely by the rudder control, clearly a more critical condition. Correspondingly, in a powered-lift airplane, there is the possibility of using sideslip to partially offset the upsetting rolling moment, hence reducing the lateral control. This, however, depends upon the existence of a significant dihedral effect.

An important feature in using sideslip is that the sense of sideslip required to offset a rolling moment is likely to be opposite to that required to offset a yawing moment. (This presumes positive dihedral effect and positive directional stability.) Hence, steady-state flight for a powered-lift airplane would likely be with the failed engine forward; for a conventional airplane, the failed engine would be aft.

The longitudinal effects of a propulsion system failure are closely interrelated with the lateral-directional effects. Again referring to Figure 8-3, we can see that the direct effect of propulsion system failure for both conventional and powered-lift airplanes is to increase rate of descent. One case involves an increased drag while the other decreased lift. In both cases, though, thrust must be increased on the remaining engines and this correspondingly increases the resulting lateral-directional asymmetry problems.

The necessary consequence of operating with a failed engine is that the incremental flight path capability is reduced but can be altered by a change in trim airspeed. If the aircraft were operating initially on the backside then pitching down to increase airspeed could have a beneficial effect on upward flight path control power. During the process of pitching down, however, the airplane would suffer a momentary increase in sink rate from the pitch down. The net effect would be to experience a significant time delay before obtaining an increase in flight path control power.

One of the most serious longitudinal deficiencies following propulsion system failure can be the increased difficulty in flare and landing. There are probably a number of variations in the kind and degree of difficulty, but they are variations likely to be a result of decreased flight path control power and a substantial change in the nominal operating point. The latter might involve pitch angle, airspeed, and lateral-directional conditions. The simulator models considered in this program revealed a number of problems in the flare and landing which will be described in this section. While not a systematic study, it is considered to touch upon the elements involved in a wide variety of powered-lift designs.

The final effect we shall mention in connection with asymmetric powered-lift propulsion system failures is the effect of dynamic cross coupling. This was not studied in this program either analytically or experimentally, but was found while analyzing the results of STOL-X for this report. This coupling effect involves the creation of longitudinal and lateral-directional cross coupling moments. These can be described as rolling and yawing moments arising from angle of attack and airspeed perturbations. As discussed below, the direct cause is asymmetric powered lift.

One of the important properties of jet-flap powered lift is an increase in the slope of C_L versus α, C_{L_α}. If there is asymmetric blowing from an engine failure, then the C_{L_α} on one wing can differ from that on the other. Hence, an angle of attack perturbation can produce a rolling moment through differential lift. This would create a coupling stability derivative, L_w. Correspondingly, there can be a differential drag with angle of attack which would produce a net yawing moment with angle of attack, N_w. Similar reasoning can be applied to differential lift and drag due to an airspeed perturbation. This would produce rolling and yawing moments through airspeed perturbations, L_u and N_u.

The cross coupling effect described above has not been addressed either analytically or experimentally. Therefore, the potential effect is not fully appreciated. One should note that the cross coupling possible in powered-lift aircraft exists in other flight vehicles. Two notable examples are helicopters and some fixed wing aircraft at very high angles of attack. We should hasten to mention, though, that the specific nature of cross coupling can differ greatly between these examples. The mere existence of coupling may be the only common factor.

While the impact of axis cross coupling on closed loop pilot/vehicle control has not been thoroughly addressed for helicopters, it has been for the high angle of attack condition. Reference 45 describes an analytical approach to nose slice divergence in an attack aircraft. The important factors in the lateral divergence were found to be coupling stability derivatives L_W and N_W in conjunction with regulation of pitch attitude. A similar form of analysis could be applied to powered-lift aircraft.

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FINDING:

While the general effects of powered-lift loss are always present to some degree, many important effects cannot be completely generalized because they are related to a specific design or mode of operation.

DISCUSSION:

The BR 941 and STOL-X vehicles are cited as examples of symmetric and asymmetric powered-lift loss. The cause-and-effect relationships connected with steady state operation OPUI could be typical of a number of powered-lift designs. The AWJSRA vehicle, on the other hand, involves certain design complexities which alter the powered-lift loss characteristics. Cross ducting of cold thrust caused the aircraft to roll in the opposite direction to that expected, i.e., failure of the right engine produced a left wing-down rolling moment. The cross ducting effect, of course, was minimized by the vectored nozzle hot thrust which provided the usual sense of rolling moment. This rolling moment balance, however, was destroyed if the nozzles were vectored from a vertical orientation to a more horizontal orientation as was the practice. This produced still another set of unusual applied moments for the pilot to counter.

Other features which could have an effect on the nature of a powered-lift loss are aerodynamically augmented (blown) control surfaces, automatic reconfiguration devices, and various forms of cross ducting or cross shafting.

The powered-lift concept employed (e.g., EBF, USB, IFB, etc.) could be a determining factor in the nature of powered-lift loss. For example, in a four engine EBF or USB airplane, failure of an outboard engine might produce a greater rolling moment than for a comparable IBF design due to distributed blowing of the latter.

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FINDING:

Airworthiness standards should require adequate OPUI roll performance for all likely trim conditions.

DISCUSSION:

A minimum rolling maneuver capability is recognized as an important requirement in all phases of flight independent of any failure condition. This finding is stated here to clearly associate this kind of deficiency with an asymmetric powered-lift loss condition.

The vehicle referred to here is the STOL-X model. The OPUI roll performance capability of this model is described in Figure 8-4. The performance is given in terms of maximum steady-state roll rate but it is apparent that no single value can be assigned to this vehicle for the OPUI condition. Rather, roll rate capability is a function of airspeed (or flight reference), thrust setting, and sideslip (in this case, zero sideslip compared to that sideslip obtained with zero pedal input).

The roll rate performance for the nominal flight reference and flight path angle in zero sideslip was 16 deg/sec. This is actually in excess of the minimum acceptable value suggested in Reference 6. It is hypothesized, though, that the pilot rarely experienced such a high level of roll performance during most of his approach. The pilot's displeasure with roll performance was due to the tendency toward high power setting and adverse sideslip as a result of zero pedal displacement. The nominal roll performance was probably adequate but it degraded so rapidly for likely off-nominal trim conditions that the pilot really experienced a lower effective roll performance.

The implication of the above discussion is that a roll performance requirement should account for likely off-nominal trim conditions especially in throttle, sideslip, and flight reference. One particularly demanding flight condition is the return to the glide path following failure. Here the pilot is required to roll away from the dead engine with maximum thrust on the remaining engines.

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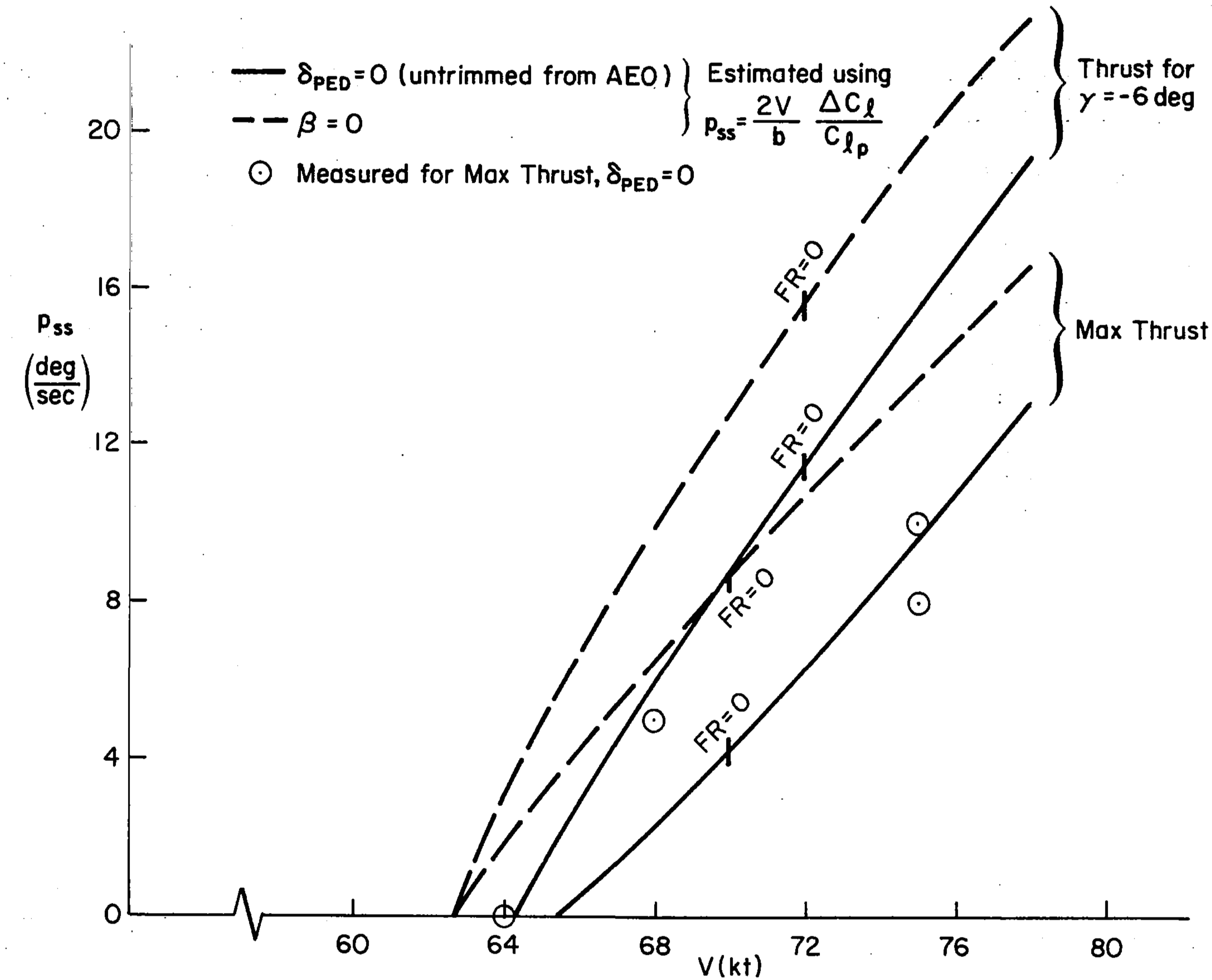

Figure 8-4: Roll Rate Capability STOL-X Vehicle, OPUI

FINDING:

High lateral control forces were an important problem experienced in a configuration involving asymmetric powered-lift loss.

DISCUSSION:

High lateral control forces were cited as a problem for the STOL-X engine failure transient condition. These high forces remained a problem for the steady state condition even though the magnitudes of the forces were significantly less. One pilot commented that the large forces and deflections involved caused him to make coarse inputs resulting in poor lateral-directional control. Further, he claimed it was difficult to hold a steady wheel angle and was always hunting in roll even though thrust was held constant.

In the STOL-X case the reason for the high forces was saturation of a command augmentation system. Had the system not saturated, the effective gearing and resulting forces would have been significantly more favorable.

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FINDING:

Subject pilots exhibited a reluctance to trim out lateral or directional control forces during an OPUI approach.

DISCUSSION:

It was expected that the high forces mentioned previously would be eliminated by manual trimming. A number of pilots, however, were reluctant to trim and preferred to simply hold the high wheel forces. There were several factors which contributed to this reluctance and these should be considered. The first was that the lateral trim rate was low. This tended to demand excessive attention of the pilot. Another important factor was that the amount of trim required changed radically with thrust, flight reference, and sideslip as noted previously.

The implication is that the mere existence of a lateral trim system may not be effective in countering high lateral control forces. Nevertheless, some attention should be given to providing a reasonable trim system.

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FINDING:

Lateral-directional SAS saturation creates handling problems, but saturation is not necessarily obvious to the pilot.

DISCUSSION:

Lateral-directional stability and control augmentation was used in all of the vehicles in this program as well as those of other STOL vehicles to improve roll response and turn coordination. In the STOL-X simulation it was found that certain features of a propulsion system failure can saturate important augmentation paths thereby degrading lateral-directional handling qualities. This kind of degradation, though, was not especially obvious to the subject pilots even though it caused an increased workload.

The SAS used in the STOL-X vehicle is described in Reference 14 and can be considered reasonably typical of a lateral-directional SAS for powered-lift aircraft. The lateral stability augmentation system employed feedback of roll and yaw rates to the lateral surfaces to improve roll damping and spiral stability. A roll rate feed forward path was used to provide good control sensitivity. The lateral SAS authority was about one quarter of the total lateral control capability. These augmentation authority limits were seldom reached except during a propulsion system failure condition.

When lateral SAS saturation occurred the main effect was to reduce the effective lateral control sensitivity by a factor of approximately 2.5. As a result, it took 2.5 times as much control deflection and force to generate the same rolling moment. This effect was readily apparent to the pilot as noted in a preceding finding.

Another important effect of SAS saturation was loss of stability augmentation functions. The reduction in spiral stability appeared to be the most prominent effect. It was noted specifically that the spiral divergence was rapid and that it was difficult to keep the bank angle constant. The lateral workload was characterized as intense.

While the lateral SAS was totally saturated, the directional SAS was only partially saturated, that is, it was continually driven in and out of saturation during a dutch roll cycle. One function of the directional SAS was to provide much needed turn coordination. This was done through feedback of bank angle and yaw rate to rudder. A pedal-to-rudder surface command augmentation was used as well as a wheel-to-rudder crossfeed. Each of these stability and control augmentation elements were limited in authority to approximately one fifth the total directional control authority.

Because the subject pilots avoided holding any steady rudder pedal forces, the directional SAS was not driven to saturation through the use of rudder pedal. Instead the directional SAS was driven to the borderline of saturation by the wheel-to-rudder crossfeed. This, then, is an example of a peculiar case where the directional SAS was saturated indirectly by use of the lateral control. This condition was not foreseen but is presented here to illustrate a configuration-dependent problem.

This partial saturation condition was not readily apparent to the pilot. One reason is that the turn coordination function of the SAS is only partially impeded. In fact, for a turn in a favorable direction this system is likely to function normally but for a turn in the other direction, turn coordination could be lost completely. Removal of SAS authority limits produced a significant improvement in overall pilot/vehicle performance.

This finding serves as a warning of some of the potential handling qualities problems stemming from a propulsion system failure where a heavily augmented airplane is concerned. The problems cited are peculiar to the vehicles studied but could be present in other vehicles along with other unexpected problems.

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FINDING:

Use of sideslip to improve lateral control was not an obvious course of action.

DISCUSSION:

The STOL-X simulator model was generally regarded as being deficient in roll control power for the reasons cited previously. It possessed, however, a positive dihedral effect. Therefore, some amount of nulling rolling moment was available to relieve the total dependence on lateral control. This benefit was not apparent to either the pilots or engineers during the simulation experiment itself. It became apparent only after the post-simulation analysis. In fact, not only was this beneficial effect not detected but there was a distinct tendency to slip the aircraft in an adverse direction.

In the STOL-X simulation, two of the subject pilots tended to fly with zero sideslip but the majority flew with zero pedal force which corresponded with approximately five degrees of sideslip in the adverse sense. It is presumed that the reason for the preference demonstrated was i) it avoided holding rudder forces or retirmming directionally, and ii) it was in the direction of slip natural for conventional aircraft, i.e., in the direction to counter a yawing moment rather than a rolling moment.

In early familiarization stages, there was a strong tendency for the subject pilots to hold rudder pedal in the conventional sense. This resulted in a nearly complete loss of lateral control and divergence in lateral flight path. Consciously avoiding use of rudder control markedly increased the pilot mental workload. Unfortunately, no attempt was made to evaluate the benefits of slipping in a favorable direction.

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FINDING:

Inadequate flight path control power is viewed as the main longitudinal problem during the OPUI approach phase for powered-lift aircraft.

DISCUSSION:

Whether conventional or powered-lift, an aircraft naturally suffers degraded flight path performance following a propulsion system failure. For a powered-lift airplane, however, the degradation takes place far more rapidly because of the direct loss of vertical force. This, however, was adequately discussed in the subsection dealing with the transient condition. Nevertheless, it is worth noting that for the continued OPUI approach condition, degradation of steady state control power is an important and identifiable effect. This was borne out most clearly by the STOL-X simulation for which this condition was made marginal intentionally.

Total flight path control power (maximum down to maximum up) will obviously be less OPUI. Airworthiness requirements on OPUI flight path control power must be carefully considered as they can have a dramatic impact on vehicle design. It may be acceptable to require less downward capability for OPUI. Then a reconfiguration (e.g., spoiler or speed brake retraction) could be used to regain adequate upward capability. Reduced requirements for OPUI might be justified on the basis of the probability of exposure. Existing regulations for conventional aircraft require less performance with an engine failed.

It should also be noted that flight path control power can be a strong function of flight reference and sideslip angle, just as roll control power. Further, enhancement of roll control power by sideslip can degrade flight path control power. Hence there can be a complex tradeoff between the two. This is illustrated in Figure 8-5. Note that the upsetting moments are dependent on speed and throttle while available control moments are dependent on speed and sideslip. The shaded boundaries represent steady state flight path capability for various available control moment conditions. As more sideslip is used to produce a higher control moment, more speed is required to maintain flight path control power.

Increased roll rate capability can be directly related to sideslip and incremental flight path control power in the following manner.

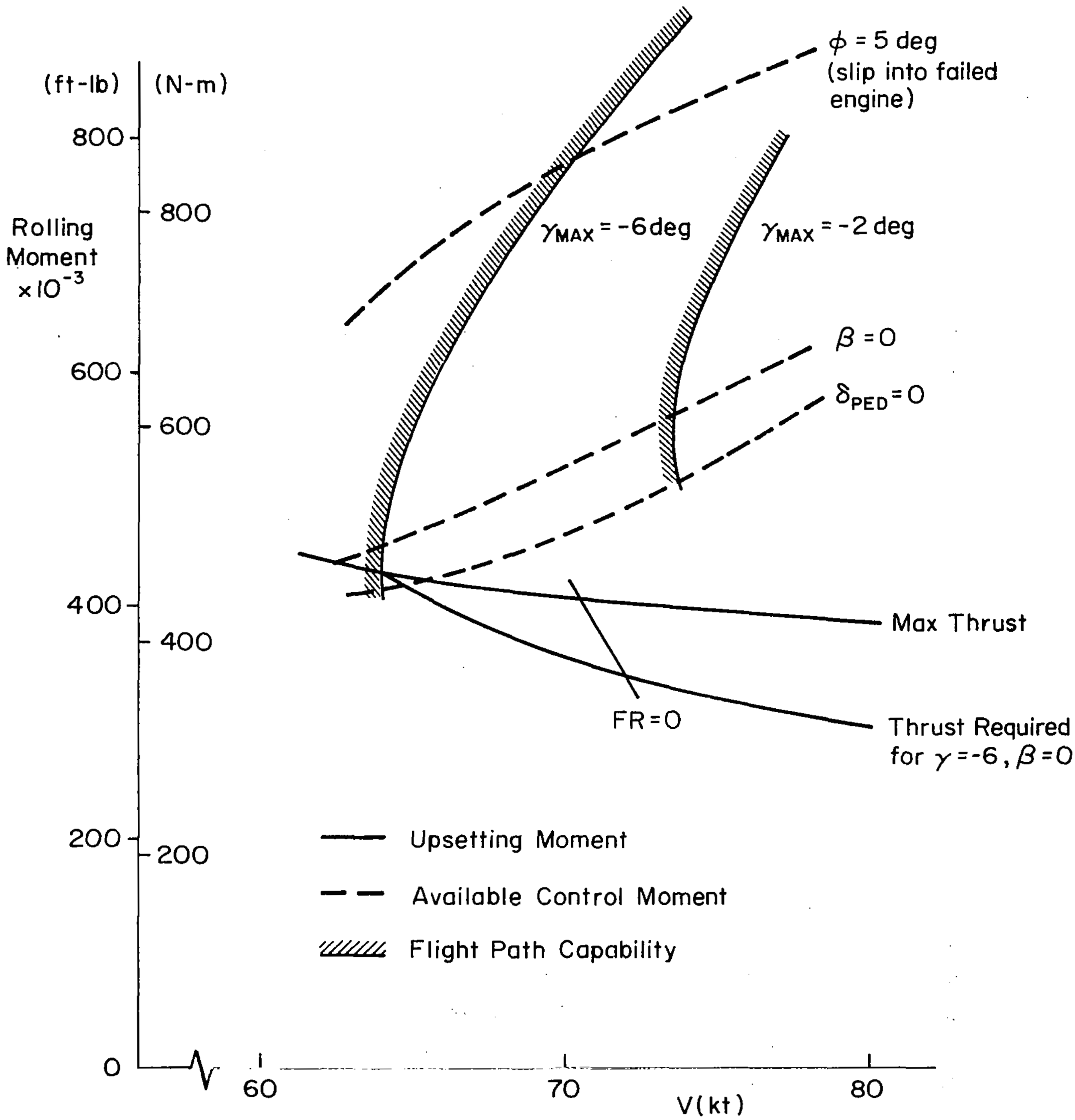

Figure 8-5: Upsetting and Restoring Moments
STOL-X Vehicle, OPUI

First, from increased drag: $\Delta\gamma \doteq \frac{C_Y \sin\beta}{C_L} \doteq \frac{C_{Y_\beta}}{C_L}\Delta\beta^2$

Next, $\Delta p \doteq \frac{2V}{b}\frac{C_{\ell_\beta}\Delta\beta}{C_{\ell_p}}$

Combining these: $\Delta p \doteq \frac{2}{b}\frac{C_{\ell_\beta}}{C_{\ell_p}}\sqrt{\frac{W/S}{\rho/2}\frac{\Delta\gamma}{C_{Y_\beta}}}$

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FINDING:

OPUI continued approaches tended to follow a path beneath the glide slope.

DISCUSSION:

This is a tendency that was observed during the STOL-X simulation and is suspected to be strongly associated with the initial amount of sink and difficulty in increasing flight path angle, that is, lack of flight path control power. The ultimate advantage of this course of action was to approach the flare with a lower than normal sink rate. This can be beneficial in the flare and landing where there can be significant problems. This will be discussed next. A similar tendency was noted in the powered-lift simulation reported in Reference 39.

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FINDING:

The flare and landing phase was complicated significantly by propulsion system failure.

DISCUSSION:

Two important ways that the flare and landing characteristics can be altered by a failure are:

- A significant change in operating point as related to pitch attitude and airspeed
- A degradation in flare control power and introduction of lateral-directional asymmetry if thrust is normally used in the flare maneuver.

Both of these factors played a role in the STOL-X simulation.

Recall that the normal flare technique used for the STOL-X vehicle was to flare using throttle. If the engines responded rapidly, this was a satisfactory means of flaring. Following a failure airspeed was increased and pitch attitude was lowered. The resulting pitch attitude was sufficiently nose-down that a pitch-up maneuver was required prior to an attempted flare with power. The pitch attitude change required was so large that it significantly reduced sink rate without a throttle input. The throttle by itself was relatively ineffective because of inadequate flight path control power. The pilot was effectively forced into using a flare with pitch attitude. It should be noted, however, that this change in technique was not regarded as specially troublesome. The main problem associated with this condition was a tendency to balloon in the flare because of the large pitch attitude change required to avoid touching down nose wheel first.

Flare and landing during the failure transient stage was significantly more difficult than that just described.

If the pitch attitude change and resulting change in flare technique were not involved as in the above case, then the critical aspect of the flare would be lack of flare control power. In cases where no change in operating point is involved and the normal flare maneuver is with attitude, then no significant increase in flare and landing difficulty should be expected OPUI. The BR 941 was the only such example considered in this simulation program.

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SECTION 9

AUGMENTATION SYSTEMS FAILURE

For any aircraft careful consideration must be given to potential failures of stability and command augmentation systems (SCAS). This is likely to be a more serious problem for powered-lift aircraft than for modern subsonic jet transports. The basic cause is the rather poor handling qualities which are typical of many powered-lift designs. In the approach configuration, a powered-lift aircraft could exhibit several of the following problems:

- Low longitudinal static stability or low short period frequency
- Small phugoid/short-period frequency separation
- Slow thrust response
- Operation well on the backside of the power required curve
- Poor heave damping
- Poor turn coordination
- Unstable spiral mode
- Low roll damping
- Low dutch roll damping and frequency.

These deficiencies can be corrected by a SCAS but the system may have to be relatively complex. For example, the SCAS for the McDonnell Douglas YC-15 is a dual redundant system (Reference 46). Inputs to the SCAS come from a sensor complement which includes:

- Column, wheel, and rudder pedal forces
- Pitch, roll, and yaw rates
- 3-axis accelerometer

- Vertical gyro (pitch and roll attitudes)
- Heading
- Sideslip
- Air data.

The system outputs include:

- Stabilizer trim
- Elevator
- Ailerons
- Upper and lower rudder segments.

The more complex the system, the greater the probability of a failure of some element and the more the different failure conditions which must be considered. Another important aspect of the problem is the potentially large change in aircraft characteristics if a SCAS function is lost. Thus the pilot may have to cope with a wide range of failure conditions and some of these could drastically alter the aircraft characteristics.

As with propulsion system failures, the problems due to a SCAS failure can be divided into the failure transient and steady state operation after the failure. It is hard to generalize about SCAS failure transients because the transient depends so strongly on the system mechanization. One important consideration is the extent to which the SCAS augments the basic lift and drag characteristics (e.g., by manipulating power and flaps). A failure could leave the pilot with an unusual configuration which presents problems in maintaining adequate performance or safety margins.

After the initial failure transient the pilot may have a variety of steady state problems. We will consider a few possibilities.

A failure may simply increase the pilot workload but not really affect pilot/aircraft performance. Loss of the roll damper could cause the pilot to work harder to maintain adequate lateral control.

A failure could change the magnitude of the aircraft responses to the pilot's inputs. The severity of the change would depend on which responses were affected and how much time the pilot had to adjust to the changes. One potentially serious failure would be of a system which augments heave damping and n_{z_α}. The pilot may not fully appreciate the effects of the failure until he attempts the flare. If he makes the normal pitch change, he may suddenly find he is not adequately breaking sink rate.

Probably the worst type of failure would be one which required a complete change in piloting technique. The SCAS could provide automatic flight reference regulation so the pilot could fly a CTOL technique using the column to control pitch and vertical path. Failure of this SCAS could require the pilot to switch to the STOL technique. The severity of this problem would depend on the altitude where the failure occurred and how familiar the pilot was with the STOL technique.

During this program a brief attempt was made to investigate failures of this last type. This was done during the AWJSRA simulation (Reference 12). An automatic speed control system was used in a few runs. This system, essentially an autothrottle, permitted the pilot to control vertical flight path using a conventional piloting technique. This technique was not usable with the bare airframe because it was well on the backside of the drag curve. The test procedure was to familiarize the pilot with the system and then to explore the difficulties when the system failed. One aspect of particular interest was how easily the pilot could adapt his piloting technique from the CTOL technique back to the STOL technique required for the bare airframe.

The results of this short experiment were relatively inconclusive. It was not possible to produce any significant element of surprise. After a series of familiarization runs using the augmentation system, a failure was tried. The pilot found it easy to readapt to the STOL technique following this failure. At the same time, he suggested that his use of the STOL technique in recent experiments was a factor. The implication was that significant long term training and experience were required using the augmented vehicle before one could make a valid determination of the effect of this kind of augmentation system failure.

Some of the results reported in Reference 21 also bear on this problem. That report describes a simulation experiment to investigate potential problems of powered-lift aircraft operating in wind shears. Several different powered-lift designs were simulated, including one with a sophisticated SCAS of the type under discussion here. The SCAS provided automatic speed regulation by feedbacks to the throttle and a DDC. The pilot flew with a CTOL technique using only the column.

The simulation experiment did not include SCAS failures but did allow manual disconnect of the system. In a few cases the pilot did disconnect the SCAS when he observed an excessive airspeed error (the DDC had saturated). After disconnecting the system, the pilot had a great deal of difficulty controlling the aircraft even though he had flown without the SCAS earlier in the program. He rated the situation as quite hazardous. In repeat runs with the same shear, the pilot did not disconnect the SCAS and had little problem completing the landing.

The manual disconnect is analogous to a system failure in causing a drastic change in aircraft characteristics. One might argue that the adverse conditions at disconnect exaggerated the problems relative to a SCAS failure. This may be true if the SCAS were completely fail-passive but is questionable if hardover failures were possible. The applicability of these simulation results to the problem of SCAS failures is admittedly debatable.

The paragraphs above describe some of the concerns and considerations regarding augmentation system failures in powered-lift aircraft. While this program did little directly to quantify the problem or develop criteria, this area is an important one relative to safety. It should be addressed in the airworthiness standards.

SECTION 10

GO-AROUND

The go-around maneuver for powered-lift aircraft can be expected to differ from conventional aircraft in three main areas. These are:

- Piloting procedure required
- Vertical path control qualities
- Control of asymmetries due to a propulsion system failure.

The approach used in the simulator was to consider specific airplane examples and their behavior in the go-around maneuver. The particular examples used were the BR 941 and AWJSRA simulations (References 11 and 12). While no formal parameter variation was run, these examples exposed some of the features which must be considered in establishing airworthiness standards.

Prior to discussion of individual elements of the go-around maneuver as indicated above, we will mention some of the main features of powered-lift aircraft which influence or determine go-around characteristics.

One of the prime influencing factors on the go-around is the likelihood of steep approach flight path angles combined with the relatively high thrust-to-weight setting necessary to sustain powered lift. The arrestment of sink rate and the establishment of a positive climb gradient must be accomplished by either a large change in thrust-to-weight ratio or a change in configuration to decrease the drag-to-weight ratio. The first of these implies, perhaps, a design penalty in the choice of propulsion systems, i.e., more thrust than is required for an economical cruise. The second implies the possibility of increased pilot workload connected with a configuration change. Both of these elements entered into the examples which were examined in the simulator.

The near-vertical effective thrust inclination to be found in many powered-lift aircraft seems to have special relevance when considering the go-around maneuver. This seems to be the feature which is the direct cause of the problems mentioned in the above paragraph. A large vertical thrust component is relatively inefficient in providing steady state climb performance. This is qualitatively illustrated in the comparison of powered-lift with conventional aircraft in Section 5.

Based on approximate factors from Appendix A the following shows the general relation between steady state flight path change and thrust-to-weight change:

For $u = 0$

$$\frac{\Delta\gamma}{\delta_T} = \frac{X_{\delta_T}\left(Z_w - \frac{g}{V}\sin\gamma_o\right) + Z_{\delta_T}\left(\frac{g}{V}\cos\gamma_o - X_w\right)}{g\,(Z_w\cos\gamma_o - X_w\sin\gamma_o)}$$

or $$\Delta\gamma = \frac{\Delta T}{W}\left[\frac{n_{z_\alpha}\cos\theta_T + n_{x_\alpha}\sin\theta_T}{n_{z_\alpha}\cos\gamma_o + (1 - n_{x_\alpha})\sin\gamma_o}\right]$$

$$\doteq \frac{\Delta T}{W}\left(\cos\theta_T + \frac{n_{x_\alpha}}{n_{z_\alpha}}\sin\theta_T\right)$$

Thus, $\frac{\Delta\gamma}{\Delta T/W} \doteq 1$ for conventional aircraft

and, $\frac{\Delta\gamma}{\Delta T/W} \doteq \frac{n_{x_\alpha}}{n_{z_\alpha}} << 1$ for powered-lift aircraft

Just as the near-vertical thrust inclination was the prime determining factor in approach and landing flight path dynamics, in a very real way, it is also the prime factor in the go-around situation.

The following deals with some of the more detailed aspects concerning go-around and findings from the simulator experiments.

10.1 PILOTING PROCEDURE; GO-AROUND

Based on the performance of the airplanes viewed in the simulator, it is likely that the go-around task can involve a configuration change. Airworthiness standards should consider this likelihood. The complexity of a configuration change can vary widely. In this program we viewed two extreme examples.

The BR 941 simulation (Reference 11) involved a relatively simple procedure for transition to go-around. This consisted of adding full power and activating a thumb switch on the throttle which made the first-stage configuration change of flaps and transparency. The performance increase was large in magnitude and occurred quickly. It produced a rapid change in flight path ending in a positive climb and allowed the pilot, at this convenience, to transition to an even improved climb configuration with further flap retraction.

The opposing example was the AWJSRA simulation (Reference 12). It required a fully manual resetting of power, flaps, and nozzle angle. In this case, the assistance of the copilot was important.

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FINDING:

A configuration change procedure such as that used in the BR 941 simulation was acceptable for the go-around maneuver.

DISCUSSION:

Quantification of the elements which made this go-around situation acceptable is not really possible. Some of the factors, however, are important to note. Probably the main feature of this go-around mechanization was that actuation of the thumb switch which initiated go-around was part of the same movement involved in selecting maximum thrust. The go-around switch was mounted atop the throttle such that when the throttle was advanced the actuation of the thumb switch was a simple action. There was no additional hand movement required of the pilot. In addition, the switch was not required to be put in any particular position, rather, it only had to be pushed forward and held. Completion of the entire go-around configuration change required that the pilot hold the switch forward for approximately two seconds.

The full go-around procedure, in detail, consisted of the following:

- Advance throttle to maximum and actuate the thumb switch on the throttle handle (flaps were automatically retracted to 70 deg and transparency was reduced from 12 to 5 deg)
- Stabilize airspeed at 60 kt
- After a stabilized climb condition is established, manually raise flaps to 45 deg and increase airspeed to 70 kt.

By contrast, the AWJSRA simulation model involved a complex and entirely manual go-around procedure. Aside from a throttle advance, flap deflections and nozzle angle had to be reset in proper sequence. Subject pilots usually desired the assistance of a copilot.

It is recommended that airworthiness standards permit a configuration change for the go-around maneuver if it involves the degree of ease exemplified by the BR 941 simulation as described here. The salient feature seems to be the specific degree of added workload in effecting the configuration change necessary to establish a stabilized, positive rate of climb.

10.2 VERTICAL PATH CONTROL; GO-AROUND

Vertical path control is really the prime objective of the go-around task. It involves consideration of both dynamic and steady-state performance. The use of powered lift is likely to have an impact on both of these. An additional complication is that path performance is dependent upon a changing configuration rather than on a static configuration.

The steady-state vertical path performance is synonomous with vertical path control power as described in the approach section. Dynamic vertical path control depends on the speed and magnitude of the configuration change as well as the airframe heave damping, thrust inclination, and thrust response. It is clear that if the configuration change is done manually, then a variability in pilot response or technique can produce a variability in short-term path performance. The short-term response is the most critical with regard to arresting sink rate. To set any performance standards, the imprecise and variable actions of the pilot must be considered.

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FINDING:

The dominant criterion for a go-around is the height loss after initiation of the maneuver.

DISCUSSION:

Less important features would be time to arrest sink or time to achieve a given climb gradient. The real test of a go-around maneuver, however, is whether terrain contact is avoided.

Putting configuration change aside, the parameters which determine the height loss following go-around initiation are: airframe heave damping, thrust response, effective thrust inclination, and available thrust. These combine in the manner described in the approach section. The effect of a configuration change is generally to reduce the thrust inclination and the drag. If a configuration change is to be effective then it must take place quickly, but any configuration change introduces the potential of pilot variability.

The BR 941, as simulated in this program (Reference 11), represented something of an ideal with respect to configuration change variability. The only variability potential was really the difference in thumb switch actuation relative to throttle advance. Since these were accomplished by a single hand on a single controller lever, they tended to occur at essentially the same time.

It is recommended that in order to look after the combined effects of airframe, propulsion, and configuration change in the go-around maneuver, a limit be put on altitude loss directly. Further, this altitude loss should be demonstrated and should allow for reasonable pilot responses and reaction times. Finally, it is suggested that the procedure involve only the pilot and that he initiate the go-around without taking hands off the primary and secondary controls.

10.3 PROPULSION SYSTEM FAILURE; GO-AROUND

A propulsion system failure in conjunction with a go-around maneuver represents the ultimate aggravation of piloting problems and degradation of aircraft performance as they relate to the go-around task. The same general considerations with regard to go-around remain, that is, piloting procedure and vertical path control. In addition, though, one must treat the problems associated with an inoperative power unit. The following finding is related to this situation.

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FINDING:

An OPUI go-around suffers from increased pilot workload, degraded performance, and possible lateral-directional asymmetry which is aggravated by use of maximum thrust.

DISCUSSION:

This finding was demonstrated in a qualitative way by the BR 941 and AWJSRA examples (References 11 and 12). The BR 941 was a case in which pilot workload during the go-around was essentially unaltered by an engine failure. The propeller cross-shafting precluded lateral-directional asymmetry. At the same time, the propulsion failure significantly degraded the go-around path performance. Following go-around initiation, the aircraft descended far lower than without a failure and the steady-state climb performance was reduced considerably.

In the AWJSRA all features were degraded in the go-around maneuver. A change in configuration was required for either an engine failure or go-around initiation. The engine failure produced tricky lateral-directional asymmetries and the single engine climb performance was only marginal.

Probably the main lesson learned in this area was that propulsion failure in conjunction with go-around can present a serious safety problem. This safety problem is connected directly to both pilot workload and degraded airplane performance. Further, this presents a difficult tradeoff between the degraded performance allowed versus the lower probability of occurrence.

SECTION 11

TAKEOFF

The discussion of the takeoff flight phase is divided into three main parts:

- Limiting flight conditions and safety margins
- Stability, control, and performance
- Propulsion system failure.

Each of these parts is approached in the same way as for the landing flight phase but involves the special considerations pertaining to takeoff.

Generally speaking, powered lift is not expected to be as influential a factor in takeoff as in landing, at least where pitch rotation is used to become airborne. If lower flap settings are used for takeoff, the effective thrust inclination is more nearly horizontal and there is less of a powered-lift effect. These tendencies were observed in those aircraft simulated in this program. In fact, those particular examples would have been more fairly characterized as high thrust-to-weight conventional aircraft rather than powered-lift in their takeoff configurations.

It should be noted that no consideration was given in this program to takeoffs using methods other than pitch rotation (e.g., thrust vectoring). Therefore, the results should be viewed within this limited scope.

The simulator examples examined in the takeoff flight phase consisted of the BR 941 and AWJSRA airplanes. These two examples were sufficiently different to reveal a variety of features associated with the takeoff flight phase. The biggest difference was connected with the degree of asymmetry following failure of a propulsion unit. Another was the effect of propulsion failure on performance.

The simulation facility and models used in this program allowed starting with a takeoff roll at approximately 20 kt, acceleration to V_R, takeoff rotation, and continuing through second segment climb to 335 m (1100 ft). The experimental procedure used for the takeoff flight phase was to have the subject pilots try a variety of abuses. Specifically, abuses were applied to V_R and V_2 with and without propulsion system failures. In addition, the effects of wind and turbulence were studied. Since basic performance capabilities of each aircraft were known, the simulator allowed us to observe the combined effect of the pilot and aircraft with regard to takeoff performance.

11.1 LIMITING FLIGHT CONDITIONS AND SAFETY MARGINS; TAKEOFF

Limiting flight conditions and safety margins for the takeoff flight phase involve similar considerations as those discussed for the approach and landing flight phases, at least after the aircraft has become airborne. Prior to lift-off, however, there is the additional consideration of ground related limiting flight conditions and safety margins. In this respect, there is nothing fundamentally different between conventional aircraft and powered-lift aircraft.

Once airborne, limiting flight conditions are associated primarily with aerodynamic stall just as in the case of approach and landing. Further, limiting flight conditions are definable in the same terms. It should be noted that there is a simplification because only one power setting is involved.

Safety margins in connection with takeoff flight phase have a dynamic aspect. Safety margins are constantly changing as the airplane is accelerated up to the point of establishing a steady initial climb. The safety margin ideas developed for the approach and landing flight phase also apply to the takeoff, at least after becoming airborne. This includes such items as:

- Speed margins
- Angle of attack margins
- Maneuver margins
- Gust margins.

The numerical values of these margins may differ from those in the approach and landing because the pilot actions are different. In particular, maneuvering and tracking are not as important a part of the takeoff flight phase. In addition to these flight conditions margins, one must consider safety margins with respect to the terrain.

Prior to becoming airborne, safety margins are defined mainly in terms of the geometric constraints as delineated by runway boundaries. There is, perhaps, a fine distinction in this case between safety margins and performance in that safety margins are the difference between runway available and actual takeoff performance. As such this aspect of safety margins will not be addressed here but will be covered in Section 11.2 under performance considerations.

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FINDING:

For those powered-lift airplane examples considered (BR 941 and AWJSRA), takeoff was relatively conventional in nature with the added advantage of being forgiving of pilot abuses.

DISCUSSION:

In general, takeoffs with all engines operating were easy in that they did not require precise pilot actions. The stall limiting flight condition had no practical significance in these cases because acceleration was rapid, regardless of the pilot's actions. The aircraft would not become airborne until a safe speed had been reached.

It was not practical in these cases to distinguish between V_1 and V_R. This was due to the rapid acceleration and short time interval between the two points.

While this implies that it may be convenient to set V_1 equal to V_R when acceleration is very high, we did not look into the performance consequences of such an assumption. Specifically, it could excessively penalize an aircraft with good continued takeoff capability but poor ability to stop.

11.2 STABILITY, CONTROL, AND PERFORMANCE; TAKEOFF

For the takeoff flight phase the emphasis is clearly more on performance than on stability and control although basic attitude control is clearly necessary. The items of prime importance in defining takeoff performance are the takeoff field length, followed by the climb profile relative to terrain. As mentioned previously, these considerations are really no different for powered-lift aircraft than for conventional aircraft. This is largely reflected in the following finding.

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FINDING:

Conventional methods of describing takeoff performance were considered adequate for the powered-lift simulations considered.

DISCUSSION:

This is a general statement meant to reflect the lack of any peculiar features for powered-lift airplanes during the takeoff flight phase. Concepts such as field length, balanced field length, and climb gradient all apply to powered-lift vehicles in the same way they do for conventional aircraft. This finding also extends into the condition of propulsion system failure. In fact, propulsion system failure is the major element in defining most aspects of takeoff performance. For this reason, most performance details will be handled in the next subsection.

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FINDING:

Actual climb performance as demonstrated by the pilot was measurably less than the theoretical obstacle clearance plane performance.

DISCUSSION:

Climb performance data from the BR 941 simulation were compared with theoretical values, Reference 11. A theoretical obstacle clearance plane was obtained from the γ - V curves for the aircraft. This was compared with a measurement of the minimum obstacle clearance plane achieved on each takeoff. In calm air the difference was relatively small. In moderate turbulence, σ_{u_g} = 0.91 m/s (3 ft/s), differences of 1 deg were not uncommon and on one takeoff a value of 1.4 deg less was measured.

These data provide some indication of the required margins between real obstacles and the theoretical performance of the aircraft.

11.3 PROPULSION SYSTEM FAILURE DURING TAKEOFF

Propulsion system failure during takeoff can have a potentially broad ranging effect depending upon the design of the airplane. The two vehicles studied probably span the range of expected powered-lift STOL designs in terms of the impact of propulsion system failures on takeoff. The BR 941 (Reference 11), with its propeller cross-shafting, completely lacked any lateral-directional asymmetry following a propulsion system failure. Also, because it was a four-engine vehicle, the net thrust loss was relatively small. The AWJSRA airplane (Reference 12), on the other hand, had substantial lateral-directional asymmetries following propulsion system failure and because it was a twin-engine aircraft, suffered a large net loss in takeoff thrust-to-weight ratio. It should be noted that the lateral-directional asymmetry in the AWJSRA was minimized somewhat by the cold-thrust cross-ducting, but even so, a substantial amount of lateral-directional control and pilot effort was necessary to overcome this asymmetry.

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FINDING:

The only observed impact of propulsion system failure on limiting flight conditions was a minimum speed for directional control while on the ground, i.e., V_{MCG}.

DISCUSSION:

Such a finding is highly configuration dependent. The V_{MCG} condition observed here was purely a function of the AWJSRA simulation model. The determining factors were simply lateral thrust moment, aerodynamic directional control, and nose-wheel steering effectiveness. Existing standards adequately address this particular problem.

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FINDING:

The effect of speed at engine failure on takeoff field length was found to be highly configuration dependent.

DISCUSSION:

This observation is, again, based on only the two models considered in this program. As in other areas, however, these models represent the extremes which could be expected from powered-lift airplanes. An experimentally determined plot of takeoff distance to 11 m (35 ft) versus speed at engine failure is shown in Figure 11-1. The effect is nearly negligible in the case of the BR 941 while it is substantial for the AWJSRA. The reasons are worth noting. The AWJSRA, a twin-engine airplane, experienced a simple 50% loss in thrust at the point of engine failure, thus takeoff field length was correspondingly strongly affected. The relative field length effect is likely similar to any other twin-engine jet airplane. The BR 941, on the other hand, was driven by four cross-shafted propellers. Without considering cross-shafting, use of four engines represents only 25% loss in thrust at the point of engine failure. Because of the cross-shafted propellers, loss of power at low speeds involves transition to a more efficient propeller operating point, with or without propeller pitch governing. Any increase in propeller efficiency means that percent thrust loss is not as great as percent power loss. In the case of this powered-lift airplane model, these effects combine to make takeoff field length nearly independent of the speed at propulsion system failure.

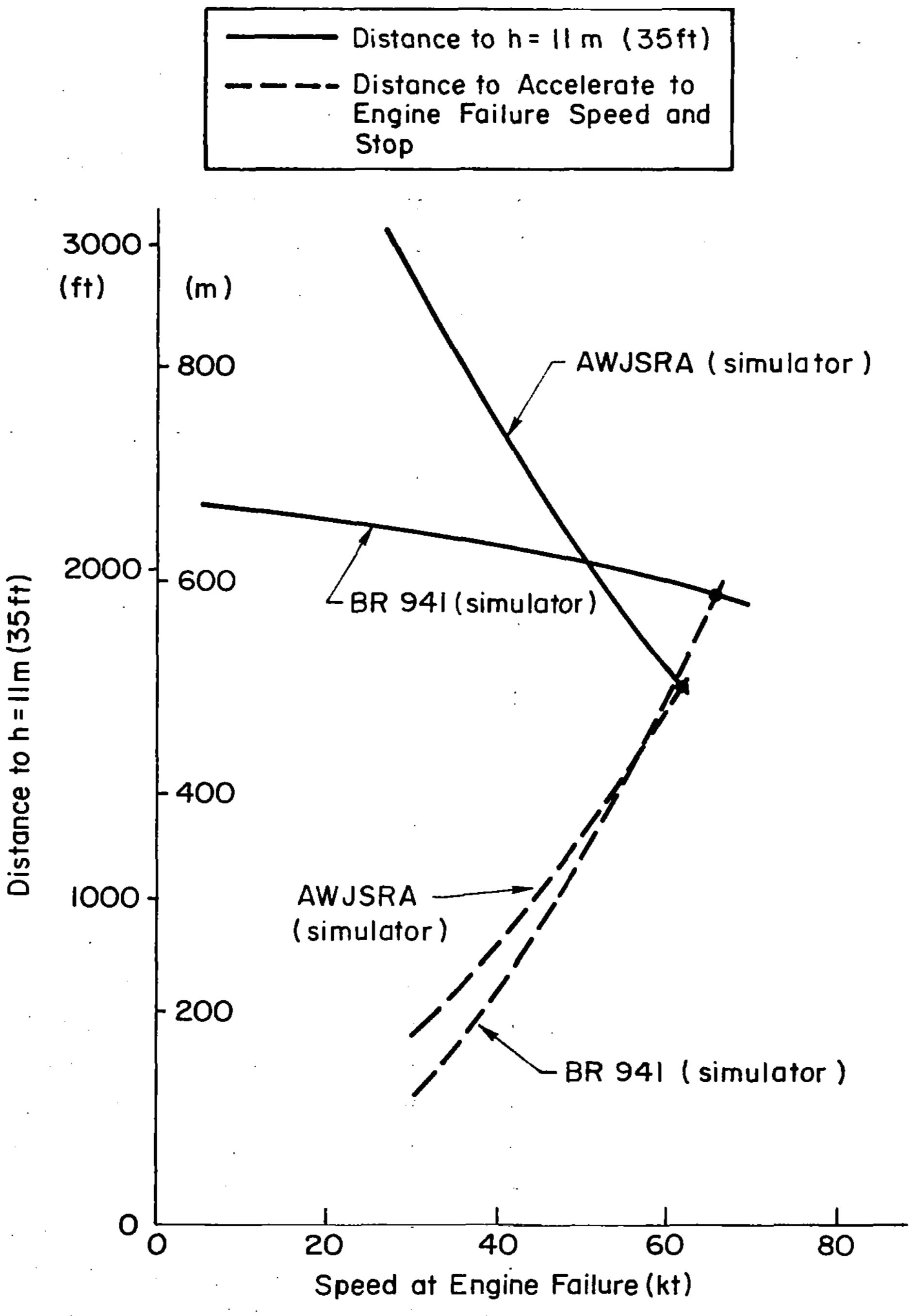

Figure 11-1: Effect of Engine Failure on Takeoff and Stopping Distance

While this effect may vary significantly for different powered-lift aircraft, existing airworthiness standards should adequately cover the situation.

SECTION 12

CONCLUSIONS AND RECOMMENDATIONS

The following is a summary of conclusions and recommendations. Details are omitted but can be found in the body of the report by the specific page references given in parenthesis. The purpose of this section is to indicate the current status of airworthiness criteria in various areas and recommendations for additional work. It is not to reiterate specific findings.

This section is organized similarly to the body of this report. Each subsection corresponds to a respective report section. For example, Subsection 12.5 corresponds to Section 5. Subsection 12.1 is the only exception and deals with subjects of a general nature.

12.1 GENERAL

Powered-lift aircraft constitute a fundamentally different category of airplane from conventional jet transports. At the same time, this category appears less broad than originally thought. The features of powered-lift aircraft which affect airworthiness criteria are not necessarily strong functions of the specific powered-lift concept involved. The main features which powered-lift aircraft generally have in common are:

- High C_L generated by direct or indirect thrust
- Strong effect of both angle of attack and thrust on C_L and C_D
- Strong effect of thrust on $C_{L_{max}}$ and stall speed
- Nearly vertical thrust inclination
- Lift asymmetry from engine failure rather than horizontal thrust asymmetry.

Piloting technique is an important factor in all aspects of powered-lift flight. Longitudinal control tasks, in particular, are likely to require fundamentally different techniques than those used in conventional jet transports. Because of the important role of piloting technique it must be directly addressed in the establishment of airworthiness criteria.

12.2 EXPERIMENTAL APPROACH

A moving-base simulator is an effective tool for exploring and developing airworthiness criteria. In this program, a number of criteria forms and numerical limits were established. These limits, however, may tend to be conservative because of simulator limitations which may have made some tasks somewhat more difficult than actual flight; or optimistic because of pilots' perception of the task as being more difficult in the simulator than in flight (Reference 10). Visual and motion cues were not as good as in flight. Subject pilots generally felt the turbulence model was overly severe.

The question of the turbulence model realism is probably the most serious one. While it cannot be resolved at this time, the following results are pertinent.

- A short simulator experiment conducted during this program (page 15) indicated that the turbulence model used is at least as good as other available models.
- Reference 10 describes a related powered-lift program which utilized both ground-based simulation and the Princeton Variable Stability Navion airplane. Both used the same turbulence model employed in this program. Even with the basic Navion airplane, the evaluation pilot considered the model a valid representation of real-world turbulence.
- Reference 47 describes another FSAA simulation program which used this same turbulence model. That program simulated a current short-haul jet transport. Pilots with flight experience in the aircraft felt the turbulence model was somewhat too severe.

Additional research to resolve this question is clearly needed. There is also a need for research on the potential problems of wind shear encounters in powered-lift aircraft. A preliminary simulator investigation of this problem is reported in Reference 21. The results of that program indicate that powered-lift aircraft may have more difficulty in wind shears than conventional aircraft.

While criteria developed from simulation may be either conservative or optimistic, the simulation did provide basic insights and understanding of factors involved in flying powered-lift aircraft. In turn, this led to more effective analyses.

12.3 LIMITING FLIGHT CONDITIONS

The main limiting flight conditions for powered-lift aircraft, like conventional aircraft, are connected with low speed, high angle of attack flight in the vicinity of aerodynamic stall. While it can be defined in terms of speed or angle of attack, a limiting flight condition should be based on the occurrence of certain conditions. Limiting flight conditions are ultimately defined by certain "hard" conditions of a catastrophic nature such as loss of control, abrupt loss of lift, etc. Aerodynamic stall itself, though, is not necessarily in this category. Where it is not a catastrophic event then aerodynamic stall should be considered as a "soft" limiting flight condition (page 31).

The influence of throttle on limiting flight conditions of powered-lift aircraft is important not only in the definition of limiting flight conditions but also in the potential rate of onset. The most critical aspect is that a rapid throttle reduction produces an almost equally rapid approach to a limiting flight condition (page 37).

The nature of limiting flight conditions for the bare airframe of powered-lift aircraft was reasonably well explored in this program but there is still the need to form a more precise quantitative description of the components of limiting flight conditions, i.e., the relative effect of maneuvering, pilot abuse, and disturbances. Finally, the matter of limiting flight conditions for heavily augmented* vehicles remains relatively unexplored and should be investigated.

12.4 SAFETY MARGINS

There are two types of limiting flight conditions, hard and soft, therefore, there are two general kinds of safety margins. Both can be speed and angle of attack related. These can take on several forms depending on the nature of protection, i.e., for maneuvering, pilot abuse, or atmospheric disturbances. Limits were established for a variety of conditions (pages 69 through 80). They were based not only on simulation but also on some flight test data.

The validity of the proposed safety margins depends on the realism of the maneuvers, abuses, and disturbances. The realism was probably good with regard to the first item, but for the latter two there is some doubt. Again, there are special problems regarding heavily augmented vehicles.

It is recommended that further study be made of safety margin requirements involving pilot abuses and atmospheric disturbances and that this be based on suitable flight test data. Further, safety margin requirements for heavily augmented vehicles should be investigated.

12.5 LONGITUDINAL STABILITY, CONTROL, and PERFORMANCE; APPROACH

In the approach phase there is heavy emphasis on flight path/flight reference control. Piloting technique is an important determining factor in the nature of flight path/flight reference (and pitch attitude) control requirements.

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* Heavily augmented refers to control systems which substantially alter the basic lift and drag characteristics and therefore the basic flight path dynamics.

For powered-lift aircraft, a STOL technique is most likely. This requires that attention be given to the dynamic response of flight path to throttle and diminishes the role of pitch attitude control. A combination of airframe and propulsion dynamics is involved. The main determining factors in powered-lift flight path dynamics are a high C_L and a near-vertical effective thrust inclination (page 95).

Tentative airworthiness criteria for flight path control were determined and included the areas of dynamic response (page 117) and control power (pages 124, 130). A suitable criterion for cross coupling, an important factor, was not found (page 137).

The nature of flight reference control was studied and found to be dependent upon the specific mechanization of the flight reference. While a number of aspects were recognized (page 140), no criteria were defined.

It is recommended that the tentative criteria concepts and numerical values given here be confirmed or further refined by flight test. In so doing, special attention should be given to atmospheric disturbances, especially wind shears. There is need for further simulation and analytical study of flight path/flight reference cross coupling and flight reference control. Regarding the criteria proposed, further study should be given to the problems of heavily augmented vehicles and effects of improved displays, such as flight directors.

12.6 LONGITUDINAL STABILITY, CONTROL, AND PERFORMANCE; LANDING

The flare and landing involves an extension of the same ideas developed for the approach phase. The choice of piloting technique used in the flare is a major factor. Important airplane dynamics are mainly related to flight path, while flight reference control is of considerably lesser importance.

In addition to the conventional flare technique involving use of pitch attitude, one must also consider use of throttle as a primary flare control (page 149) or, under certain conditions, a combination of throttle and pitch attitude (page 150). For powered-lift aircraft, flaring with pitch attitude is similar to conventional aircraft but larger excursions of pitch attitude, airspeed, and angle of attack are involved. The two important factors are n_{z_α} and Z_w (heave damping) but one cannot separate the effects based on existing data (page 163). Use of throttle to flare can be treated just as in the approach phase. A tentative response criterion was established.

A landing demonstration with specified abuses was considered to take care of certain features not otherwise easily measured or quantified such as short-term control power and ground effect. This also addresses safety margins. A calm-air demonstration with abuses was found to be feasible (page 167) as a reasonable substitute for demonstrating landings in high turbulence.

Future efforts should be directed at airframe qualities required to flare with pitch attitude, especially regarding the separate roles of n_{z_α} and heave damping. In addition, flight tests should be performed to provide better numerical definition of the specified abuses to be used in the landing demonstration and to verify the response criterion for flare with power.

12.7 LATERAL-DIRECTIONAL STABILITY AND CONTROL; APPROACH AND LANDING

Lateral-directional stability and control for powered-lift aircraft is not fundamentally different from conventional aircraft, but basic aircraft characteristics tend to be worse. The common powered-lift problems in approximate order of importance are:

- Poor turn coordination
- Rapid spiral divergence
- Low roll damping
- Low dutch roll frequency and damping.

There is no reason to suspect that piloting problems are different or that there is need for different criteria. At the same time, it should be recognized that turn coordination or heading control criteria are not well established even for conventional aircraft.

12.8 PROPULSION SYSTEM FAILURE; APPROACH AND LANDING

A propulsion system failure in a powered-lift aircraft has important features which are fundamentally different from those in conventional aircraft. These derive largely from the use of powered-lift so that a failure produces an asymmetric vertical, rather than horizontal, force. This can lead to a variety of significant piloting problems which are configuration and control system dependent.

Recognition of the failure seems to take longer than for conventional aircraft, possibly because the large lateral acceleration cue is missing (page 182). The most immediate effect of a failure is an increase in sink rate but the pilot may not notice this initially. His best cue is the roll disturbance. The appropriate pilot responses are also unconventional — large wheel input with relatively little rudder (possibly of either sign) (page 186).

One potentially serious aspect of the problem, not treated in this program, is that the failure characteristics depend on the aircraft configuration. During most of a flight, powered-lift is not being used and propulsion failures should be similar to those in conventional aircraft. During approach, though, propulsion failures would involve the special features of powered lift. Having two distinctly different sets of failure characteristics and required pilot responses in the same aircraft could have an adverse effect.

For failures which occur at low altitude, the most critical factor is the time to reach a flight condition which has adequate safety margins, stability and control, and performance. The results of this program suggest a configuration change may be allowable provided it can be done quickly and simply, e.g., using a switch on the throttle to retract spoilers.

Powered lift also has an impact on the problems during a continued approach with a propulsion system failure. Specific problems can include decreased flight path control power, possible saturation of augmentation systems, decreased lateral control power, and use of a different operating point (page 203). It was found that the magnitude of sideslip had a significant effect on all of these problems.

The decreased flight path control power is probably the most significant problem (page 213). A propulsion system failure will certainly reduce the upward capability, perhaps to an inadequate level. Adequate upward performance might be restored with a configuration change but that would probably reduce the downward capability. Criteria for flight path control power after a propulsion system failure were not developed during this program but they could be a dominant design constraint.

This program was successful in exploring qualitative aspects of propulsion system failure problems. Further work is required to develop quantitative criteria. This should be considered of prime importance because of the fundamental differences from conventional aircraft and the potential impacts on aircraft design.

12.9 AUGMENTATION SYSTEMS FAILURE

This area was not directly addressed in this program but is potentially important because of the greater likelihood of complex augmentation systems in powered-lift aircraft. A number of problems are discussed in this report. The most critical of these is considered to be an augmentation system failure requiring a change in the basic piloting technique which is possible in a vehicle employing highly effective lift and drag augmentation. While there are no directly applicable data, the results of Reference 21 may indicate a potentially serious problem (page 222).

It is recommended that simulation be used to explore the subject of augmentation systems failure with regard to developing airworthiness criteria.

12.10 GO-AROUND

This program included a brief study of the go-around maneuver for two specific powered-lift vehicles. Because of design differences between these two vehicles, several important characteristics were revealed. It was found that the main aspects of the go-around phase consist of piloting procedure required, vertical path control and performance, and propulsion failure complications.

As a result of the above, the main feature of powered-lift airplanes was found to be the possibly large loss in altitude following go-around initiation, especially with a propulsion system failure. This can be addressed most directly by a criterion limiting altitude loss which would be demonstrated in flight.

12.11 TAKEOFF

Takeoff simulations were done in this program for two fundamentally different powered-lift vehicles. The only significant difference from conventional aircraft was the more rapid acceleration. Because of the higher thrust, takeoff abuses seemed less significant. As a result, some simplification of conventional standards may be possible, such as combining V_1 and V_R. Takeoffs accomplished by vectoring thrust were not investigated here but may involve a significant departure from conventional aircraft features.

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APPENDIX A

POWERED-LIFT FLIGHT PATH DYNAMICS

Longitudinal flight path dynamics can be viewed in a number of ways. Each has certain advantages, but all are interrelated. We will show this in the following manner:

- Non-dimensional lift/drag relationships
- Basic set of parameters, preliminary to stability derivatives
- Dimensional stability and control derivatives
- Simplified equations of motion to describe path dynamics
- Simplified transfer functions.

As each of the above is developed we will point out the influence of powered lift, the comparison with conventional aircraft, and the effect of various powered-lift concepts. Useful approximations will be given where possible.

A.1 NON-DIMENSIONAL LIFT-DRAG APPROXIMATIONS

First it is useful to consider some features of a pure jet flap. Actually, there is wide application to most powered-lift concepts such as EBF, augmentor wing, IBF, USB, and even deflected slipstream propeller driven aircraft. The concept of a pure jet flap is at least useful for revealing trends and general features.

Reference A-1 develops the theory for an ideal two dimensional jet flap and Reference A-2 extends this to a three dimensional wing. Important relationships are given below.

Slope of Section Lift Coefficient with α:

$$C_{\ell_\alpha} \doteq 2\pi\left(1 + .151\, C_J^{1/2} + .219\, C_J\right)$$

Three Dimensional Lift Correction:

$$\frac{C_{L_\alpha}}{C_{\ell_\alpha}} \doteq \frac{AR + 2/\pi\, C_J}{AR + 2/\pi\, C_{\ell_\alpha} - 2} \doteq \frac{AR + .637\, C_J}{AR + 2 + .604\, C_J^{1/2} + .876\, C_J}$$

Induced Drag for Elliptical Wing:

$$C_{D_i} \doteq \frac{C_L^{\,2}}{\pi\, AR + 2\, C_J}$$

or

$$\frac{\partial C_D}{\partial C_L} = \frac{2C_L}{\pi\, AR + 2\, C_J}$$

Recall the respective values for normal wing are:

$$C_{\ell_\alpha} \doteq 2\pi$$

$$\frac{C_{L_\alpha}}{C_{\ell_\alpha}} \doteq \frac{AR}{AR + 2}$$

$$C_{D_i} \doteq \frac{C_L^{\,2}}{\pi\, AR}$$

The above equations can be used to derive an expression for $C_{L_\alpha} / \left(C_{L_\alpha}\right)_o$ where the subscript zero refers to a non-jet-flap wing. This result is plotted in Figure A-1 which shows that jet flap effects can be approximated by

$$C_{L_\alpha} \doteq \left(C_{L_\alpha}\right)_o \left(1 + \frac{C_J}{4}\right)$$

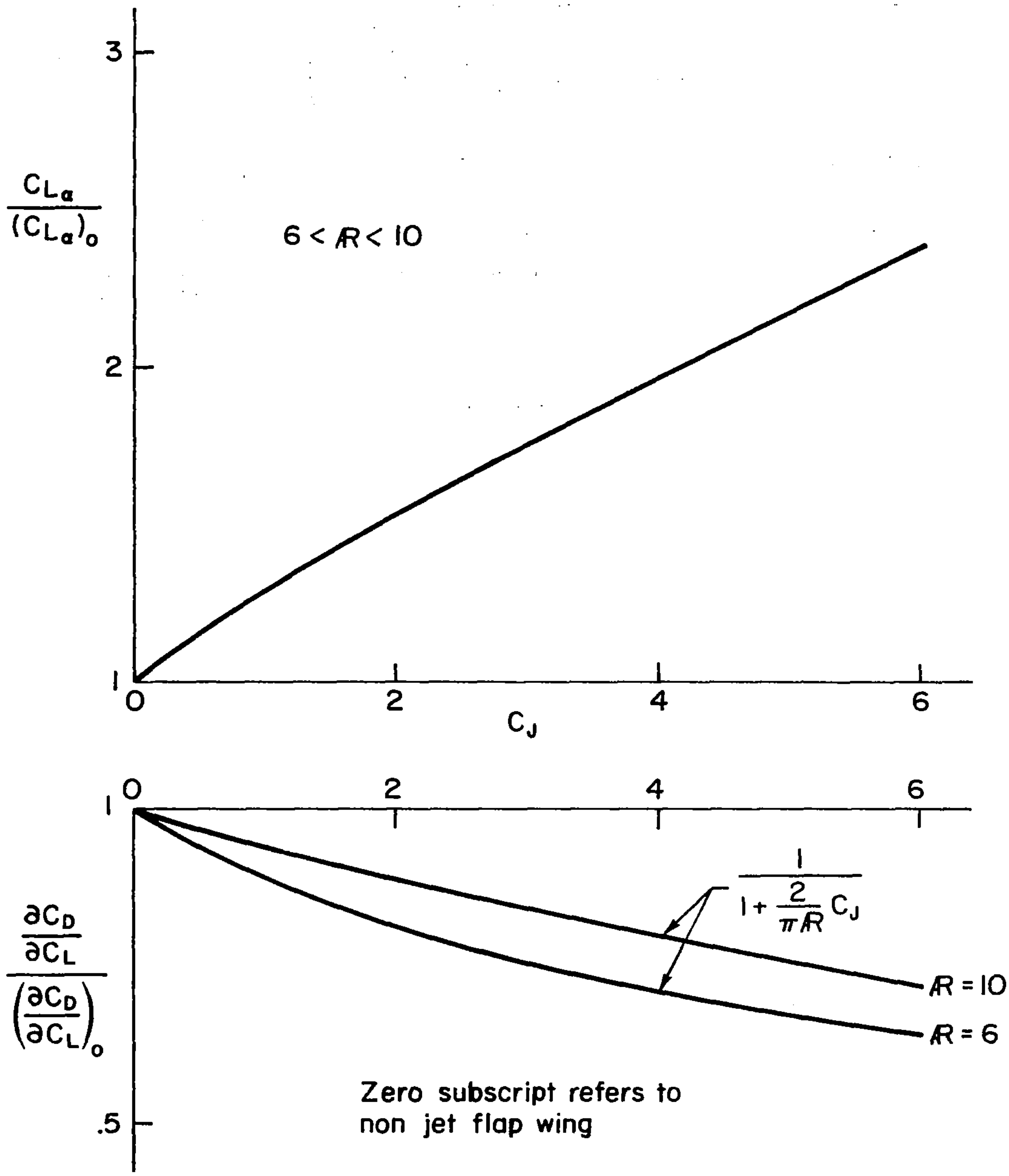


Figure A-1: Effect of Jet Flap Blowing on Lift Curve Slope and Induced Drag

The effects on induced drag are readily available from the above equations.

$$\frac{\partial C_D}{\partial C_L} \doteq \left(\frac{\partial C_D}{\partial C_L}\right)_o \frac{1}{1 + \frac{2C_J}{\pi \mathit{AR}}} \doteq \left(\frac{\partial C_D}{\partial C_L}\right)_o$$

This result is also plotted in Figure A-1.

A.2 BASIC FLIGHT PATH PARAMETERS

The following set of parameters are offered as aid in simplifying certain aspects of flight path dynamics. They are also useful in estimating stability derivatives. Most are familiar concepts, but one is newly defined. An important point is that none of these parameters uniquely determine path dynamics, i.e., there are no universal parameters. Taken together in various combinations, though, they form all the basic flight path relationships in terms of stability derivatives, transfer functions, etc. Their main value is that they are easily computed or estimated. Each tends to be reasonably invariant for given airplane category, e.g., conventional jet transport, powered lift, helicopter, etc.

These parameters are:

- g/V
- n_{z_α}
- n_{x_α}
- θ_T
- η_p

Following is a discussion of each, their meaning, approximations, and a comparison of powered-lift versus conventional aircraft.

g/V is just speed dependent but it is so expressed because it appears frequently in dimensional derivatives and transfer functions. Its dimension of frequency has significance when combined with the other non-dimensional parameters to follow. g/V is related to basic phugoid frequency, ω_p, which is approximately $\sqrt{2} \cdot g/V$. This relation, however, is considered unimportant for this discussion.

n_{z_α} is simply incremental normal acceleration due to $\Delta\alpha$. It has been widely used in handling qualities literature as a basic parameter. Its most important interpretation here is as the high frequency gain for the flight path/pitch transfer function. It is also closely related to heave damping:

$$Z_w \doteq -\frac{g}{V} n_{z_\alpha}$$

n_{z_α}, as used here, is:

$$n_{z_\alpha} = \frac{C_{L_\alpha}}{C_L} \cos \gamma_o$$

where C_{L_α} is a "trimmed" value, i.e., elevator varied to maintain zero pitching moment

The $\cos \gamma_o$ effect can be neglected for normal flight path angles, so that

$$n_{z_\alpha} \doteq \frac{C_{L_\alpha}}{C_L}$$

One can estimate n_{z_α} for powered-lift aircraft using the previous jet flap relations, i.e.,

$$C_{L_\alpha} \doteq \left(C_{L_\alpha}\right)_o \left(1 + \frac{C_J}{4}\right)$$

If we assume $\left(C_{L_\alpha}\right)_o \doteq 6$, valid for most conventional jet transports, and let $C_J = T_b/W \cdot C_L$, where T_b is thrust used for blowing the wing, then

$$n_{z_\alpha} \doteq \frac{6}{C_L} + 1.5\ T_b/W$$

Hence, n_{z_α} is mainly a strong function of C_L but increased somewhat with blowing. A plot of n_{z_α} versus C_L for various aircraft is given in Figure A-2. Most examples of powered-lift aircraft occur in a C_L range of 3 to 4, thus, $1.5 < n_{z_\alpha} < 2.5$. High C_L's of say 5 to 7 have low n_{z_α} due to C_L alone (approximately $n_{z_\alpha} = 1$) but this is likely to be raised to 1.5 by the blowing effect. Conventional aircraft operating at 1.3 V_s fall in an n_{z_α} range from 3 to 5.

$\underline{n_{x_\alpha}}$ is analogous to n_{z_α}, but tangent to flight path. It is the high frequency speed gain for $\Delta\theta$ or $\Delta\alpha$ inputs. It is comparatively uninteresting, though, because it is relatively invariant between powered-lift and conventional aircraft.

The main physical importance is tied to the ratio between n_{x_α} and n_{z_α}, i.e., the effective inclination of θ control (the counterpart to θ_T for throttle). We define this angle as θ_A.

$$\theta_A = \arctan \frac{n_{z_\alpha}}{-n_{x_\alpha}}$$

n_{x_α} is primarily a function of aspect ratio because it is tied to induced drag

$$n_{x_\alpha} = \frac{C_{D_\alpha}}{C_L} = \frac{C_{L_\alpha}}{C_L}\frac{\partial C_D}{\partial C_L} = n_{z_\alpha}\frac{\partial C_D}{\partial C_L}$$

From earlier approximations we can derive:

$$n_{x_\alpha} \doteq \left(n_{x_\alpha}\right)_o \frac{\left(1 + \frac{C_J}{4}\right)}{\left(1 + \frac{2C_J}{\pi A\!\!\!/R}\right)}$$

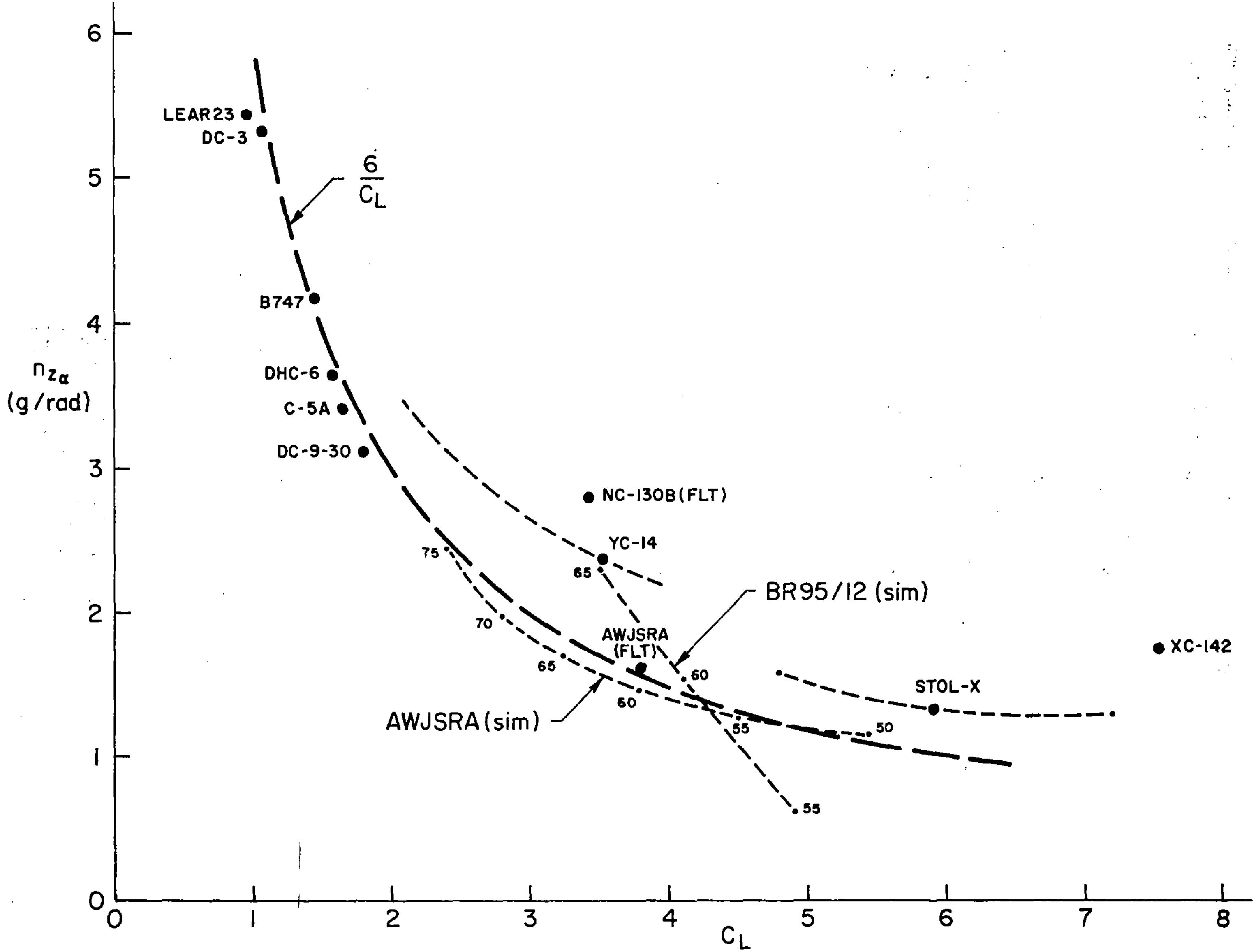


Figure A-2: Typical n_{z_α} Values

Thus we see that blowing has a relatively small effect on n_{x_α}. A typical value for both conventional and powered-lift aircraft is

$$n_{x_\alpha} \doteq 0.6 \text{ g/rad}$$

The effective inclination can be approximated by:

$$\theta_A = \arctan \frac{1}{-\frac{\partial C_D}{\partial C_L}} = \frac{\pi}{2} + \arctan \frac{\partial C_D}{\partial C_L}$$

$$\doteq \frac{\pi}{2} + \frac{2C_L}{\pi \mathcal{R} + 2C_J}$$

Thus, the blowing coefficient has a minor effect and a high lift coefficient increases the angle.

$\underline{\theta_T}$ is the effective thrust angle, i.e.,

$$\tan \theta_T \triangleq \frac{\partial \text{Lift}}{\partial \delta_T} \Bigg/ \left(- \frac{\partial \text{Drag}}{\partial \delta_T}\right)$$

θ_T indicates the relative lift and drag increments for a throttle change. It has a strong impact on the choice of piloting technique. It is one of the most distinguishing factors between powered-lift and conventional aircraft.

For a conventional aircraft, θ_T is nearly horizontal, $\theta_T < 15$ deg. Powered-lift aircraft have a strong vertical component, for deflected slipstream, typically 60 deg $< \theta_T <$ 80 deg. For jet flap or vectored nozzle, 70 deg $< \theta_T <$ 95 deg.

A good approximation to θ_T can be obtained from a γ - V plot. To derive this relationship, consider a small throttle perturbation. If pitch attitude is also changed to hold constant angle of attack, the steady state lift and drag perturbations are given by:

$$\Delta \text{Lift} = -W \sin \gamma_o \Delta\gamma \doteq \frac{2\text{Lift}}{V} \Delta V + \frac{\partial \text{Lift}}{\partial \delta_T} \delta_T$$

$$\Delta \text{Drag} = -W \cos \gamma_o \Delta\gamma \doteq \frac{2\text{Drag}}{V} \Delta V + \frac{\partial \text{Drag}}{\partial \delta_T} \delta_T$$

Combining these equations with the θ_T definition and the trim relationships

$$\text{Lift} = W \cos \gamma_o$$

$$\text{Drag} = -W \sin \gamma_o$$

give the desired approximation

$$\tan(\gamma_o + \theta_T) \doteq -\frac{2}{V}\left(\frac{\partial V}{\partial \gamma}\right)_\alpha$$

Thus for small θ_T, conventional aircraft, $\left(\frac{\partial V}{\partial \gamma}\right)_\alpha$ will be small — constant α lines will be nearly vertical. For large θ_T, powered-lift aircraft, $\left(\frac{\partial V}{\partial \gamma}\right)_\alpha$ will be large and negative — constant α lines will be nearly horizontal. If constant α contours are not available but constant θ contours are, one can use the following relationship:

$$\left(\frac{\partial V}{\partial \gamma}\right)_\alpha = \left(\frac{\partial V}{\partial \gamma}\right)_\theta + \left(\frac{\partial V}{\partial \theta}\right)_\gamma$$

This kind of relationship will be more broadly developed in Subsection A.5.

$\underline{\eta_p}$ is a newly defined parameter which indicates the proportion of propulsion supplied lift to total lift. It should not be confused with θ_T although there is a strong relationship between the two. The direct impact of η_p is on the vertical acceleration due to a horizontal gust. This is characterized as the stability derivative Z_u. η_p is really defined as a function of Z_u.

$$\eta_p \triangleq 1 + \frac{V Z_u}{2g \cos \gamma_o}$$

η_p equal to zero characterizes conventional aircraft. If η_p is unity then the aircraft is totally propulsion supported, i.e., there is no aerodynamic lift. A hovering vehicle would have $\eta_p \doteq 1$.

The powered-lift factor can easily be related to basic aerodynamic properties. In general, the derivative Z_u can be expressed as:

$$Z_u \doteq -\frac{2g}{V} \cos \gamma_o \left(1 + \frac{V}{2C_L} \frac{\partial C_L}{\partial u}\right)$$

where C_L is assumed to include all thrust effects and forces.
Therefore:

$$\eta_p \doteq \frac{-V}{2C_L}\left(\frac{\partial C_L}{\partial u}\right)$$

The lift coefficient is normally presented as a function of blowing coefficient, C_J. Therefore, η_p can be written as:

$$\eta_p \doteq \frac{-V}{2C_L} \frac{\partial C_J}{\partial u} \frac{\partial C_L}{\partial C_J}$$

Finally, if we assume that thrust is independent of airspeed

$$\frac{\partial C_J}{\partial u} = \frac{-2C_J}{V}$$

and $$\eta_p \doteq \frac{C_J}{C_L} \frac{\partial C_L}{\partial C_J}$$

also, $$\eta_p \doteq \frac{\partial \log C_L}{\partial \log C_J}$$

If lift is a function of α and C_J (or T_c', etc.) then η_p is easily approximated from a plot of log C_L versus log C_J, i.e., η_p is the slope.

An illustration of this is shown in Figure A-3. C_L is plotted versus C_J on a log-log scale for several aircraft. Some are based on wind tunnel data, some on flight test. The value of η_p appears to range from 0.3 to 0.5 for jet flap type aircraft.

If an airplane uses direct vectored thrust the η_p is simply $T_d/W \sin \theta_T$. If there is a combination of jet flap and direct thrust (e.g., AWJSRA) then an effective overall η_p can be obtained from respective components. For example:

$$\eta_{p_1} \triangleq \frac{C_J}{C_L} \frac{\partial C_L}{\partial C_J} \quad \text{(jet flap, i.e. } C_J = \frac{T_b}{S\bar{q}}\text{)}$$

$$\eta_{p_2} \triangleq \frac{T_d}{W} \sin \theta_T \text{ (direct thrust)}$$

Then the effective η_p is related by:

$$(1 - \eta_p) \doteq (1 - \eta_{p_1})(1 - \eta_{p_2})$$

An interesting approximate relationship among η_p, θ_T, and θ_A can be shown using a theoretical jet flap drag approximation.

$$C_D \doteq -rC_J + \frac{C_L^2}{\pi \mathcal{R} + 2C_J} + C_{D_o}$$

taking partial derivative with respect to C_J,

$$\frac{\partial C_D}{\partial C_J} \doteq -r + \frac{2C_L}{\pi \mathcal{R} + 2C_J} \frac{\partial C_L}{\partial C_J} - \frac{2C_L^2}{(\pi \mathcal{R} + 2C_J)^2}$$

where r is the jet flap recovery factor

Recall $$\frac{\partial C_D}{\partial C_L} = \frac{2C_L}{\pi \mathcal{R} + 2C_J} = -\operatorname{ctn} \theta_A$$

and $$\frac{\partial C_L}{\partial C_J} = \eta_p \frac{C_L}{C_J} = \frac{\eta_p}{T_b/W}$$

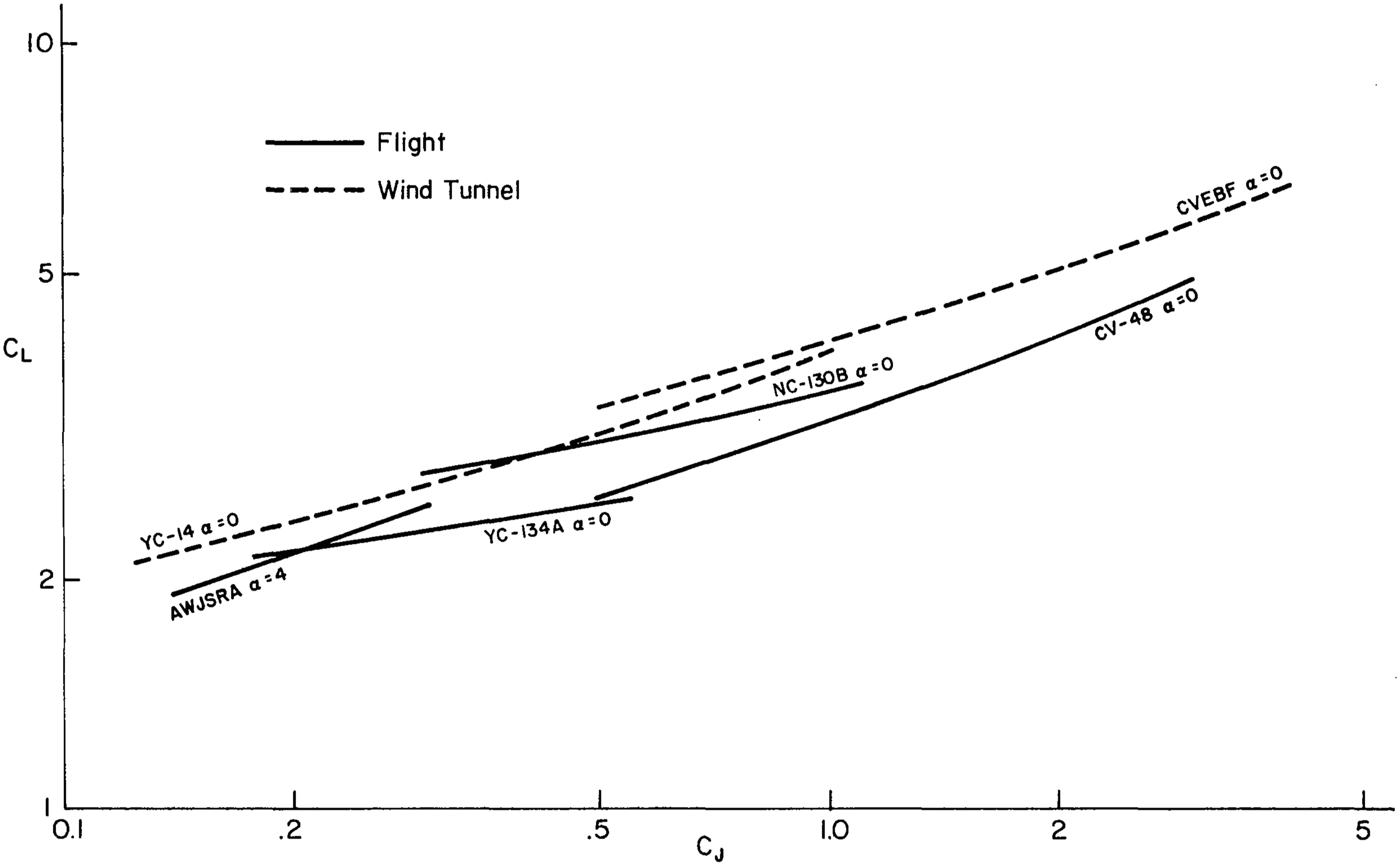


Figure A-3: Examples of C_L versus C_J data

$$\text{thus} \qquad \operatorname{ctn}\theta_T = \frac{-\dfrac{\partial C_D}{\partial C_J}}{\dfrac{\partial C_L}{\partial C_J}} = \frac{r + \frac{1}{2}\operatorname{ctn}^2\theta_A}{\eta_p} \cdot T_b/W + \operatorname{ctn}\theta_A$$

$$\text{or} \qquad \tan\left(\theta_T - \frac{\pi}{2}\right) = \tan\left(\theta_A - \frac{\pi}{2}\right) - \frac{r + \frac{1}{2}\operatorname{ctn}^2\theta_A}{\eta_p} T_b/W$$

$$\text{or} \qquad \theta_T \doteq \theta_A - \frac{r + \frac{1}{2}\operatorname{ctn}^2\theta_A}{\eta_p} T_b/W$$

Thus, for airspeed nearly constant θ_T is approximately proportional to T_b/W. Further, if r is unity and $C_L \ll \pi R$ then:

$$\frac{\partial\,\theta_T}{\partial\,T_b/W} = -\frac{1}{\eta_p}$$

To summarize the parameter relationships above:

- g/V is simply speed dependent.
- n_{z_α} is a strong function of C_L but also feels some powered-lift effect. The latter is significant only at high C_L and strong blowing.
- n_{x_α} is fairly invariant, but is a function of $\mathcal{R}$.
- θ_T without powered lift is small. It is usually about 70 to 95 deg for a jet flap, 60 to 80 deg for a deflected slipstream, and decreases as thrust increases. θ_T tends to be large for a steep flight path angle and high η_p.
- η_p is the main indicator of powered lift. It is nearly zero for conventional aircraft, and about 0.3 to 0.5 for powered-lift aircraft.

The range of these parameters is shown in the plots of Figure A-4 to illustrate the fundamental differences between powered-lift and conventional aircraft.

A.3 DIMENSIONAL STABILITY AND CONTROL DERIVATIVES

The following dimensional stability and control derivatives are useful in computing longitudinal flight path motion. This will be shown in the subsequent equations of motion and transfer functions.

The relationships shown here differ slightly from those defined in Reference A-3. The main difference is that here we assume a zero pitching moment, i.e., θ is constrained. Pitch is considered a control rather than a response variable. Only one derivative is really affected, and that is slight.

The derivatives represent x and z forces due to perturbations of airspeed, vertical speed (angle of attack), pitch attitude, throttle (neglecting thrust lags), and horizontal and vertical gusts. They are defined for a body-fixed stability-axis system in accordance with Figure A-5.

$\underline{X_u}$ is the basic speed damping stability derivative. It is a simple function of basic parameters. Starting with the basic definition,

$$X_u = -\frac{1}{m}\frac{\partial \mathrm{Drag}}{\partial u}$$

It is convenient to assume gross thrust independent of airspeed, but that there is a ram drag effect equal to $\dot{m}_a V$ where $\dot{m}_a$ is the engine air mass flow. Thus:

$$X_u = -\rho\frac{VS}{m}C_D - \frac{\frac{1}{2}\rho V^2 S}{m}\frac{\partial C_D}{\partial u} + \frac{\dot{m}_a}{m}$$

then $$\frac{\partial C_D}{\partial u} = \frac{\partial C_D}{\partial C_J}\frac{\partial C_J}{\partial u}$$

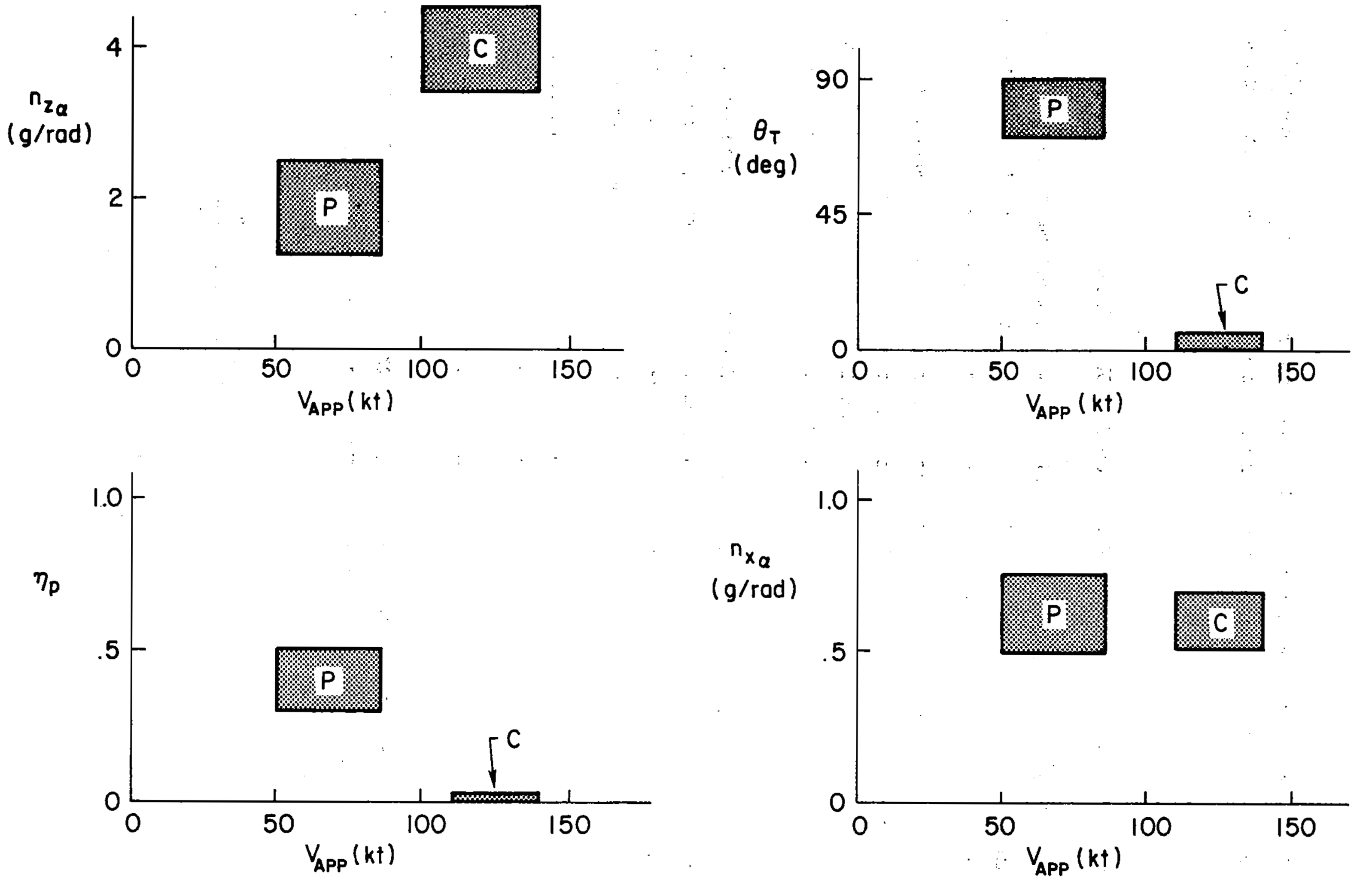
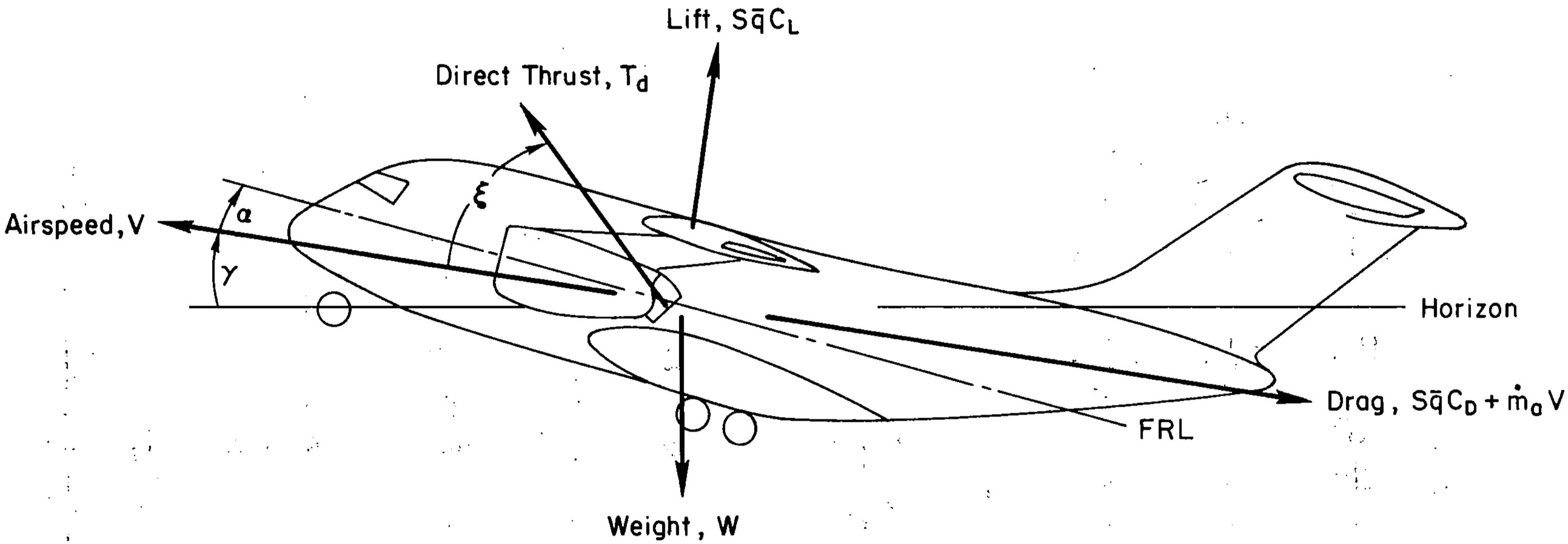


Figure A-4: Comparison of Dominant Features
(Conventional Transport Versus Powered-Lift)



Summation of Forces:

$$0 = -W \sin \gamma - S\bar{q}C_D - \dot{m}_a V + T_d \cos \xi \quad \text{(Along Flight Path)}$$

$$0 = W \cos \gamma - S\bar{q}C_L - T_d \sin \xi \quad \text{(Normal to Flight Path)}$$

Where $C_L, C_D = f(\alpha, C_J)$

and $C_J \triangleq T_b / S\bar{q}$

Figure A-5: Longitudinal Force Diagram

and $$\frac{\partial C_J}{\partial u} = -\frac{2}{V} C_J$$

thus $$X_u = -\frac{\rho VS}{m} C_D + \frac{\rho VS}{m} C_J \frac{\partial C_D}{\partial C_J} + \frac{\dot{m}_a}{m}$$

$$= -\frac{2g}{V} \cos \gamma_o \left(\frac{C_D}{C_L} - \frac{C_J}{C_L} \frac{\partial C_D}{\partial C_J} \right) - \frac{\dot{m}_a}{m}$$

$$= -\frac{2g}{V} \cos \gamma_o \left(\frac{C_D}{C_L} - \frac{C_J}{C_L} \frac{\partial C_L}{\partial C_J} \frac{\partial C_D/\partial C_J}{\partial C_L/\partial C_J} \right) - \frac{\dot{m}_a}{m}$$

but $$\tan \gamma_o = -\frac{C_D}{C_L}$$

and $$\text{ctn } \theta_T = \frac{-\partial C_D/\partial C_J}{\partial C_L/\partial C_J}$$

therefore

$$X_u = -\frac{2g}{V} \cos \gamma_o \left(- \tan \gamma_o + \eta_p \text{ ctn } \theta_T \right) - \frac{\dot{m}_a}{m}$$

For conventional aircraft it is more useful to evaluate X_u as a function of C_D/C_L (where C_D is not a function of δ_T) or

$$X_u = -\frac{2g}{V} \cos \gamma_o \frac{C_D}{C_L} - \frac{\dot{m}_a}{m}$$

$\underline{Z_u}$ is a cross coupling derivative, i.e., it is the specific z-force due to an x-velocity. It is important because it represents the high frequency vertical path disturbance due to a horizontal gust.

$$Z_u = -\frac{1}{m} \frac{\partial \text{Lift}}{\partial u}$$

$$= -\frac{\rho VS}{m} C_L - \frac{\frac{1}{2}\rho V^2 S}{m} \cos \gamma_o \frac{\partial C_L}{\partial u}$$

But $$\frac{\partial C_L}{\partial u} = \frac{\partial C_L}{\partial C_J}\frac{\partial C_J}{\partial u} = -\frac{2}{V} C_J \frac{\partial C_L}{\partial C_J}$$

or $$Z_u = \frac{-\rho VS}{m}\left(C_L - C_J \frac{\partial C_L}{\partial C_J}\right)$$

$$= \frac{-2g}{V} \cos \gamma_o \left(1 - \frac{C_J}{C_L}\frac{\partial C_L}{\partial C_J}\right)$$

$$Z_u = \frac{-2g}{V} \cos \gamma_o \,(1 - \eta_p)$$

Recall that η_p is really defined in terms of Z_u, hence the last expression is exact by definition.

Z_u normally increases for slower speeds but in powered-lift aircraft η_p offsets this. Expected values of Z_u for powered-lift aircraft seem to be roughly comparable to those of conventional aircraft.

$\underline{X_w}$ is the other cross coupling derivative, x-force due to z-velocity.

$$X_w = -\frac{1}{m}\frac{\partial \text{Drag}}{\partial u}$$

$$= \frac{-\rho VS}{2m}\left(C_{D_\alpha} - C_L\right)$$

$$= \frac{-g}{V} \cos \gamma_o \left(\frac{C_{D_\alpha}}{C_L} - 1\right)$$

Recall $$\frac{C_{D_\alpha}}{C_L} = n_{x_\alpha}$$

Therefore, $$X_w \doteq \frac{g}{V}\left(1 - n_{x_\alpha}\right)$$

Because n_{x_α} is relatively invariant, then X_w is primarily a function of g/V or speed.

Z_w is the heave damping stability derivative. It is probably the single most important stability derivative with regard to flight path control regardless of aircraft category.

$$Z_w = -\frac{1}{m}\frac{\partial Lift}{\partial u}$$

$$= \frac{-\rho VS}{2m}\left(C_{L_\alpha} + C_D\right)$$

$$= -\frac{g}{V}\cos\gamma_o\left(\frac{C_{L_\alpha}}{C_L} + \frac{C_D}{C_L}\right)$$

$$= -\frac{g}{V}\cos\gamma_o\left(n_{z_\alpha} - \tan\gamma_o\right)$$

$$Z_w \doteq -\frac{g}{V}n_{z_\alpha}$$

This is the only derivative significantly affected by the assumption of constrained attitude. This enters via C_{L_α} and the symbol $\dagger$ is used to denote "trimmed", i.e.,

$$Z_w^\dagger \triangleq Z_w - \frac{Z_{\delta_e}}{M_{\delta_e}}M_w$$

For most practical purposes, though, this distinction can be ignored.

It is important to recognize that the jet flap augmentation of C_{L_α} for powered-lift aircraft tends to maintain a significant level of heave damping. We can see this if we recall that

$$C_{L_\alpha} \doteq \left(C_{L_\alpha}\right)_o\left(1 + \frac{C_J}{4}\right)$$

or $$C_{L_\alpha} \doteq 6 + 1.5\,C_J$$

thus $$Z_w \doteq \frac{-\rho VS}{2m} (6 + 1.5\ C_J)$$

$$\doteq \frac{2\rho gV}{W/S} - \frac{1.5\ g}{V} T_b/W$$

For conventional aircraft only the first term acts, but in jet flap type powered-lift both are significant. Figure A-6 illustrates the effect. This shows an estimate of Z_w based on T_b/W = zero and 0.5.

$\underline{X_{\delta_T}}$ is the x-force throttle control derivative. It is convenient to normalize this using thrust to weight, i.e., δ_T in units of $\Delta T/W$. Thus, for a jet flap airplane:

$$X_{\delta_T} = -\frac{1}{2} \frac{\rho V^2 S}{m} \frac{\partial C_D}{\partial C_J} \frac{\partial C_J}{\partial T/W}$$

$$= \frac{g\ \eta_p}{T_b/W\ \tan\theta_T}$$

For the case of direct thrust,

$$X_{\delta_T} = g \cos\theta_T$$

$\underline{Z_{\delta_T}}$ is the z-force throttle control derivative and is expressed similarly to X_{δ_T}:

$$Z_{\delta_T} = -\frac{g\ \eta_p}{T_b/W} \text{ for powered lift}$$

$$Z_{\delta_T} = -g \sin\theta_T \text{ for direct thrust}$$

Note that $\frac{-Z_{\delta_T}}{X_{\delta_T}} \triangleq \tan\theta_T$

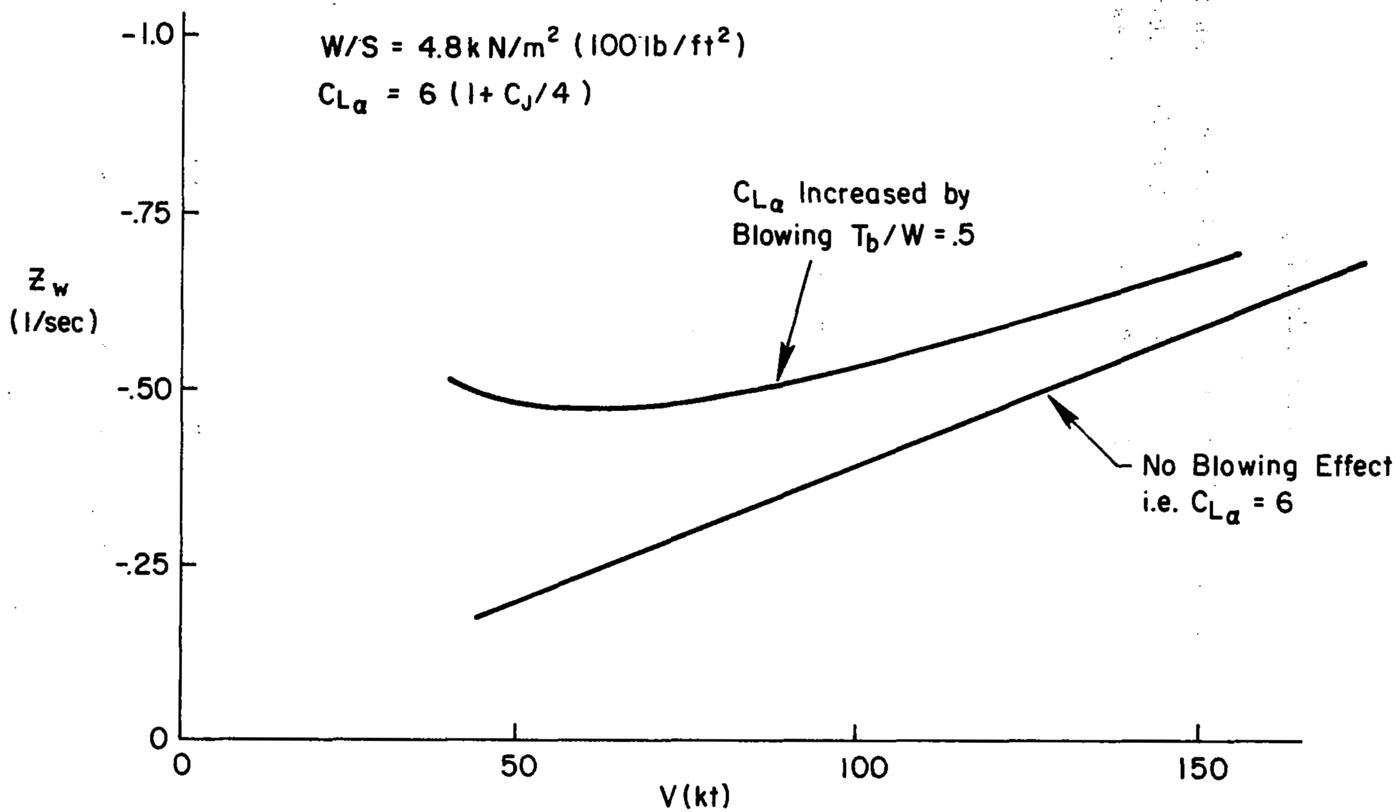


Figure A-6: Heave Damping Approximation

The dimensional derivatives can thus be summarized:

$$X_u \doteq -\frac{2g}{V}(-\tan\gamma_o + \eta_p \operatorname{ctn}\theta_T) - \frac{\dot{m}_a}{m}$$

$$\left(\text{or } -\frac{2g}{V}\frac{C_D}{C_L} - \frac{\dot{m}_a}{m} \text{ if } C_D \text{ is not a function of } C_J\right)$$

$$Z_u = -\frac{2g}{V}(1 - \eta_p)$$

$$X_w = \frac{X_\alpha}{V} \doteq \frac{g}{V}(1 - n_{x_\alpha})$$

$$Z_w^\dagger = \frac{Z_\alpha^\dagger}{V} \doteq -\frac{g}{V} n_{z_\alpha}$$

$$X_{\delta_T} \doteq \frac{g\,\eta_p}{T/W \tan\theta_T}$$

$$(\text{or } g\cos\theta_T \text{ if no jet flap effect})$$

$$Z_{\delta_T} \doteq -\frac{g\,\eta_p}{T/W}$$

$$(\text{or } -g\sin\theta_T \text{ if no jet flap effect})$$

A.4 FLIGHT PATH EQUATIONS OF MOTION

The foregoing stability and control derivatives can be used in the following convenient equation of motion scheme. It is not an exact representation, but if the $Z_w^\dagger$ distinction is recognized there is negligible error. These equations of motion (shown in Figure A-7) have two degrees of freedom with variables speed, u, and flight path excursion, d. The control variables are pitch attitude, θ, and throttle, δ_T. Disturbance

BASIC EQUATIONS

$$\begin{Bmatrix} \frac{1}{m}\sum X \\ -\frac{1}{m}\sum Z \end{Bmatrix} = \begin{Bmatrix} \dot{u} \\ \ddot{d} \end{Bmatrix} = \begin{bmatrix} X_u & -X_w \\ -Z_u & Z_w^{\dagger} \end{bmatrix} \begin{Bmatrix} u \\ \dot{d} \end{Bmatrix} + \begin{bmatrix} (X_\alpha - g\cos\gamma_o) & X_{\delta_T} \\ (-Z_\alpha^{\dagger} + g\sin\gamma_o) & -Z_{\delta_T} \end{bmatrix} \begin{Bmatrix} \theta \\ \delta_T \end{Bmatrix} + \begin{bmatrix} X_u & X_w \\ -Z_u & -Z_w^{\dagger} \end{bmatrix} \begin{Bmatrix} u_g \\ w_g \end{Bmatrix}$$

AUXILIARY RELATIONSHIPS

$u_a = u + u_g$ (airspeed)

$w_a = V\alpha_a = w + w_g$ (aerodynamic velocity normal to flight path, angle of attack)

$w = V\alpha = V\theta - \dot{d}$ (inertial velocity normal to flight path)

$\gamma = \dot{d}/V$ (inertial flight path angle)

$\dot{h} = \dot{d}\cos\gamma_o + u\sin\gamma_o$ (inertial vertical velocity, altitude rate)

Figure A-7: Simplified Flight Path Equations of Motion
Body Fixed Stability Axis System

variables are horizontal gust, u_g (positive for headwind) and vertical gust, w_g (positive for an updraft).

The transfer function denominator is given by:

$$\Delta = s^2 - (X_u + Z_w^\dagger)\, s + X_u Z_w^\dagger - X_w Z_u$$

$$= \left(s + \frac{1}{T_{\theta_1}}\right)\left(s + \frac{1}{T_{\theta_2}}\right)$$

The numerators for control inputs are:

$$N_\theta^u = (X_\alpha - g \cos \gamma_o)\, s + g\,(Z_w^\dagger \cos \gamma_o - X_w \sin \gamma_o)$$

$$= (X_\alpha - g \cos \gamma_o)\left(s + \frac{1}{T_{u_1}}\right)$$

$$N_\theta^{\dot{d}} = (-Z_w^\dagger + g \sin \gamma_o)\, s + X_u Z_\alpha^\dagger - X_\alpha Z_u \; + \; g\,(Z_u \cos \gamma_o - X_u \sin \gamma_o)$$

$$= (-Z_\alpha^\dagger + g \sin \gamma_o)\left(s + \frac{1}{T_{\gamma_1}}\right)$$

$$N_{\delta_T}^u = X_{\delta_T} s - X_{\delta_T} Z_w^\dagger + X_w Z_{\delta_T}$$

$$= X_{\delta_T}\left(s + \frac{1}{T_{u_\theta}}\right)$$

$$N_{\delta_T}^{\dot{d}} = -Z_{\delta_T}\, s + X_u Z_{\delta_T} - X_{\delta_T} Z_u$$

$$= -Z_{\delta_T}\left(s + \frac{1}{T_{\gamma_\theta}}\right)$$

The numerators for gust inputs are:

$$N^{u}_{u_g} = X_u s - X_u Z^{\dagger}_w + X_w Z_u$$

$$N^{u_a}_{u_g} = s (s - Z^{\dagger}_w)$$

$$N^{\dot{d}}_{u_g} = -Z_u s$$

$$N^{u}_{w_g} = X_w s$$

$$N^{\dot{d}}_{w_g} = -Z^{\dagger}_w s + X_u Z^{\dagger}_w - X_w Z_u$$

It is frequently convenient to further simplify these equations by assuming γ_o small, making u_a a variable, dropping the w_g term, and using horizontal gust rate (or shear) $\dot{u}_g$, thus:

$$\begin{Bmatrix} \dot{u}_a \\ \ddot{d} \end{Bmatrix} = \begin{bmatrix} X_u & -X_w \\ -Z_u & Z^{\dagger}_w \end{bmatrix} \begin{Bmatrix} u_a \\ \dot{d} \end{Bmatrix} + \begin{bmatrix} X_\alpha - g & X_{\delta_T} \\ -Z^{\dagger}_\alpha & -Z_{\delta_T} \end{bmatrix} \begin{Bmatrix} \theta \\ \delta_T \end{Bmatrix} + \begin{bmatrix} 1 \\ 0 \end{bmatrix} \left\{ \dot{u}_g \right\}$$

This is a highly useful compact form including all the variables important to the approach and landing situation.

We can express the various transfer function roots in terms of the basic parameters presented previously:

$$\frac{1}{T_{\theta_1}} \doteq \frac{g}{V} \frac{2}{n_{z_\alpha}} (1 - \eta_p)$$

(valid if reasonably near $\left.\frac{\partial \gamma}{\partial V}\right|_{\delta_T} = 0$)

$$\frac{1}{T_{\theta_2}} \doteq \frac{g}{V} n_{z_\alpha} - \frac{1}{T_{\theta_1}}$$

$$\frac{1}{T_{u_1}} \doteq \frac{g}{V} \frac{n_{z_\alpha}}{n_{x_\alpha}}$$

$\frac{1}{T_{\gamma_1}}$: no simple form

$$\frac{1}{T_{u_\theta}} \doteq \frac{g}{V} \left[n_{z_\alpha} - \left(1 - n_{x_\alpha}\right) \tan \theta_T \right]$$

$$\frac{1}{T_{\gamma_\theta}} \doteq \frac{2g}{V} \left(- \tan \gamma_o + \operatorname{ctn} \theta_T\right)$$

A.5 RELATIONSHIPS BETWEEN γ - V CURVES AND STABILITY DERIVATIVES

It is possible to take advantage of the foregoing to develop a set of relationships between steady state γ - V curves and the path dynamics as expressed by dimensional stability derivatives or other alternative parameters.

The most direct means of doing this is to consider a γ - V curve consisting of constant angle of attack and constant throttle contours. (Note that constant pitch attitude contours can be used to determine constant angle of attack contours.) Several combinations of four pieces of information can then be obtained for any given operating point. The four which we shall choose are:

$$\left.\frac{\partial V}{\partial \gamma}\right|_\alpha , \qquad \left.\frac{\partial V}{\partial \delta_T}\right|_\alpha , \qquad \left.\frac{\partial \alpha}{\partial \delta_T}\right|_V , \qquad \left.\frac{\partial V}{\partial \alpha}\right|_{\delta_T}$$

The equations of motion from the previous subsection can be used to form the following equalities:

$$\left.\frac{\partial V}{\partial \gamma}\right|_{\alpha} = \frac{g(X_{\delta_T} \sin \gamma_o - Z_{\delta_T} \cos \gamma_o)}{X_{\delta_T} Z_u - Z_{\delta_T} X_u} \qquad \text{(A.5-1)}$$

$$\left.\frac{\partial V}{\partial \delta_T}\right|_{\alpha} = \frac{X_{\delta_T} \sin \gamma_o - Z_{\delta_T} \cos \gamma_o}{Z_u \cos \gamma_o - X_u \sin \gamma_o} \qquad \text{(A.5-2)}$$

$$\left.\frac{\partial \alpha}{\partial \theta}\right|_{V} = \frac{g}{V} \frac{(X_{\delta_T} \sin \gamma_o - Z_{\delta_T} \cos \gamma_o)}{X_{\delta_T} Z_w^{\dagger} - Z_{\delta_T} X_w} \qquad \text{(A.5-3)}$$

$$\left.\frac{\partial V}{\partial \alpha}\right|_{\delta_T} = V \frac{(X_w \sin \gamma_o - Z_w^{\dagger} \cos \gamma_o)}{Z_u \cos \gamma_o - X_u \sin \gamma_o} \qquad \text{(A.5-4)}$$

Since $\tan \theta_T \triangleq \dfrac{-Z_{\delta_T}}{X_{\delta_T}}$, the equation A.5-1 along with a trigonometric identity and the approximations for X_u and Z_u can be used to show:

$$\left.\frac{\partial V}{\partial \gamma}\right|_{\alpha} \doteq \frac{-V}{2} \tan (\gamma_o + \theta_T)$$

as presented previously in Section A.2.

If thrust is proportional to δ_T and independent of V then

$$Z_{\delta_T} = \frac{-g \cos \gamma_o}{\delta_{T_o}} \eta_p$$

and $$X_{\delta_T} = \frac{g \cos \gamma_o \operatorname{ctn} \theta_T}{\delta_{T_o}} \eta_p$$

Also, $X_u = \frac{2g}{V} \sin \gamma_o \left[1 - \eta_p \operatorname{ctn} \gamma_o \operatorname{ctn} \theta_T\right]$

and, by definition

$$Z_u = \frac{-2g}{V} \cos \gamma_o \, (1 - \eta_p)$$

Substituting these into Equation A.5-2 one can show:

$$\eta_p = \frac{\sin \theta_T}{\cos \gamma_o \sin (\gamma_o + \theta_T)} \frac{\left.\frac{\partial V}{\partial \delta_T}\right|_\alpha}{\left(\left.\frac{\partial V}{\partial \delta_T}\right|_\alpha - \frac{V}{2\delta_{T_o}}\right)}$$

Finally, from Equation A.5-3

$$Z_w^\dagger + X_w \tan \theta_T = \frac{g}{V} \frac{\sin (\gamma_o + \theta_T)}{\cos \theta_T} \left.\frac{\partial \theta}{\partial \alpha}\right|_V$$

and from Equation A.5-4

$$Z_w^\dagger - X_w \tan \gamma_o = \frac{2g}{V^2 \cos \gamma_o} \frac{\left.\frac{\partial V}{\partial \alpha}\right|_{\delta_T}}{\left(1 - \frac{2\delta_{T_o}}{V} \left.\frac{\partial V}{\partial \delta_T}\right|_\alpha\right)}$$

Thus one can solve for the flight path dynamics given a γ - V curve according to the relationships summarized in Table A-1.

TABLE A-1

SUMMARY OF γ - V RELATIONS TO FLIGHT PATH DYNAMICS

GIVEN: γ versus V for constant α, δ_T

At the operating point γ_o, V and trim α_o, δ_{T_o} evaluate the partial derivatives:

$$\left.\frac{\partial V}{\partial \gamma}\right|_{\alpha}, \qquad \left.\frac{\partial V}{\partial \delta_T}\right|_{\alpha}, \qquad \left.\frac{\partial \alpha}{\partial \theta}\right|_{V}, \qquad \text{and } \left.\frac{\partial V}{\partial \alpha}\right|_{\delta_T}$$

$$\left(\text{Note } \left.\frac{\partial \alpha}{\partial \theta}\right|_{V} = \frac{1}{1 + \left.\frac{\partial \gamma}{\partial \alpha}\right|_{V}} \right)$$

Solve for θ_T from:

$$\theta_T = \arctan\left(- \frac{2}{V} \left.\frac{\partial V}{\partial \gamma}\right|_{\alpha} \right) - \gamma_o$$

Solve for η_p from:

$$\eta_p = \frac{\sin \theta_T}{\cos \gamma_o \sin (\gamma_o + \theta_T)} \frac{\left.\frac{\partial V}{\partial \delta_T}\right|_{\alpha}}{\left(\left.\frac{\partial V}{\partial \delta_T}\right|_{\alpha} - \frac{V}{2\delta_{T_o}} \right)}$$

TABLE A-1 (Concluded)

Solve for X_{δ_T}, Z_{δ_T}, X_u, Z_u from:

$$X_{\delta_T} = \frac{g \cos \gamma_o \operatorname{ctn} \theta_T}{\delta_{T_o}} \eta_p$$

$$Z_{\delta_T} = \frac{-g \cos \gamma_o}{\delta_{T_o}} \eta_p$$

$$X_u = \frac{2g}{V} \sin \gamma_o \left[1 - \eta_p \operatorname{ctn} \gamma_o \operatorname{ctn} \theta_T \right]$$

$$Z_u = -\frac{2g}{V} \cos \gamma_o (1 - \eta_p)$$

Solve for X_w and $Z_w^\dagger$ from:

$$X_w = \frac{\left[\frac{g}{V} \frac{\sin(\gamma_o + \theta_T)}{\cos \theta_T} \left.\frac{\partial \theta}{\partial \alpha}\right|_V - \frac{2g}{V^2 \cos \gamma_o} \frac{\left.\frac{\partial V}{\partial \alpha}\right|_{\delta_T}}{\left(1 - \frac{2\delta_{T_o}}{V} \left.\frac{\partial V}{\partial \delta_T}\right|_\alpha \right)} \right]}{\left(\tan \theta_T + \tan \gamma_o \right)}$$

and $$Z_w^\dagger = X_w \tan \gamma_o + \frac{2g}{V^2 \cos \gamma_o} \frac{\left.\frac{\partial V}{\partial \alpha}\right|_{\delta_T}}{\left[1 - \frac{2\delta_{T_o}}{V} \left.\frac{\partial V}{\partial \delta_T}\right|_\alpha \right]}$$

Finally,

$$n_{z_\alpha} = -\frac{V}{g} Z_w^\dagger$$

$$n_{x_\alpha} = 1 - \frac{V}{g} X_w$$

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A-2 Maskell, E. C. and D. A. Spence, A Theory of the Jet Flap in Three Dimensions, Proceedings of the Royal Society of London, A, Vol. 251, pp. 407-425, 1959.

A-3 McRuer, Duane, Irving Ashkenas, and Dunstan Graham, Aircraft Dynamics and Automatic Control, Princeton University Press, Princeton, 1973.

APPENDIX B

ATMOSPHERIC DISTURBANCE MODELING

This appendix presents a summary of the experience derived during this program regarding atmospheric disturbance modeling. It was clear that simulator results depended heavily upon the presence of atmospheric disturbances. Thus, this was a continuing area of concern. An appreciation of the important aspects of atmospheric disturbances is difficult to obtain because the literature contains many apparent conflicts in modeling forms, definitions, and areas of emphasis. The following is an effort to address some of these conflicts by considering limitations imposed by the handling qualities and performance features of a low-speed aircraft operating at low altitudes.

The organization of this appendix follows the general chronological sequence in our study of disturbance effects. We began the program with what was considered a widely accepted atmospheric model, however, midway through the program, this model was reconsidered because of concerns expressed by subject pilots. Alternatives were studied, and finally, based on the results of this study and a short simulator experiment, it was decided to continue with the original model. Thus, this appendix is subdivided in the following manner:

- A definition of the disturbance model used throughout this experiment
- A study of disturbance model alternatives
- The results of a simulation experiment to explore major modeling dissimilarities
- A summary of important factors concerning atmospheric disturbance modeling.

A few introductory remarks should be made. First, an atmospheric disturbance model may be characterized by a deterministic component and a random component. The emphasis given here is on the latter. In this particular program, the relative effects of the deterministic model were weak compared to the random turbulence effects. This would probably not be the case if the effects of strong wind shears were of particular interest. It will also be noted that special emphasis is given to the longitudinal random disturbance component. This is due to the nature of the pilot/vehicle system and the range of altitudes of interest as described shortly.

B.1 THE ATMOSPHERIC MODEL USED IN THIS PROGRAM

During this simulation program, the deterministic wind model and the random turbulence model were two relatively independent entities. The features of the deterministic wind included magnitude, direction, and shear profile. Each of these were sometimes varied on a run-to-run basis and the specific sets of characteristics used are described in the respective simulation reports (References B-1 through B-4).

The random turbulence model used during this simulation program was based on the MIL-F-8785B Dryden model as described in Reference B-5. Turbulence intensity, the only parameter which was varied, was characterized in terms of a probability of exceedance.

The specific properties of the random turbulence model used throughout this program are given in Table B-1. The horizontal gust intensity, σ_{u_g}, was held constant during any given run. The standard level of turbulence used corresponded to a nominal probability of exceedance of 10%. It should be noted that in the basic MIL-F-8785B model, σ_{u_g} is actually a weak function of altitude as shown in Figure B-1.

To summarize, the important features of the model used are:

- Horizontal gust intensity is a specified constant independent of altitude
- Horizontal scale length varies with the cube root of altitude

TABLE B-1

PROPERTIES OF THE RANDOM TURBULENCE MODEL USED IN THIS PROGRAM

SPECTRAL FORM: Dryden

TURBULENCE FILTERS:

Horizontal: $$\frac{1}{1 + \frac{L_u}{V} s}$$

Lateral: $$\frac{1 + \frac{\sqrt{3}\,2L_v}{V} s}{\left(1 + \frac{2L_v}{V} s\right)^2}$$

Vertical: $$\frac{1 + \frac{\sqrt{3}\,2L_w}{V} s}{\left(1 + \frac{2L_w}{V} s\right)^2}$$

SCALE LENGTHS:

$$L_u = 2L_v = \sqrt[3]{h_o^{2}\, h}$$

$$h_o = 533.4 \text{ m } (1750 \text{ ft})$$

$$2L_w = h$$

TURBULENCE INTENSITY:

σ_{u_g} = constant; 1.4 m/sec (4.5 ft/sec) for $p_e = 10\%$ (Standard)
2.0 m/sec (6.5 ft/sec) for $p_e = 1\%$

$$\frac{\sigma_{u_g}^{2}}{L_u} = \frac{\sigma_{v_g}^{2}}{2L_v} = \frac{\sigma_{w_g}^{2}}{2L_w}$$

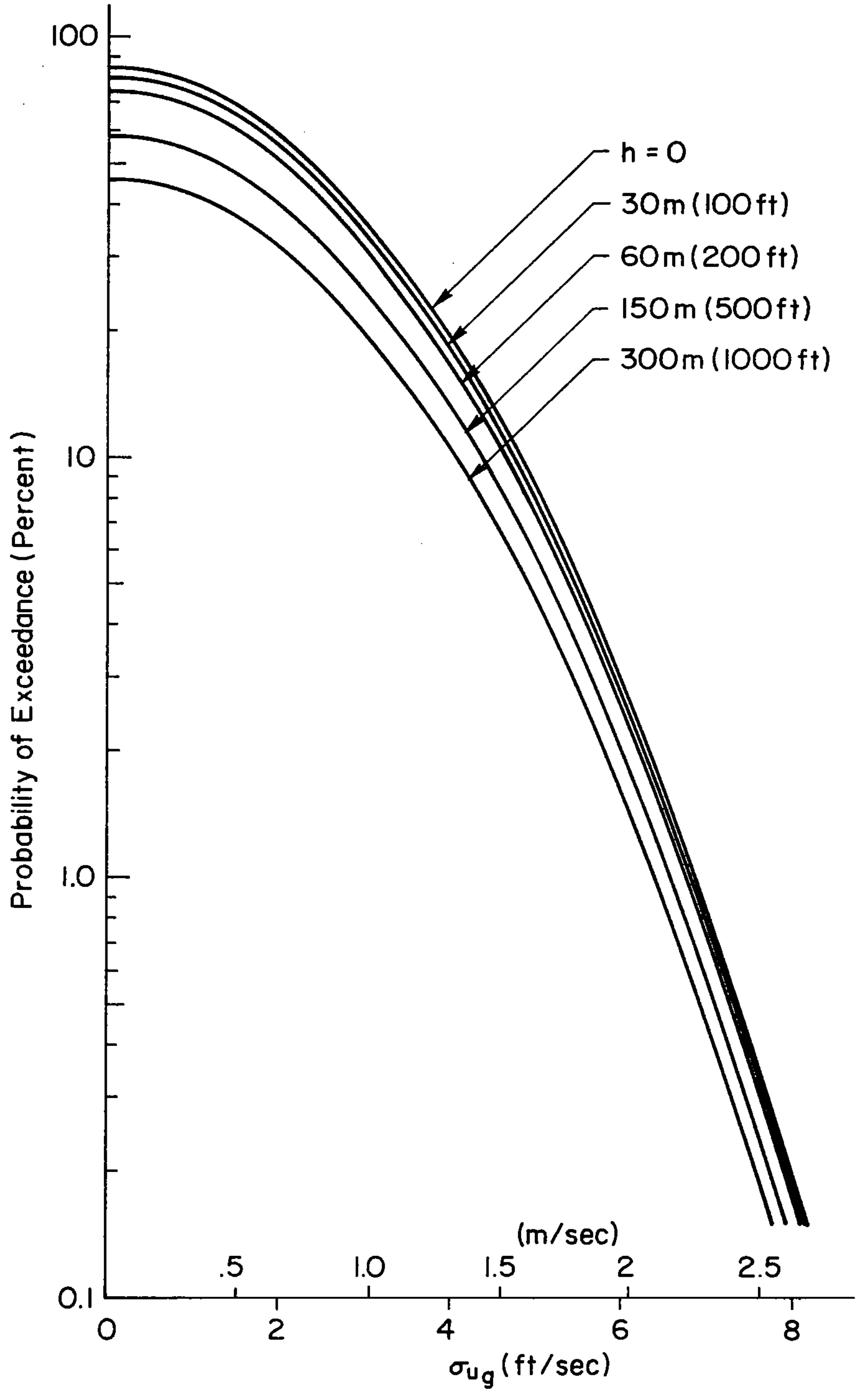

Figure B-1: Probability of Exceedance for Horizontal Gust Intensity of MIL-F-8785B Turbulence Model

- Vertical gusts are characterized by a decreasing intensity and increasing choppiness as low altitudes are approached
- The dominant characteristics are the horizontal gust intensity and scale length (which will be shown more clearly in the following paragraphs).

B.2 A SURVEY OF MODELING ALTERNATIVES

While use of the standard level of turbulence for the model previously described seemed to be effective in revealing key features and limitations of powered-lift aircraft, some pilots expressed the opinion that the turbulence seemed unrealistically severe. During the period between the AWJSRA simulation and the first Generic STOL simulation, the matter of turbulence modeling was studied in order to either confirm the original turbulence model used or to recommend an alternative which might be regarded as more realistic. This study involved a review of low altitude turbulence data along with other atmospheric models. Unfortunately it was found that the background data and the various models differed widely, both in mathematical forms and numerical definition. A factor which helped to simplify this overview, though, was consideration of the closed loop pilot/vehicle system. This resulted in showing that seemingly extremely different atmospheric models can have a nearly equivalent effect on flight path performance of the pilot/vehicle. Ultimately, it was found that the original model was probably no worse nor no better than any reasonable alternative.

We begin our discussion of modeling alternatives by considering the restrictions imposed by the pilot/vehicle in combination with the random disturbance. This allows us to focus on a limited spectral range and on a particular axis of the disturbance. Then, with this point of view, three competing models are compared.

We can gain some appreciation of the important aspects of atmospheric disturbances on flight path performance by the following approach. First, we assume that the pilot/vehicle is approximated by pitch attitude and throttle held constant. Next, as a metric for flight path dispersion, we use altitude rate, $\dot{h}$. Finally, we will consider horizontal and vertical gust disturbances represented simply by the normal Dryden spectral forms.

The key airframe transfer functions are:

$$\frac{\dot{h}}{u_g} \doteq \frac{-Z_u s}{\left(s + \frac{1}{T_{\theta_1}}\right)\left(s + \frac{1}{T_{\theta_2}}\right)}$$

and $$\frac{\dot{h}}{w_g} \doteq \frac{1}{\left(T_{\theta_2} s + 1\right)}$$

In the following paragraphs, we consider first the effect of horizontal gust alone, next the effect of vertical gusts alone, and finally the combined effects of horizontal and vertical gusts. The result of this will be to identify the important relationships between pilot/vehicle frequency response and gust frequency response, and particularly the dominance of the horizontal gust component.

In order to illustrate the sensitivity of flight path to horizontal gusts let us consider the following example.

$$\frac{1}{T_{\theta_1}} = 0.1 \text{ sec}^{-1}$$

$$\frac{1}{T_{\theta_2}} = 0.5 \text{ sec}^{-1}$$

$$Z_u = -.3 \text{ sec}^{-1}$$

These values are representative of a fairly wide range of vehicles including not only powered-lift aircraft but also conventional aircraft. Representation is at least adequate for the general argument presented here.

The ratio of RMS altitude rate to RMS horizontal gust versus the horizontal gust break frequency, $\frac{V}{L_u}$, is shown in Figure B-2. This figure shows that if the horizontal gust break frequency falls generally between the effective speed damping, $1/T_{\theta_1}$, and heave damping, $1/T_{\theta_2}$, of the basic airframe, then the RMS altitude rate is fairly insensitive to the gust break frequency, $\frac{V}{L_u}$. This is also the frequency range for which the airframe is most sensitive to horizontal gusts. In the following pages we shall see that horizontal gust break frequencies do in fact appear to fall within this range. Thus the relative insensitivity to horizontal gust break frequency is an important aspect.

Now consider the effect of the vertical gust component in a similar manner. First note that the main feature characterizing the airframe is the effective heave damping, $1/T_{\theta_2}$. If we use the same value as in the previous example, i.e., $1/T_{\theta_2} = 0.5 \text{ sec}^{-1}$, then we can generate a similar plot of RMS altitude rate to RMS vertical gust intensity versus the effective vertical gust breakpoint. This is shown in Figure B-3. Note that for vertical gusts having higher frequency content, the effect on flight path decreases, i.e., the airframe simply acts as a low pass filter. It will be found that as the aircraft descends in altitude for the latter stages of the approach, the vertical gust break frequency tends to be somewhat higher than the effective heave damping, $1/T_{\theta_2}$. Further, the intensity of the vertical gust component is somewhat less than the horizontal component.

We can get a better view of the comparative effects of the horizontal and vertical gust components if we consider a given turbulence model. In particular, we need a means of relating horizontal and vertical gust intensity and scale length to altitude. For the purposes of illustration we will use the MIL-F-8785B model to provide these relationships. It will be shown in the next subsection that the results obtained from this model are similar in character to those from other models considered.

Figure B-4 shows the RMS altitude rate to RMS horizontal gust versus altitude for each of the two gust components. The airframe dynamics are the same used in the two previous figures. Note that at an altitude of about 100 m (330 ft) the two components are about equally influential, but

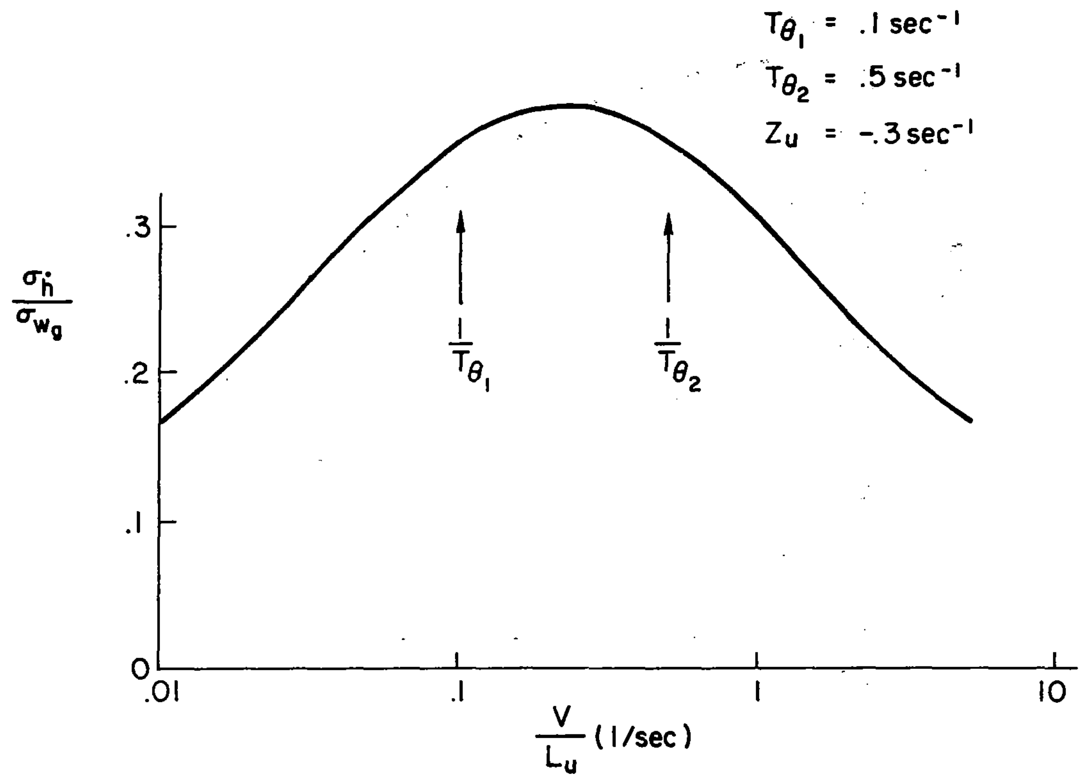

Figure B-2: Effect of Horizontal Gust on Altitude Rate Versus Horizontal Gust Break Frequency (Attitude and Throttle Fixed)

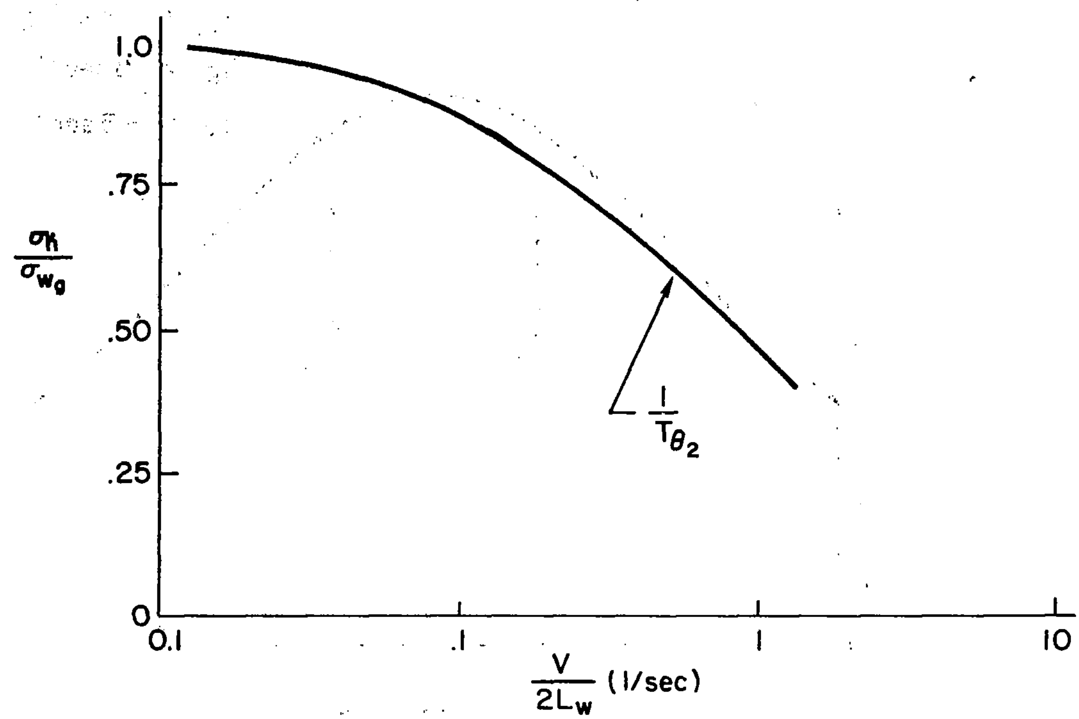

Figure B-3: Effect of Vertical Gust on Altitude Rate Versus Vertical Gust Break Frequency (Attitude and Throttle Fixed)

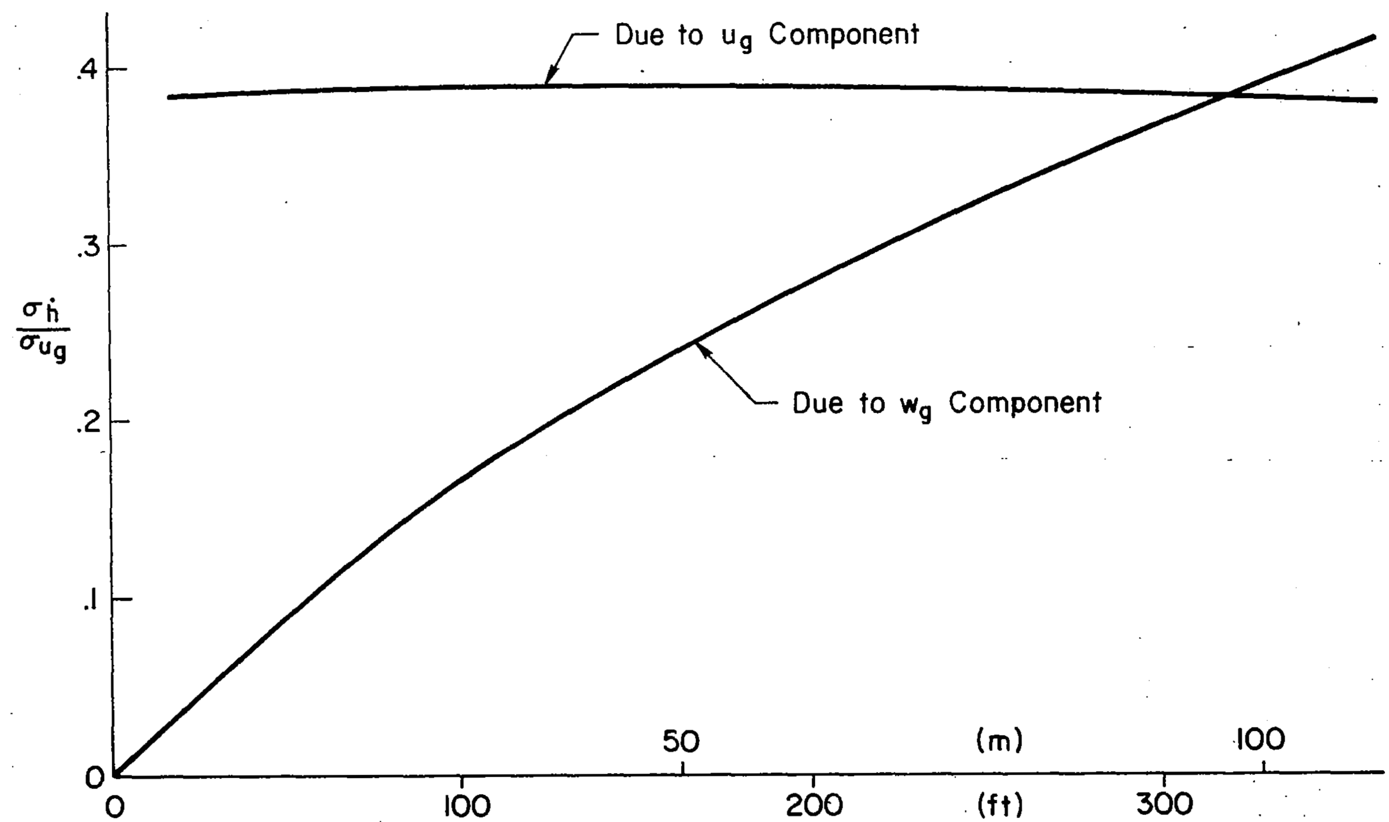

Figure B-4: Effect of Horizontal and Vertical Gusts on Altitude Rate Versus Altitude (Attitude and Throttle Fixed) below that, the effect of the vertical component becomes considerably weaker. In particular, in the critical altitude range below 60 m (200 ft) only the horizontal gust component has a significant effect on the RMS altitude rate excursion.

To the extent that the above line of reasoning and assumptions are sound, we can then direct our attention to the horizontal gust component restricted to the spectral range defined by the basic speed damping and heave damping of the bare airframe. This restricted view of random atmospheric disturbances greatly facilitates handling the various conflicts in model forms and numerical values.

With the above ideas in mind, we now compare three competing atmospheric models, and in particular, their random turbulence properties. The three models are:

i) The MIL-F-8785B Dryden Model (Reference B-5)

ii) The Etkin low altitude turbulence model (Reference B-6)

iii) The Boeing atmospheric model (Reference B-7)

We will discuss the following aspects of the above models:

- Probability density function
- Spectral form
- Intensity
- Scale length
- Mean wind dependence.

Where possible, we limit discussion to only the horizontal gust component since the vertical component can be considered less important for reasons stated previously.

All three of the atmospheric models considered here assume a Gaussian probability density function to describe random turbulence. While this seems to agree reasonably well with actual measurements, the most compelling reason for this assumption is probably in the ease of handling analytically. Reference B-10 presents the results of a study in which a non-Gaussian model is considered. While this report indicates that pilots were able to discern differences in probability density function to some degree, it appears to be not really a first-order effect.

The spectral description of the turbulence model describes the time varying nature of each gust component. For each component the spectral description is composed of the spectral form, the intensity (e.g., RMS level), and the characteristic scale length. Let us begin by considering the spectral form. For the sake of brevity, we will consider only the horizontal (longitudinal) component.

The two spectral forms most commonly used are the Dryden and Von Karman. Each of these has a certain advantage. The Dryden form is easily simulated since it is equivalent to white noise through a simple first-order low-pass filter. Unfortunately, it is commonly considered to be an inadequate representation of real world spectral properties of turbulence. The Von Karman spectral form, on the other hand, is considered to be a good representation of the real world, but it cannot be modeled with a physically realizable filter. In order to approach the Von Karman form, a high-order filter more complex than the Dryden filter must be used. In the following paragraphs, though, we show that the distinctions between these two spectral forms are not necessarily great, at least when viewed in the context of the pilot/vehicle system of interest here.

At this point we should note that the three turbulence models we are considering here employ both spectral forms. The MIL-F-8785B model employs the Dryden form (the MIL-F-8785B model also includes a Von Karman form as an alternative but we shall not consider it in our discussions). Both the Boeing and Etkin models are based on a Von Karman spectral form. Recognizing the inability to physically realize this form, the Boeing model actually utilizes a second-order filter which approximates the Von Karman form out to a reasonably high frequency.

Now, consider some important aspects of the Dryden versus Von Karman spectral form. First, let us compare the power spectral density of the two forms. This is shown in Figure B-5, a normalized frequency (no distinction is made between the so-called Dryden scale length and Von Karman scale length, at this point they are assumed equal). Note that the two power spectral densities are within 1/2 db of one another for the majority of the region one decade below and one decade above the breakpoint, $\frac{V}{L_u}$ rad/sec. Only at high frequencies do the two curves show a significant divergence. The implication is that if our spectral range of interest lies near the filter breakpoint then either spectral form is adequate. We shall see that this is, in fact, the case. Let us make this comparison, though, in a slightly more direct way.

A useful way of viewing spectral power is to plot the product of power spectral density and frequency normalized by the variance. This function versus the log of frequency gives a direct indication of the power contained in a given spectral range*. This is shown in Figure B-6. The main feature of this plot is that both spectral forms concentrate the power in a region near the breakpoint. In this region, a slightly better match could be obtained by simply decreasing the Dryden intensity by about 8%. If, on the other hand, the frequency range of interest were at higher frequencies, then the two spectral forms would be matched by sliding them laterally, i.e., adjusting the effective scale lengths.

We can conclude from the above that if our spectral range of interest is in fact centered about the gust filter break frequency, i.e., that the airplane frequency response as indicated by the $1/T_{\theta_1}$ to $1/T_{\theta_2}$ range, then there is no major distinction between the two spectral forms and that one form could be used in place of the other. Clearly, the Dryden form is the more attractive because of its convenience.

Now consider the aspect of gust intensity. Again we shall treat only the horizontal component, u_g. We will choose to make our comparison of turbulence intensity of various models and data sources in terms of probability of exceedance.

Table B-2 shows a comparison of RMS u_g for two data sources and two turbulence models. The point of comparison consists of probability of exceedance of 10% and 1% at an altitude of approximately 30 m (100 ft).

* The area under the curve for any frequency band is proportional to the power in that frequency band.

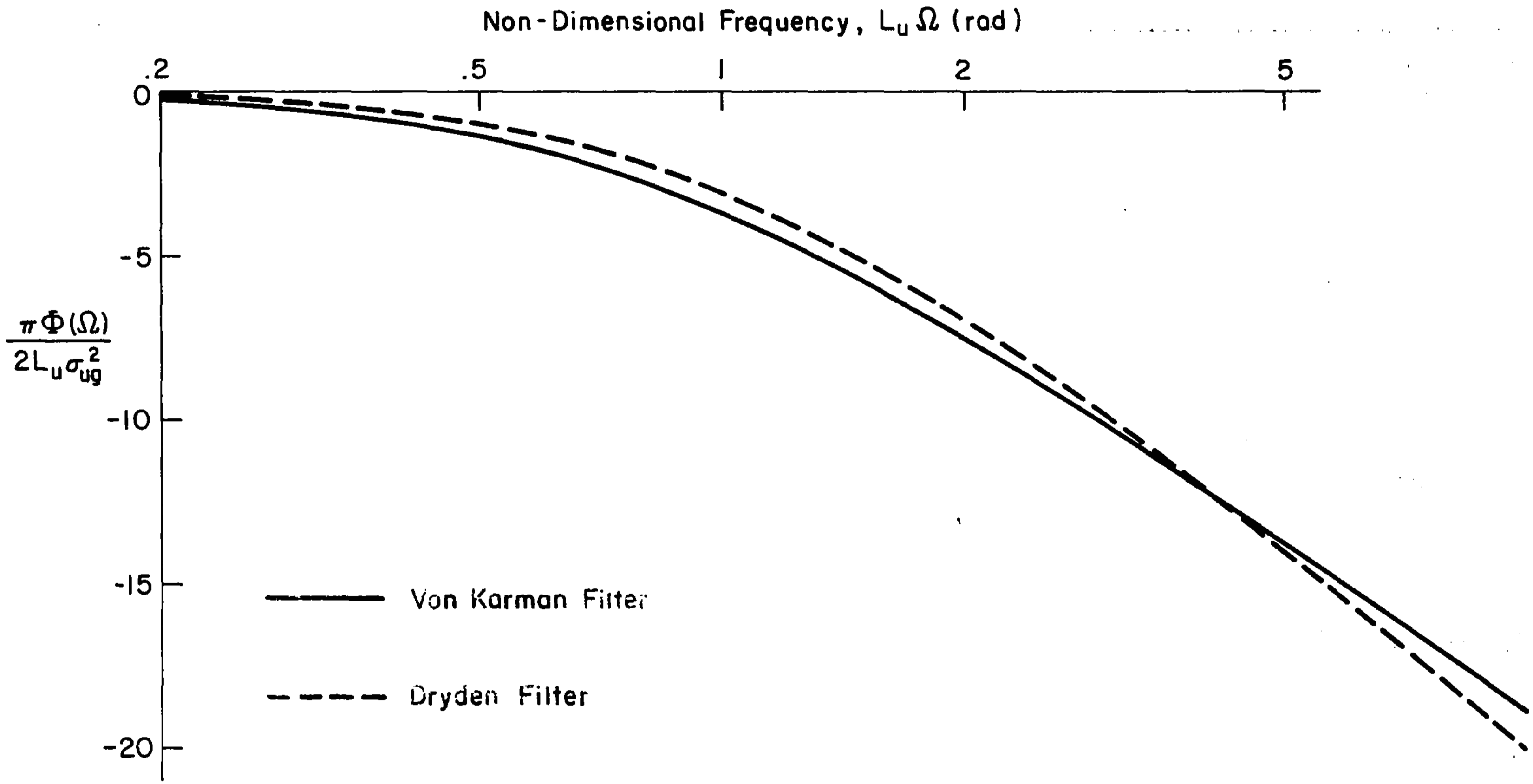

Figure B-5: Comparison of Normalized Horizontal Gust Power Spectral Density for Von Karman and Dryden Spectral Forms

Notes:

- Area under curves for a given frequency range corresponds to spectral power
- Total area under either curve is .4343 if unit on abscissa is taken as one decade
- Spatial frequency, Ω related to temporal frequency, ω by airspeed, V or $\Omega = \omega/V$

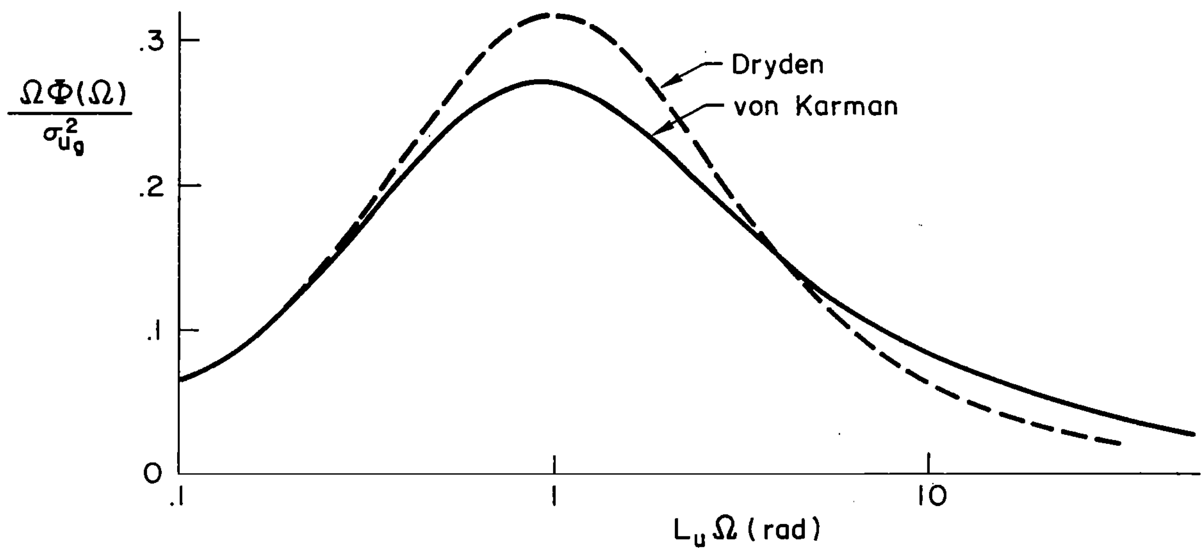

Figure B-6: Direct Comparison of Spectral Forms

TABLE B-2

COMPARISON OF RMS u_g FOR SEVERAL DATA SOURCES AND MODELS

SOURCE	σ_{u_g}, m/sec (ft/sec) PROBABILITY OF EXCEEDANCE 10%	1%
MIL-F-8785B (30 m)	1.43 (4.7)	2.07 (6.8)
Boeing (30 m, neutral lapse)	1.34 (4.4)	2.09 (6.85)
Battelle (Lake Union, 25 m, adjusted*)	1.49 (4.9)	2.38 (7.8)
LO-LOCAT III (overall, 75 m)	1.30 (4.25)	1.83 (6.0)
Nominal values used in this simulation program	1.4 (4.5)	2.0 (6.5)

* The data was adjusted to account for a difference between the mean wind measured during the study period and the climatological mean wind, assuming that RMS gust intensity is approximately proportional to mean wind. In this case the mean wind measured was 76% the climatological mean. Therefore, the RMS values were adjusted upward 32%.

Values for the MIL-F-8785B model are taken directly from Figure B-1 for an altitude of 30 m (100 ft). In using the Boeing model we assume a neutral lapse rate, a surface roughness corresponding to a normal airport (4.6 cm), and an altitude of 30 m. The Etkin model is not involved because it contains no probability of exceedance relationships. The data from Reference B-8 is based on a recent study involving V/STOL port sites, and the data from Reference B-9 is of particular interest because it involves relatively long-term turbulence measurements made at a reasonably low altitude (approximately 75 m). As noted in the table, it has been adjusted to reflect the climatological mean for the area measured.

The tabulated RMS u_g values for the various models and data sources do in fact compare closely with those nominal values used during this simulation program. This is particularly true of the standard turbulence level taken at 10% probability of exceedance.

The variation of turbulence intensity with altitude is shown in Figure B-7. The Boeing model varies the most as a function of altitude. However, at those altitudes corresponding to the critical part of the approach (below 60 m) the comparison is good. We should note also that the Etkin model assumes σ_{u_g} is constant with altitude.

The next model feature we consider is the turbulence scale length. Up to this point there has been little to distinguish the three models we are considering. In the case of scale length, there appears to be a substantial difference.

From the foregoing discussion of spectral properties, it is clear that the scale length directly determines where the spectral power is centered. If the scale length is such that the spectral power of the disturbance is centered within the bandwidth of the pilot/vehicle then we can expect some significant effect. This appears to be the case for aircraft in the latter stages of the approach phase.

First, let us consider how scale length is specified by each of the three models we are considering. In each case, scale length is a direct function of altitude as summarized in Table B-3. The forms are clearly quite different. However, a better illustration of their differences is

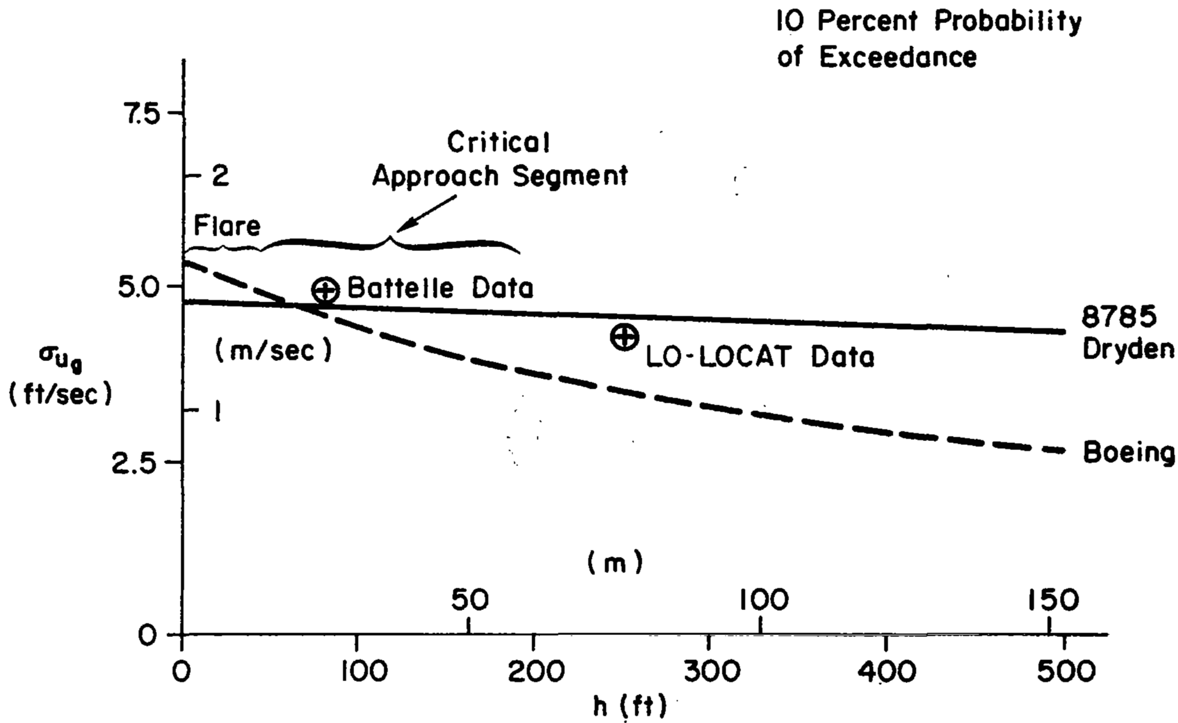

Figure B-7: RMS u_g versus Altitude

TABLE B-3

HORIZONTAL SCALE LENGTH, L_u, FOR VARIOUS MODELS

MIL-F-8785B (DRYDEN):

$$L_u = \sqrt[3]{h_o^2 h} \qquad h_o = 533.4 \text{ m (1750 ft)}$$

BOEING:

$$L_u = \frac{h}{\left(0.177 + 0.723 \frac{h}{h_o}\right)^{1.2}} \qquad h_o = 304.8 \text{ m (1000 ft)}$$

ETKIN:

$$L_u = \sqrt{h_o h} \qquad h_o = 121.9 \text{ m (400 ft)}$$

shown in Figure B-8 in which scale length is plotted versus altitude. This shows a wide variation especially in the critical low altitude region. In general, the Etkin and the MIL-F-8785B models bracket the range of scale lengths from numerous measurements reported in the literature. Reference B-9 is considered to have one of the better sets of scale length measurements because of its relatively long-term measurements. It shows that scale lengths do, in fact, vary widely in the real world. In fact, the data plotted in Figure B-9 suggests some relationship between scale length and gust intensity. Milder gust intensities involve scale lengths of the same order of magnitude as those of the Etkin model, while higher gust intensities correspond to scale lengths comparable to the MIL-F-8785B model. According to these data, the standard 10% level of turbulence used in this program was in agreement with the relatively long scale length provided by the MIL-F-8785B Dryden model.

In the last analysis, though, it will be seen that the overall pilot/vehicle performance is not really all that sensitive to a large variation in scale length. To see this, let us now consider how the widely varying horizontal scale lengths of the three models affect pilot/vehicle performance. We can do this in a manner similar to that done in the first part of this subsection, i.e., compute the RMS altitude rate as a function of gust scale length for a sample airplane. In addition to the case of no active control of flight path, i.e., attitude and throttle fixed, we shall also consider the approximate effect of moderate to tight control of flight path. This can be done in a simple fashion by assuming that the pilot has modified the denominator of the airframe transfer function by regulation of flight path. Based on observations made in the simulator, we assume that tight flight path control can be represented by a closed-loop frequency of 0.5 rad/sec and a damping ratio of 0.3. A moderate degree of flight path control can be represented by a frequency of 0.3 rad/sec and damping ratio of 0.5. For the previously used example, a speed of 75 kt, and a horizontal gust intensity of 1.4 m/s (4.5 ft/s), the RMS altitude rate is plotted versus horizontal scale length in Figure B-10. Horizontal scale lengths are indicated for each of the three models over the critical approach altitudes of 60 m (200 ft) down to 15 m (50 ft). This plot suggests that over the wide range

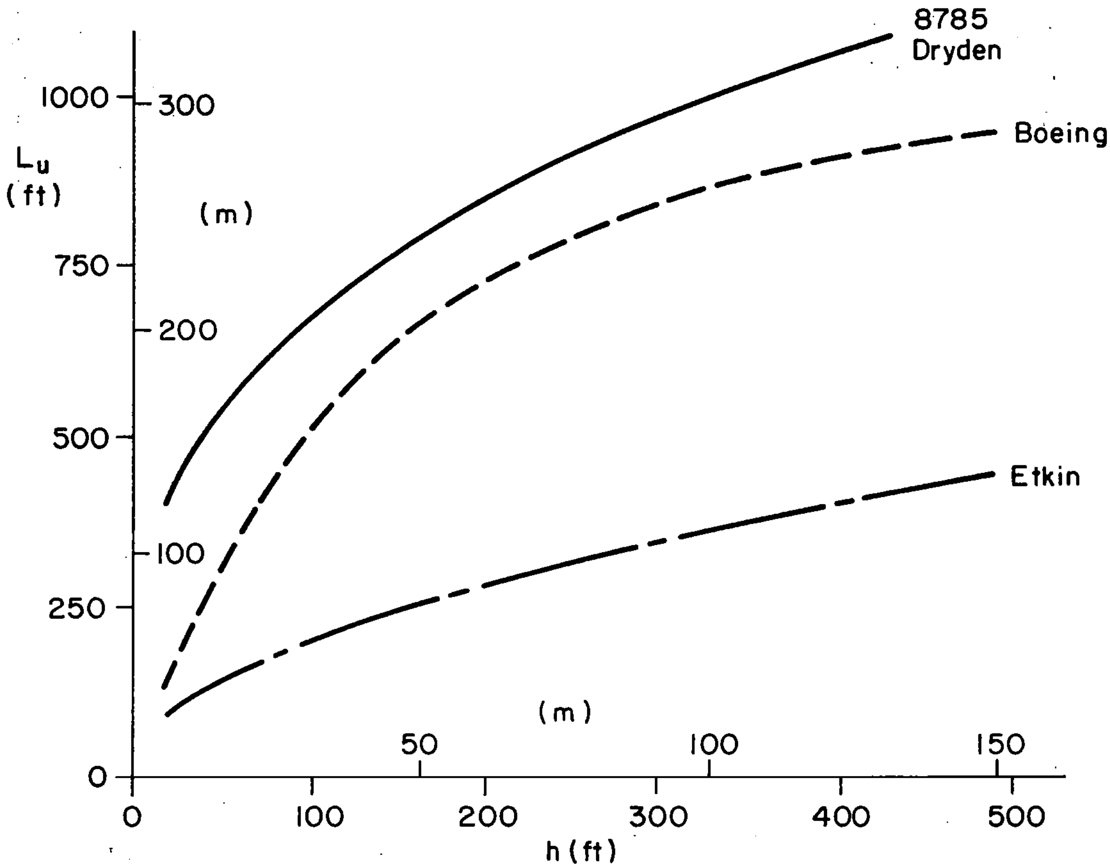

Figure B-8: Horizontal Scale Length Versus Altitude

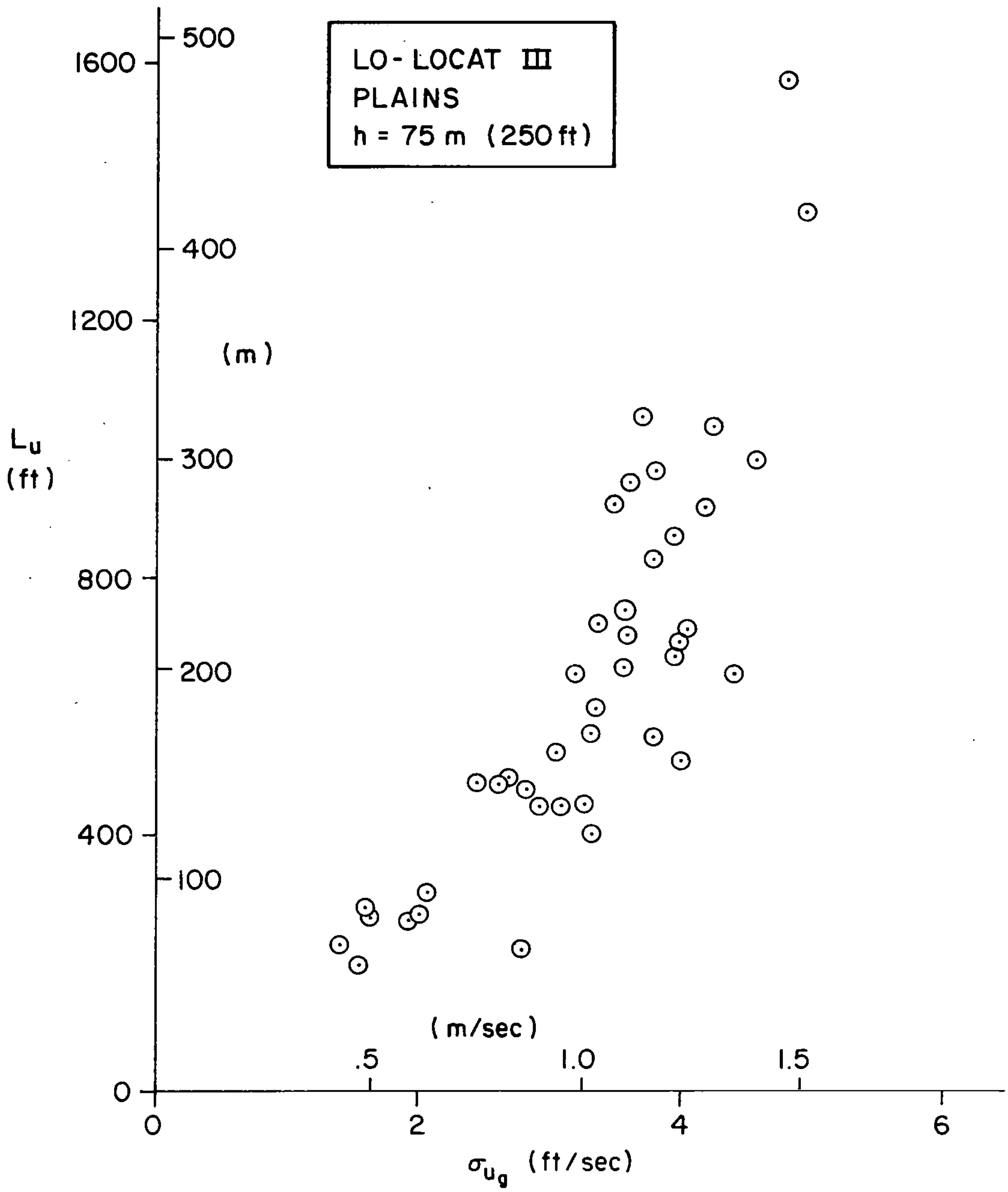

Figure B-9: Measured Values of Scale Length Versus RMS u_g

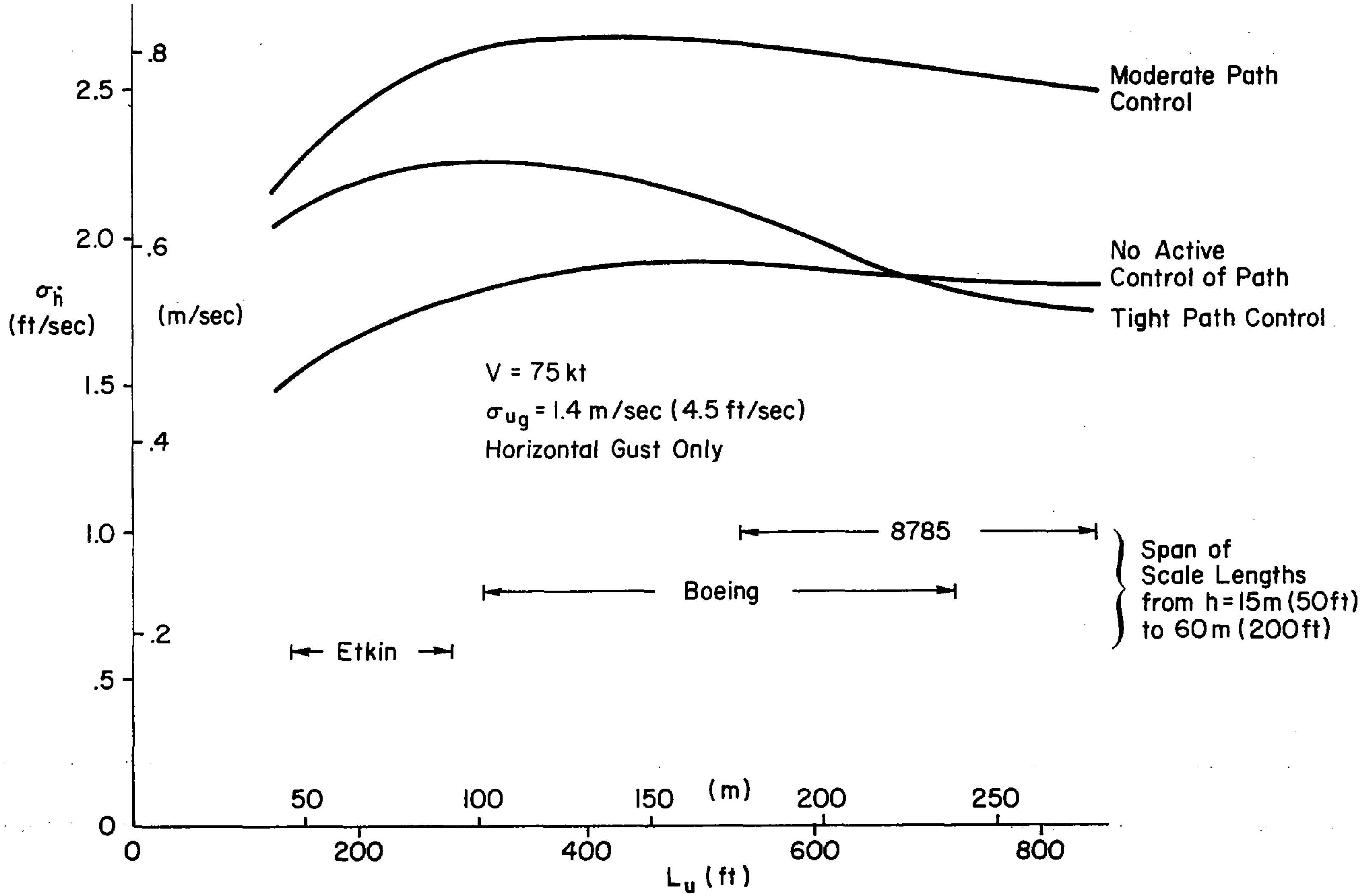

Figure B-10: Effect of Horizontal Gust on Altitude Rate Versus Horizontal Gust Scale Length (Attitude and Throttle Fixed) of scale lengths represented in these three models, there is relatively little effect on the flight path performance as indicated by RMS altitude rate.

Up to this point we have dealt mainly with the turbulence aspect of the wind models. Let us briefly consider the mean wind aspect.

Mean wind dependence on the random turbulence model is a feature unique to the Boeing model. This dependence is associated with the behavior of a planetary boundary layer. Thus there is a specified mean wind profile which combines with the random wind shear. The dominant feature, however, is simply the magnitude of the mean wind. For example, the probability of exceedance of 10% leads to a mean wind of 20 kt at 30 m (100 ft) and 15 kt at 6 m (20 ft). When using the MIL-F-8785B and Etkin models, the mean wind must be arbitrarily defined and combined with the random turbulence as was done in this simulation program.

Now let us summarize our analysis of the effects of turbulence and of the three turbulence models considered.

- The horizontal gust is the most important component in the critical approach altitudes below 60 m (200 ft).
- The RMS gust intensity has the most direct effect on pilot/vehicle performance.
- Horizontal scale length is the most variable feature among the models considered but does not have a corresponding effect on pilot/vehicle performance, at least not on sink rate.
- Von Karman and Dryden spectral forms and scale lengths may be used interchangeably in the spectral band of interest in this program.

The above factors led to the brief simulator experiment described in the following subsection.

B.3 A SHORT SIMULATION EXPERIMENT TO EXPLORE RANDOM TURBULENCE CHARACTERISTICS

The study of turbulence models described in the previous subsection showed that the most extreme differences of any likely consequence were in turbulence scale lengths. The standard model used in this program (based on the MIL-F-8785B model) was characterized by relatively long scale lenths while, at the other extreme, the Etkin model involved relatively short scale lengths. It was decided to perform a simulator test on this one aspect of random turbulence in order to measure its influence on turbulence realism and apparent severity. As mentioned previously, analysis had shown that these extremes in scale length should not really have a large impact on pilot/vehicle performance, but the effect on perceived realism was not known.

Turbulence model characteristics which were compared experimentally are summarized in Table B-4. The most prominant point of comparison was the horizontal scale length.

An airplane model which approximated the DHC-6 Twin Otter was used to explore these two different sets of characteristics. This airplane model is described in Reference B-3.

The simulator experiment consisted of flying normal visual approaches similar to those performed during previous powered-lift investigations. Two subject pilots participated in the evaluation. Each one flew a series of approaches with one turbulence model, then a change was made to the second model. The pilots were informed only when the change was made but the models were not identified.

The results of this brief experiment did not favor one turbulence model over the other. While there were indications that the pilots could sense the difference in choppiness between the two models*, neither could be termed more realistic. Both pilots considered the turbulence with either model to be realistic for some runs and unrealistic for other runs. A summary of pilot comments is included in Reference B-3.

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* Etkin model had more high frequency content because of smaller L_u.

TABLE B-4

CHARACTERISTICS COMPARED EXPERIMENTALLY

	ORIGINAL MODEL	ALTERNATIVE MODEL
Basic	MIL-F-8785B	Etkin
σ_{ug} m/sec (ft/sec)	1.4 (4.5)	1.4 (4.5)
L_u m (ft)	$\sqrt[3]{h_o^2 h}$ $h_o = 533.4$ (1750)	$\sqrt{h_o h}$ $h_o = 121.9$ (400)
σ_{vg}	$\sqrt{\frac{2L_v}{L_u}}\,\sigma_{ug}$	$0.8\ \sigma_{ug}$
$2L_v$*	L_u	$2L_w$
σ_{wg}	$\sqrt{\frac{2L_w}{L_u}}\,\sigma_{ug}$	$0.5\ \sigma_{ug}$
$2L_w$*	h	0.4 h**

* The coefficient of 2 is included to clearly denote the use of the L_v and L_w definitions most frequently used in the literature. The spectral break frequencies are thus $V/2L_v$ and $V/2L_w$ while for the horizontal component it is V/L_u.

** This was an error in interpreting the Etkin model, $2L_w$ should have been 0.8 h in which case it would have been nearly equal to the MIL-F-8785B model.

Due to the apparent equivalence of the two sets of characteristics studied, it was decided to continue using the original turbulence model for the remainder of this simulator program, with consistency being the deciding factor.

B.4 A SUMMARY OF THE CONSIDERATIONS IN MODELING ATMOSPHERIC DISTURBANCES

The foregoing discussion of atmospheric disturbance modeling has included a number of important ideas which will be briefly summarized below. These ideas apply to flight in the low-altitude stages of the approach and landing flight phase, which is considered the most critical part of all terminal area operations. Further, these remarks apply to flight path control aspects as opposed to attitude control aspects which occur in a higher frequency range.

The most important feature overall is the horizontal gust intensity, σ_{u_g}.

The frequency band of importance is approximately bounded on the low end by the airframe speed damping, $1/T_{\theta_1}$, and on the high end by heaving damping, $1/T_{\theta_2}$. Typically this is in the range between 0.1 rad/sec and 0.5 rad/sec. The spectral form need be valid only over the above range. Hence both the Dryden and Von Karman forms are for all practical purposes equivalent.

The longitudinal scale length can vary over a wide range without significantly changing the net effects of atmospheric disturbances. Therefore, it is difficult to distinguish between various scale length variations with altitude or between large differences in magnitudes.

Vertical gusts are of relatively minor importance because of their characteristically lower magnitude and shorter scale length compared to horizontal gusts.

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