REPORT NUMBER 162

FLIGHTWORTHINESS AND RELIABILITY SUMMARY REPORT

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XV-5A LIFT FAN FLIGHT RESEARCH AIRCRAFT

Contract No. DA 44-177-TC-715

145
265

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ADVANCED ENGINE AND TECHNOLOGY DEPARTMENT
GENERAL ELECTRIC COMPANY
CINCINNATI, OHIO 45215

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LIST OF SYMBOLS

R/C	Rate of climb, feet/minutes
GW	Gross weight
N_G	Gas generator RPM
M_θ	Rate of change pitching moment with respect to pitching velocity
I_y	Moment of inertia, Y axis
i_t	Incidence angle, horizontal tail
I_x	Moment of inertia, X axis
I_z	Moment of inertia, Z axis
C_m	Pitching moment coefficient
K_e	Elevator effectiveness parameter
$C_{m_{\delta e}}$	Pitching moment coef, due to elevator δ
δ_e	Elevator defl
$C_{\ell_{\delta a}}$	Rolling moment due to aileron δ
δ_{a_L}	Left aileron defl.
δ_{a_R}	Right aileron defl.
C_ℓ	Rolling moment coeff.
C_n	Yaw moment coeff.
$C_{n_{\delta a}}$	Yaw moment due to aileron δ
δ_d	Aileron droop
$C_{n_{\delta r}}$	Yaw moment due to rudder defl.

SYMBOLS (Continued)

V_T	Velocity, true
δ_r	Rudder deflection
L_ϕ	Rate of change of rolling moment with respect to roll velocity
TAS	True airspeed, knots
V_e	Equivalent airspeed feet/sec.
V	True airspeed along flight path, knots
V_L	Structural limit speed
V_S	Stalling speed, knots
α	Angle of attack, degrees
β_v	Wing fan louver angle, degrees
Δ	Denotes an increment
δ_f	Flaps, angle of deflection, degrees
CAS	calibrated airspeed
KIAS	indicated airspeed, knots
FPS	Feet per second
$C_{L_{CM}}$	Lift coefficient, complete model
$C_{M_{CG}}$	Pitching moment coeff at CG
F_{r_p}	Force - rudder pedals
F_{S_A}	Force - stick aileron
β_{S_L}	Beta stagger left
β_{S_R}	Beta stagger right
β_S	Beta stagger

SYMBOLS (Continued)

$C_{1/2}$ Cycles to half amplitude

ϕ Bank angle, degrees

$\frac{\phi}{v_e}$ Rolling parameter degrees/ft/sec

$$\frac{\phi}{v_e} = \frac{57.3}{V_e} \cdot \frac{\phi}{\beta}$$

1.0 INTRODUCTION

Theoretical studies, simulator evaluations, ground tests and flight tests were conducted to substantiate the flightworthiness of the XV-5A Research Vehicles. This report describes the work accomplished, the procedures followed to provide the substantiation; describes the development and design of the aircraft, and delineates operational reliability information.

Submittal of this report is made in accordance with Government Contract No. DA 44-177-TC-715.

The technical section of the report is divided into eight principal sections. These are:

3.0 Aircraft Performance
4.0 Strength Requirements and Compliance
5.0 Design and Construction - General
6.0 Propulsion System
7.0 Equipment
8.0 Operating Limitations and Information
9.0 Reliability Data
10.0 Components

Section 3.0 provides a discussion and substantiating curves of stalling speeds, takeoff, climb and landing performance, and the speed-altitude envelope. A VTOL single engine minimum recovery envelope is also presented.

Section 4.0 describes the structural design requirements, their suitability to this aircraft, and confirms that the aircraft meets requirements.

Section 5.0 discusses the general considerations of design and construction, and shows the application of acceptable aircraft practice for the choice and use of materials, manufacturing methods, and quality assurance.

The propulsion system discussed in Section 6.0 provides flightworthiness qualifying data, and lists the pertinent General Electric reports. The section describes the suitability, and the operating characteristics of the propulsion system, its accessories and subsystems.

Adequacy and flightworthiness of instruments, electrical system, hydraulic system, control system, stability augmentation system, cockpit environment, landing gear and specific safety provisions are shown in

Section 7.0. Presented are design philosophy, installed performance and the references, which substantiate the fact that these systems are safe and proper, and will provide aircraft dependability.

Section 8.0 summarizes, and provides a guide to other published data related to the operating limits of the XV-5A aircraft.

Reliability is discussed in Section 9.0. Applicable curves, tables and figures are presented.

Section 10.0 lists all parts which are not classified or "standard qualified", and discusses the acceptability of each such unqualified part for use in the XV-5A aircraft.

2.0 CONCLUSIONS

The data provided in this report indicate that the XV-5A aircraft is safe and airworthy. This conclusion ~~has been~~ substantiated by analysis, ground test and flight test.

The XV-5A is shown to be structurally sound and suitable for use in a flight test program of at least 250 hours. The airplane was manufactured to exacting aircraft standards in choice and use of materials, components and subsystems, and was manufactured and tested with strict quality control standards maintained. Safety and airworthiness of the XV-5A VTOL aircraft, using the lift fan concept, has been demonstrated.

Performance predictions were substantiated by test. Stalling speeds are slightly higher than predicted, but are sufficiently close to indicate correct predictions of speeds. Slight buffet occurs as a stall warning, but normal quick recovery results. Takeoff and climb performance indicate safe margins and stable flight. Landing characteristics are normal in CTOL. VTOL stability is good at all rates of descent. The aircraft flies with adequate control at the boundaries of the predicted speed-altitude envelope, through conversion, and at speeds higher and lower than conversion speed.

Flight tests indicate that controllability is adequate and in agreement with acceptable standards. Control is satisfactory in VTOL and CTOL throughout the flight envelope, and during ground roll and taxi. Flutter analysis, and experimental ground, wind tunnel and flight tests indicate that the aircraft is free of flutter within the prescribed flight envelope.

Reliability and failure analyses confirm that the overall failure pattern followed the typical failure incidence curve, and that early failure rates were reduced as "infant mortalities". The failure curve levelled off after the fourth reporting period. Total system failure rate (in terms of failures per hour of system time as defined in the report) reduced from an initial value of 5.3349, to between 2.0000 and 2.5000 for the later reporting periods during the flight test program.

3.0 AIRCRAFT PERFORMANCE

3.1 PERFORMANCE SUMMARY

3.1.1 Stalling Speeds

Predicted stalling speeds in both the conversion configuration and the conventional flaps-down configuration are presented in Reference 1, which indicates the thrust in pounds to determine the power-on stall speeds. These data were derived from wind tunnel tests of models simulating the specified configurations and are presented in Figures 1 through 3.

Flight tests obtained stall speeds at all conditions. Results of these tests indicate that the estimates are close to actual values. The indications are based on chase plane reported values, and are compared to estimated values in Figure 4.

Data obtained from the Ames full scale wind tunnel test facility indicate that the quoted stall speeds are somewhat conservative. The maximum lift coefficients and stall angles of attack obtained were greater than those used in the prediction of the stall speeds. This was an expected result, since the small scale data were not corrected for the benefits of increased size of the actual aircraft.

Comments concerning the controllability at and near the stall condition will be found in Section 3.2.4 of this report.

3.1.2 Takeoff Performance

Takeoff performance is presented in three modes, VTOL, STOL, and CTOL. (See Reference 2 for complete performance predictions).

VTOL Mode

Figure 5 is a plot of total trimmed lift vs. altitude for standard and hot atmospheres and represents maximum available lift. Takeoff weights are obtained by dividing the quoted values by a factor to allow for control margin. A factor of 1.05 allows a 5 percent control margin, a factor of 1.10 allows a 10 percent margin, etc.

These data are based on Reference 3. This report presents the static performance of the lift and pitch fan systems as it pertains to the XV-5A installation. Average wing and pitch fan performance based on

Flightworthiness and Acceptance Tests of the fans was used in conjunction with minimum J-85 gas generator performance.

STOL Mode

Figure 6 is a plot of total distance over a 50-foot obstacle at two different altitudes for an ARDC Standard Day. These results were estimated using propulsion data based on the abovementioned General Electric memorandum, and aircraft characteristics derived from model wind tunnel tests.

No flight tests have been made in which minimum takeoff distances have been measured. Fan mode takeoffs and landings have been made, but not with the object of attaining maximum performance.

CTOL Mode

Figure 7 is a plot of total distance over a 50-foot obstacle for the same conditions as specified in the STOL take-off data. These results were estimated on minimum J-85 gas generator performance adjusted for installation losses, and aircraft characteristics derived from model wind tunnel tests.

No specific flight tests have been made in which takeoff distances have been measured. Flight test results do indicate that these predictions are reasonable.

3.1.3 Climb Performance

Figures 8 and 9 present altitude vs. maximum rate of climb, and the altitude vs. velocity for maximum rate of climb for the conventional flight configuration.

Estimated rates of climb were derived from J-85 gas generator minimum performance and wind tunnel test aircraft characteristics.

3.1.4 Landing Performance

Figure 10 presents the landing distance over a 50 foot obstacle in the conventional flight mode. Ground roll distance is also presented.

These data were estimated from wind tunnel aircraft characteristics and assume that the thrust spoilers balance exactly 100% of the idle thrust.

3.1.5 Speed-Altitude Envelope

Figure 11 presents the estimated speed-altitude envelope at four basic weights. These data were generated using minimum J-85 gas generator performance and tunnel test derived aircraft characteristics.

Figure 12 presents a comparison of the flight experience envelope of March 3, 1965, with the estimated speed-altitude envelope for the 10,000 pound gross weight aircraft in the clean configuration. The maximum power line, labeled as 102% RPM, was derived from the original engine specifications. The 102% RPM applies to the present rerated engines, as does also the point labeled 98% RPM. The respective power settings for the engine as originally rated are 100% and 96%. The point labeled 98% RPM (406 KIAS at 8,000 ft.), when compared with predicted values on the basis of equivalent engine ratings, agrees exactly.

Figures 13 and 14 show the flight experience envelopes in various flaps-down configurations with landing gear extended, and one with the flaps up, gear extended. Note that the preconversion configuration is one of the configurations presented.

3.1.6 VTOL Single Engine Minimum - Recovery Envelope.

The data of Figure 15 were derived from model wind tunnel tests and J-85 gas generator minimum performance. The information presented also was verified by test pilots flying the Ryan flight simulator. Flight test data have been obtained in fan mode using a two-engine power setting to simulate one engine at full power.

3.2 CONTROLLABILITY

Qualitative flight test data on a point check basis verified the estimated controllability and stability limits. The information contained in the following paragraphs is based on the analyses of References 4, 5, and 6 as well as on pertinent pilot comments.

3.2.1 VTOL and CTOL Controllability

3.2.1.1 VTOL

Investigation of control characteristics was conducted on the Ryan VTOL flight simulator, and is reported in detail by Reference 4. The flight simulator consisted of a cockpit mockup, a visual display for 6 degrees-of-freedom, actual electrical, mechanical, and hydraulic control systems of the airplane, and the necessary analog computer equipment.

Quantitative measurements of control positions were obtained with potentiometers. Airplane forces, moments, rates, positions, etc., were determined from the analog computer solution of the equations of motion. Cockpit control forces were measured by means of strain gages.

Longitudinal Control

Aircraft pitch angle response and requirements for one inch, and full control displacements are shown in Figure 16. The condition is for the specified pitch damping of Reference 7 and for a c.g. location at Station 243. The angular response requirements are exceeded for both the one inch and full control inputs. The available pitching moment from trim for full stick displacements is dependent upon c.g. position, since the nose fan is used for trim as well as for control.

The total pitching moment developed on the aircraft at zero angle of attack for neutral, full aft, and full forward longitudinal stick positions is shown in Figure 17. The available pitching moment from trim, for control in the nose down direction, is minimum in the speed range from 40 to 50 knots true airspeed, although the required 10% of the maximum attainable moment in hovering flight is exceeded.

Lateral Control

Aircraft response to roll control inputs is shown in Figure 18 as the roll angle achieved, after 1/2-second following the control input. Results are shown for both the basic aircraft with inherent fan damping only, and for the required roll damping level of -8750 ft. lb. /rad. /sec. The angular response requirements are exceeded for both 1 inch and full lateral stick inputs.

Directional Control

Yaw angular response to rudder pedal control inputs is given in Figure 19 for the basic aircraft and for the specified yaw damping of -25,000 ft. lb. /rad. /sec. The yaw angle produced by 1 inch rudder pedal displacements with the above damping is below the required 5.07° (by about 2°). The yaw angle obtained for full control inputs exceeds the requirement by approximately 5°.

3.2.1.2 CTOL

The data presented are the result of extensive analysis of small-scale and full-scale wind tunnel test data. The analysis is reported in Reference 5.

Longitudinal Control

Elevator effectiveness is presented in Figures 20, 21 and 22. The applicable c.g. range and Mach number range are noted.

Lateral Control

Aileron effectiveness and yawing moment due to aileron deflection characteristics are presented in Figures 23 and 24.

Directional Control

Figure 25 presents rudder effectiveness versus Mach number.

3.2.2 VTOL Trim

Hovering lateral translations to the left and right at various speeds are shown in Figures 26 and 27. The maximum translational speed attained during flight test was 16 knots, compared with 35 knots as specified by Reference 1, and was limited by a combination of the available roll control power and lateral speed stability, which were simulated.

3.2.3 VTOL and CTOL Stability

3.2.3.1 VTOL

Static stability is reported in detail in Reference 5 for both the fan mode and conventional flight mode. Static stability estimates are based on small-scale wind tunnel data alone.

Dynamic stability investigations are reported in Reference 6. These investigations used the Ryan VTOL flight simulator, which is described in Section 3.2.1.1.

Static Longitudinal

Estimated longitudinal static stability in the transition speed range is presented on Figure 28. While the absolute stability level is not well defined, the data indicate a destabilizing influence due to nose fan operation. The airplane is statically stable at thrust coefficients less than 0.92, which is equivalent to a flight speed of approximately 70 knots.

Dynamic Longitudinal

Longitudinal stick-fixed damping requirements in terms of period of oscillation and time to damp are shown in Figure 29. Some results are shown of the transient response of the aircraft to vertical gusts imposed on the flight simulator. A long peroid, well damped oscillation was apparent at 40 knots flight speed. At very low speeds, the oscillation was of similar period with nearly neutral damping.

Control system adjustability characteristics with respect to the hovering longitudinal control criteria outlined in Reference 7 are shown in Figure 30. The pointin the acceptable zone corresponds to the damping level specified in Reference 7, and also to the control sensitivity determined from the slope of the pitch control power curve through neutral longitudinal stick position. The point in the desirable zone at a damping-to-inertia ratio of 2.0, illustrates an arbitrary change in damping level obtainable from gain changes in the stability augmentation system. The damping moment available is not independent of control inputs, due to the limited authority of the stabilization system or, expressed another way, the limiting pitch rate below which the damping moment is linear varies inversely with the damping level. For example, for the damping-to-inertia ratio of 2.0, the stabilization system "saturates" at a pitch rate of approximately 9°/sec.

For the reasons discussed above, the terminal pitch angular velocity is undefined. The required pitch rate of 20°/sec. is the saturation rate for a damping level of 13,700 ft.lb./rad./sec.

Directional and Lateral

Steady sideslip angles at various transition speeds shown in Figures 31 through 34, show positive directional stability and dihedral effect for all of the speeds investigated. A maximum sideslip angle of 37° was obtained at 41 knots with less than 80% lateral control, but this angle is well beyond the wind tunnel test data used to define the lateral-directional stability characteristics of the aircraft. Maximum sideslip angle varied from 16° at 53 knots, to 9° at 96 knots. Sideslip angles were limited by roll control at 53 knots and by yaw control at 71 and 96 knots. Reasonably linear variations of both lateral-directional control positions and forces were obtained for all speeds.

Lateral control system characteristics (with respect to the control criteria of Figure 4 of Reference 7) are shown in Figure 35. The slope of the hovering roll acceleration curve through neutral lateral stick gives a control power to inertia ratio of .342, and the specified damping level of 8750 ft. lb./rad./sec. provides a damping-to-inertia ratio of 2.05 which falls within the acceptable zone of Figure 35. A value of damping-to-inertia ratio of 3.0 requires an increase in the augmented damping to 12,800 ft. lb./rad./sec. which is only 20% of the maximum stabilization system damping capability. At the damping level of 12,800 ft. lb./rad./sec., roll rates up to 11°/sec. result in linear damping moments with roll rate.

Control system adjustability in yaw requires adjustable cockpit control travel to provide variable control sensitivity. Yaw damping flexibility is provided by the stabilization system as for the pitch and roll axes.

The required rolling velocity in hovering flight of 30°/sec. was achieved with approximately 80% lateral stick displacement for the lateral control power simulated. In the case of yaw, assuming a maximum available yawing moment of 15,000 ft. lb., a yawing velocity of 50°/sec. requires reducing the specified yaw damping of -25,000 ft. lb./rad./sec. by about 30%.

3.2.3.2 CTOL

Static stability is reported in detail in Reference 5, and is based on small-scale wind tunnel data. Dynamic stability investigations utilized the Ryan VTOL flight simulator.

Static Longitudinal

The static longitudinal characteristics are indicated in Figures 36 through 40. These figures indicate that characteristics are satisfactory at all speeds up to Mach 0.8. Neutral static stability may be encountered above Mach 0.7 at lift coefficients corresponding to high normal load factors. Deterioration in high speed, static longitudinal stability with increasing lift coefficient, is gradual, except near Mach 0.8, where an abrupt pitch-up is anticipated at the higher attainable load factors at high altitude. Above Mach 0.8, the static stability is unsatisfactory and requires that extreme caution be exercised during flight investigations of high speed maneuvering characteristics, particularly at high altitudes or high normal load factors.

Dynamic Longitudinal

In the conventional, clean airplane configuration, the longitudinal short period mode meets the damping requirements of Reference 7 throughout the flight envelope as shown in Figure 41. The natural frequency of the short period mode is less than that required by the specification at 40,000 feet, and for speeds less than M = .75 at 30,000 feet, M = 0.60 at 20,000 feet and M = 0.30 at sea level. While the low natural frequency may be undesirable for a fighter-type aircraft, this characteristic, where it exists, should not affect the utility of the aircraft for its intended purpose, or require any unusual piloting techniques.

Freeing the controls reduces to a slight degree the speed-altitude range, wherein the short period requirements of Reference 7 are satisfied because of a small reduction in natural frequency and increase in damping ratio, as depicted in Figure 42.

The longitudinal dynamic stability characteristics in the conventional flight landing configuration are satisfactory for flight testing at all flight conditions. Static longitudinal stability becomes marginal at high angles of attack, but the flight characteristics are satisfactory, primarily due to high pitch damping.

Directional and Lateral

The dutch roll characteristics in the conventional flight landing configuration meet the requirements of Reference 7 at all speeds above approximately 120 knots at sea level. The dutch roll damping is estimated to be only slightly less than the requirement between 95 and 120 knots. These characteristics are indicated in Figure 43.

The characteristics of the lateral-directional oscillation, or dutch roll mode, in the clean airplane configuration, as indicated in Figure 44, meet the requirements of Reference 7 at all speeds from 15% above the stall speed, to Mach 0.8 at altitudes below about 25,000 feet. At altitudes from 25,000 to 40,000 feet, the requirements are satisfied for speeds above approximately Mach 0.7. At speeds below about Mach 0.6, and at altitudes above 25,000 feet, the relative magnitude of the rolling motion to sideslipping in the dutch roll mode increases with little change in damping as a result of increasing dihedral effect at high angles of attack. This characteristic is common at high altitude and low speed for aircraft without artificial damping, and is not expected to affect the utility of the aircraft for research purposes.

Aeroelastic and controls-free considerations had no significant effect on the dutch roll characteristics for any of the flight conditions investigated. This is shown in Figures 45 and 46.

The static and dynamic stability characteristics above Mach 0.8 up to the structural speed limit of Mach 0.9 are unsatisfactory, due to rising static longitudinal instability, rapid loss in pitch damping and rapid loss of control power about all three axes.

Pitch-yaw coupling may result in exceeding the vertical and lateral limit load factors during rapid, 360 degree rolling maneuvers at high speeds with rudder and elevator fixed. Prolonged rolling maneuvers with lateral control displacements up to one-half of full throw at dynamic pressures less than 250 to 300 pounds per square foot produce only small variations in load factor. The effects of pitch-yaw coupling at all flight conditions have not been investigated at the present time.

3.2.4 Stalls

From comments of pilots, it appears that there are no adverse stall characteristics. Stalls in straight, climbing and turning flight all exhibit the same characteristics. A dropping of the right wing at the stall is encountered, and recovery is normal. A light buffeting is encountered prior to the stall, which gives adequate warning.

3.2.5 Spinning

Spinning characteristics have not been investigated.

3.2.6 Ground Handling

Taxi control information consists of pilot comments both during and after flight. In general, the pilot reported excellent stability during both high and low speed taxi runs. The stiff nose wheel damping produced good longitudinal stability and the pilot reported no divergent directional oscillatory motions.

Since there is no nose wheel steering, the aircraft required more than average differential braking for maneuverability. Caution is required in using the brakes, as the airplane could spin on one main wheel. Excessive braking can cause overheating and brake fade if maximum continuous braking is employed to come to a full stop from 80 knots. Such a procedure will necessitate replacement of brake discs.

Most fade and overheat problems were due to residual thrust produced by the engines at idle power acting against the brakes. Residual thrust in the idle power position can propel the airplane at ground speeds up to

50 knots in a no-wind, no-brakes situation. Brake effectiveness was considered marginal, but satisfactory in terms of the intended use of the aircraft (i.e., essentially a prototype vertical flight research vehicle).

Thrust spoiler use to aid in decelerating the vehicle after landing or high speed taxi was initiated about half-way through the test program. The pilot reported excellent results; the airplane was easier to slow down, and the brakes remained much cooler. It is recommended that the spoilers be used on any long or high speed taxi runs to avoid rapid deterioration of the brake discs. The best procedure is intermittent operation of the spoilers to control desired maneuvering speed, with the brakes applied only as necessary.

Cross-wind taxi control was reported as satisfactory but with a weather-cocking tendency. This tendency was more severe with the landing gear in the VTOL position, but was not uncontrollable, even in a 20 knot cross-wind. Straight and level traverse was accomplished in a cross-wind by intermittent application of brakes and the use of rudder controls. Rudder deflection alone was sufficient to maintain directional control at speeds as low as 20 knots.

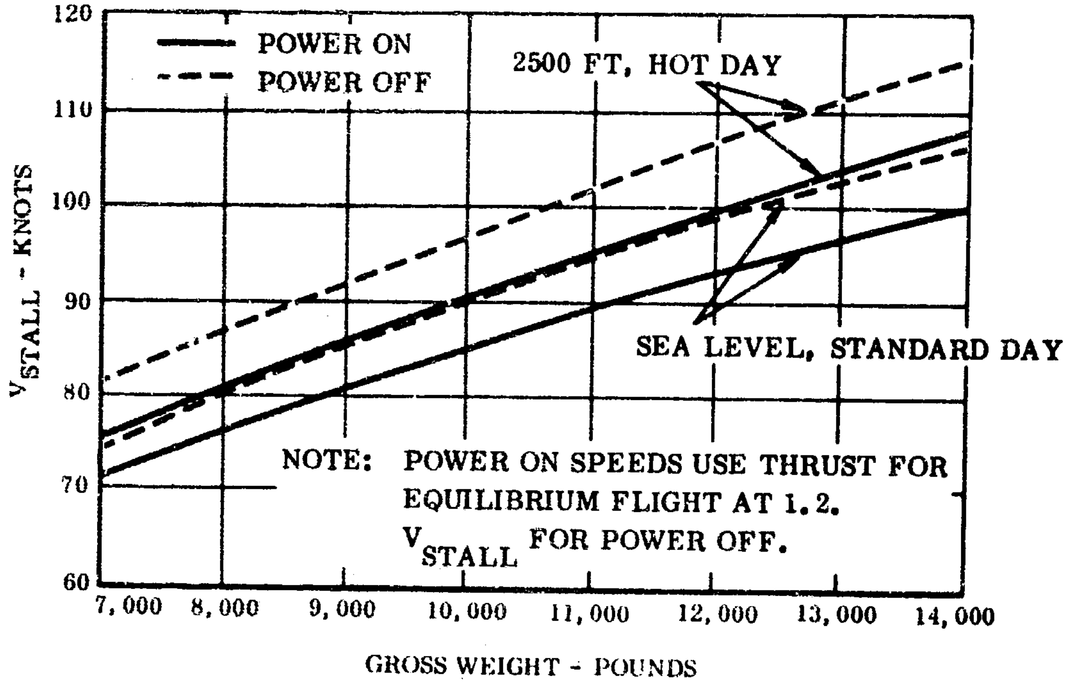


Figure 1 Stall Speed vs Gross Weight Conversion Configuration

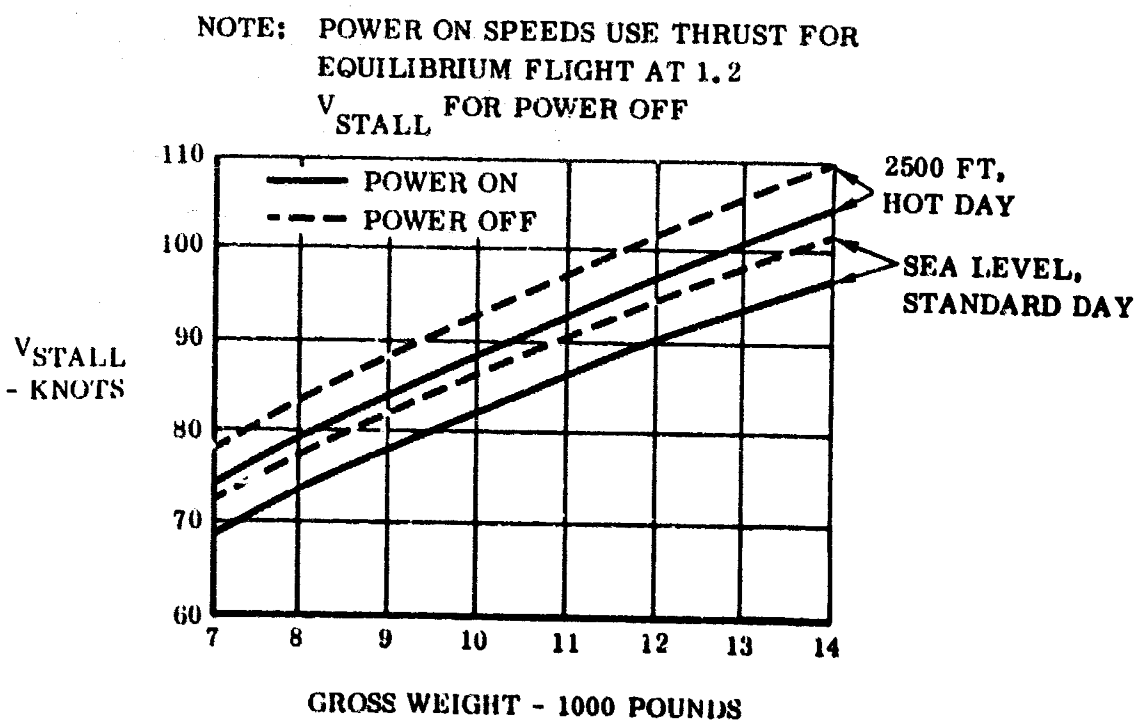


Figure 2 Stall Speed vs Gross Weight, Flap 45°, Ailerons 15°, Gear Down

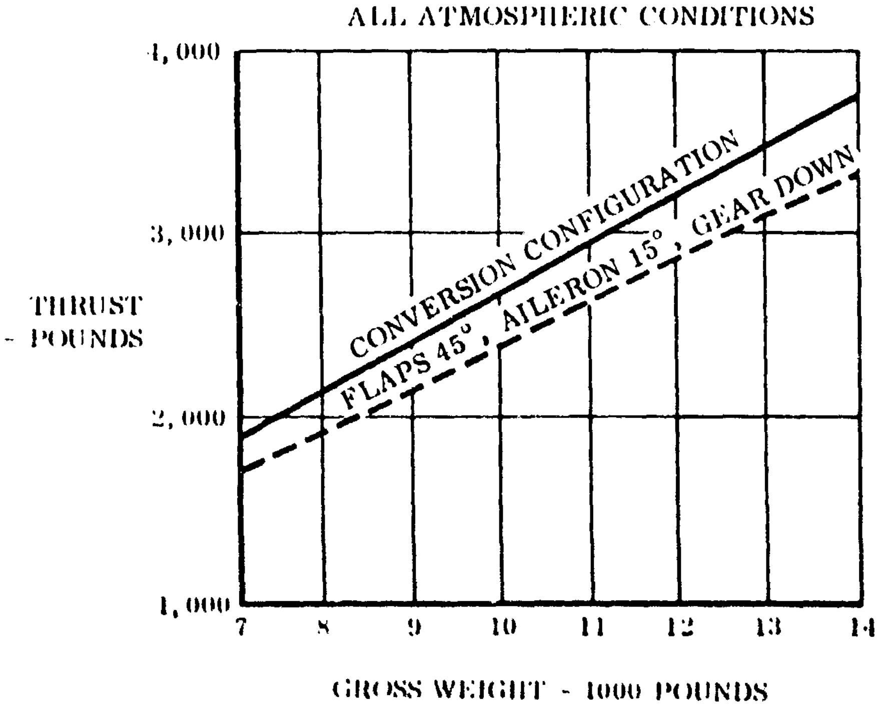


Figure 3 Thrust Required at 1.2 V_{Stall} (Power Off)

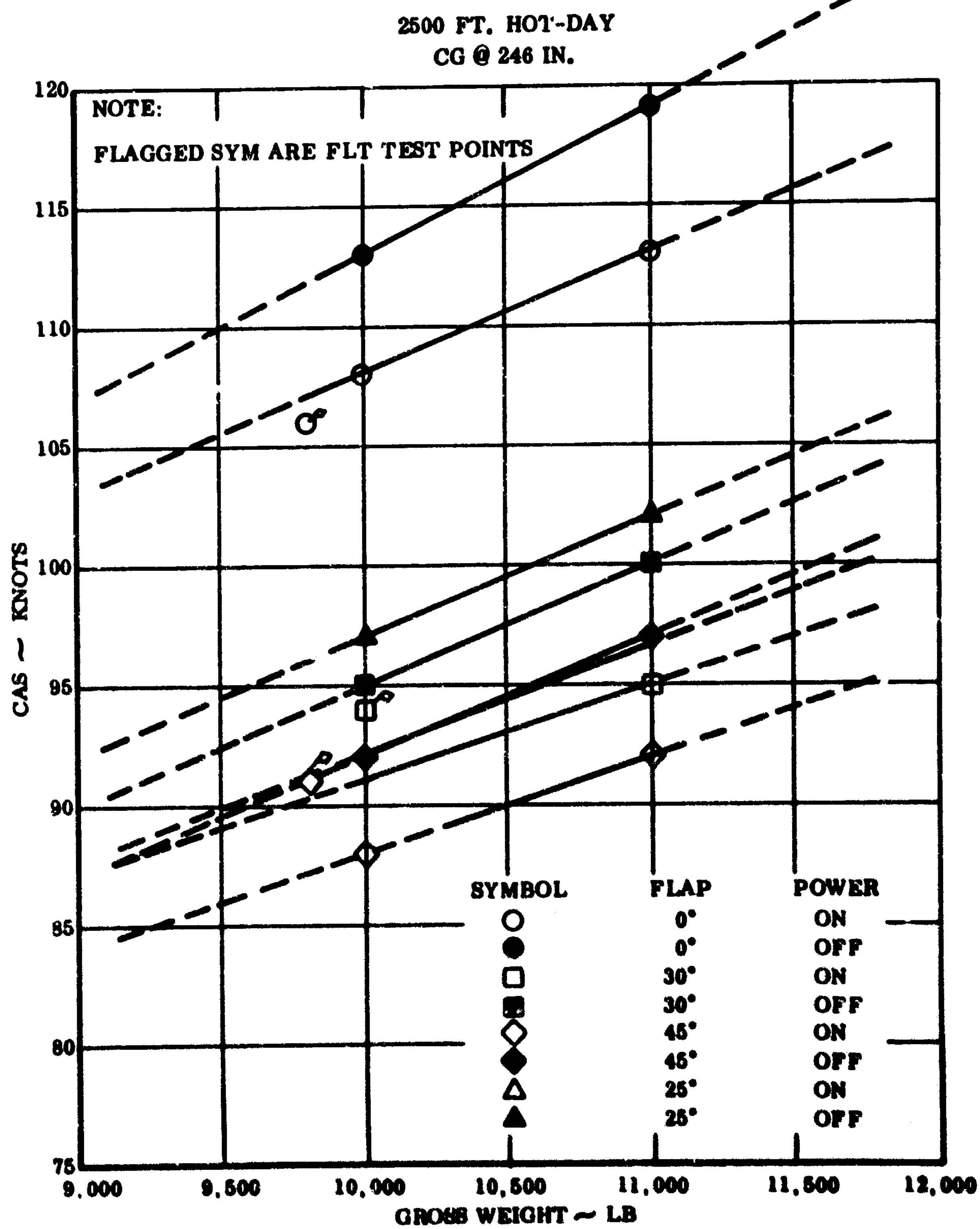


Figure 4 Flight Test and Estimated Stall Speeds, CTOL

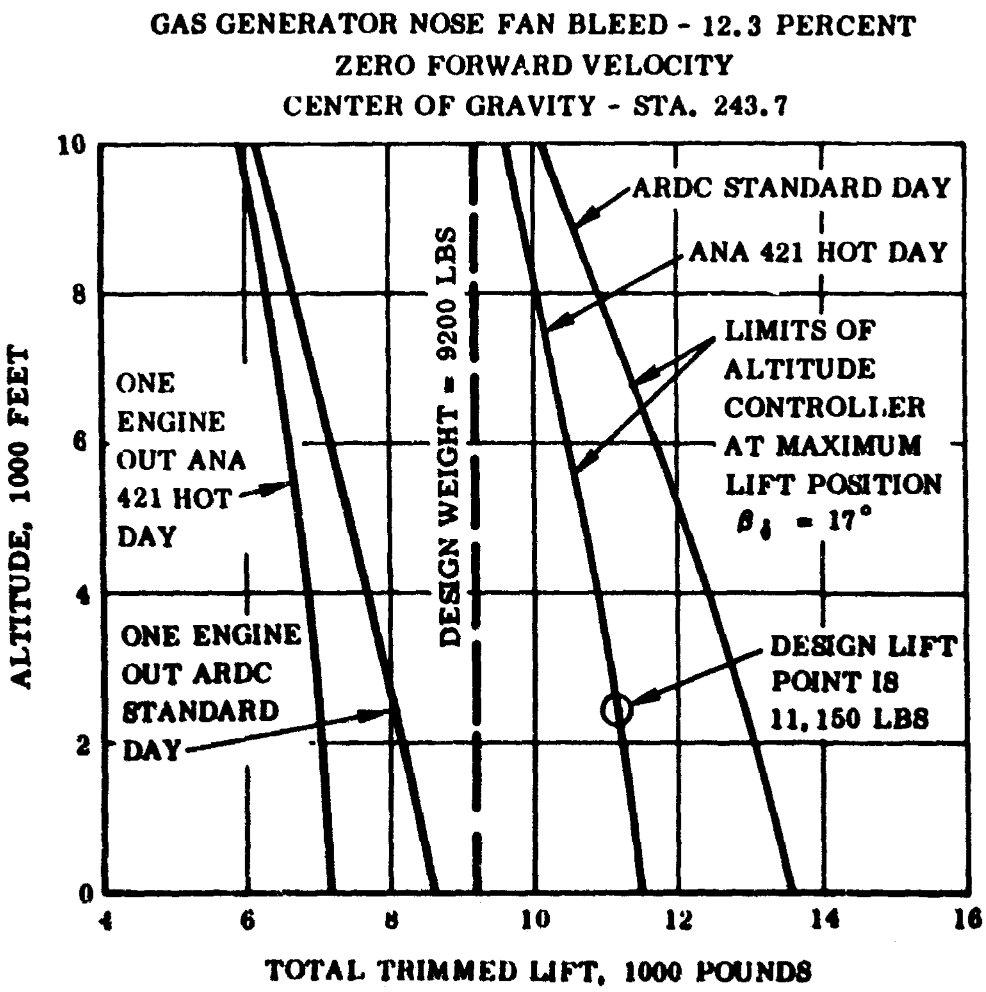


Figure 5 Total Trimmed Lift vs Altitude

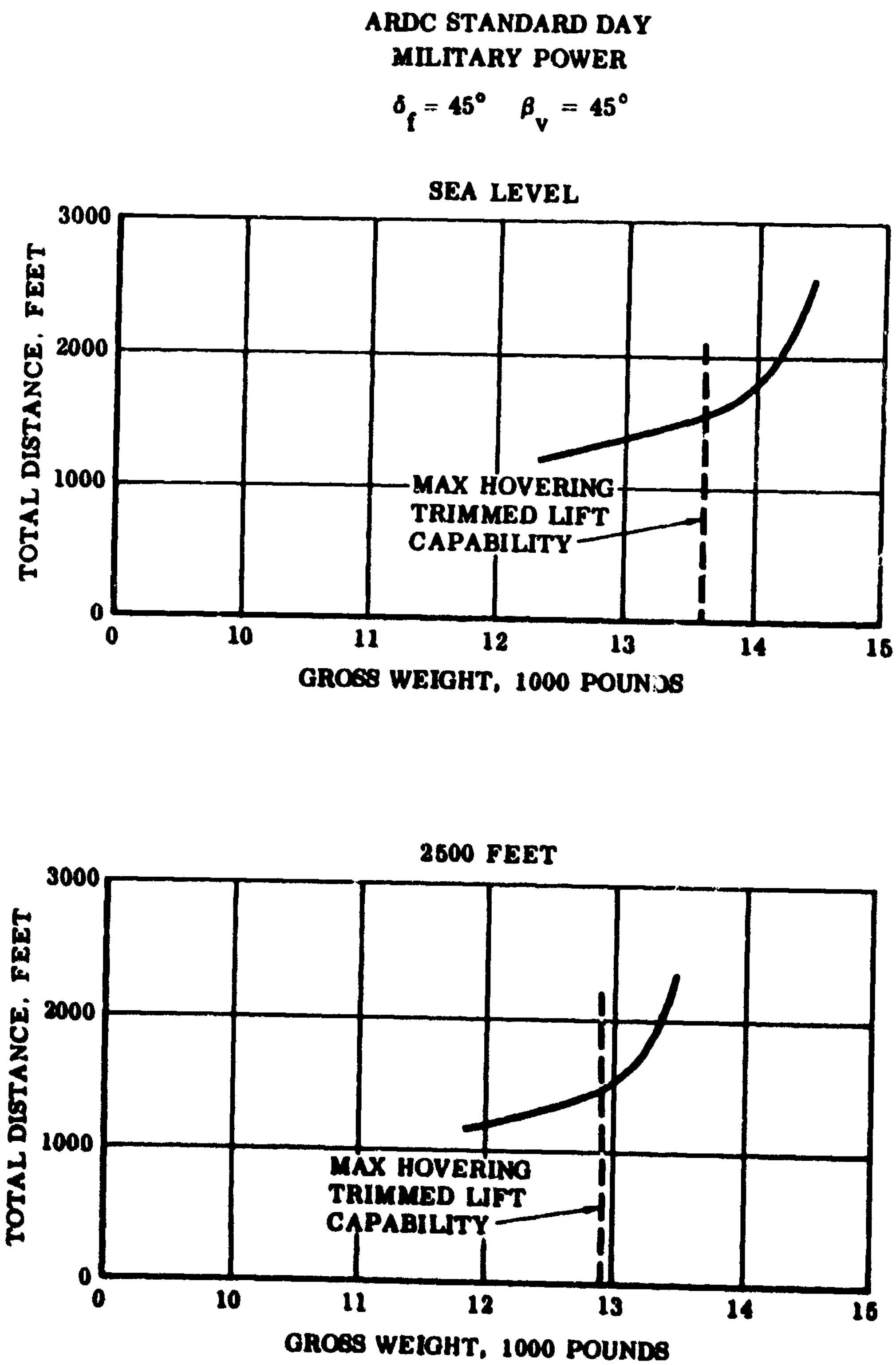


Figure 6 STOL Takeoff Distance Over 50 Foot Obstacle

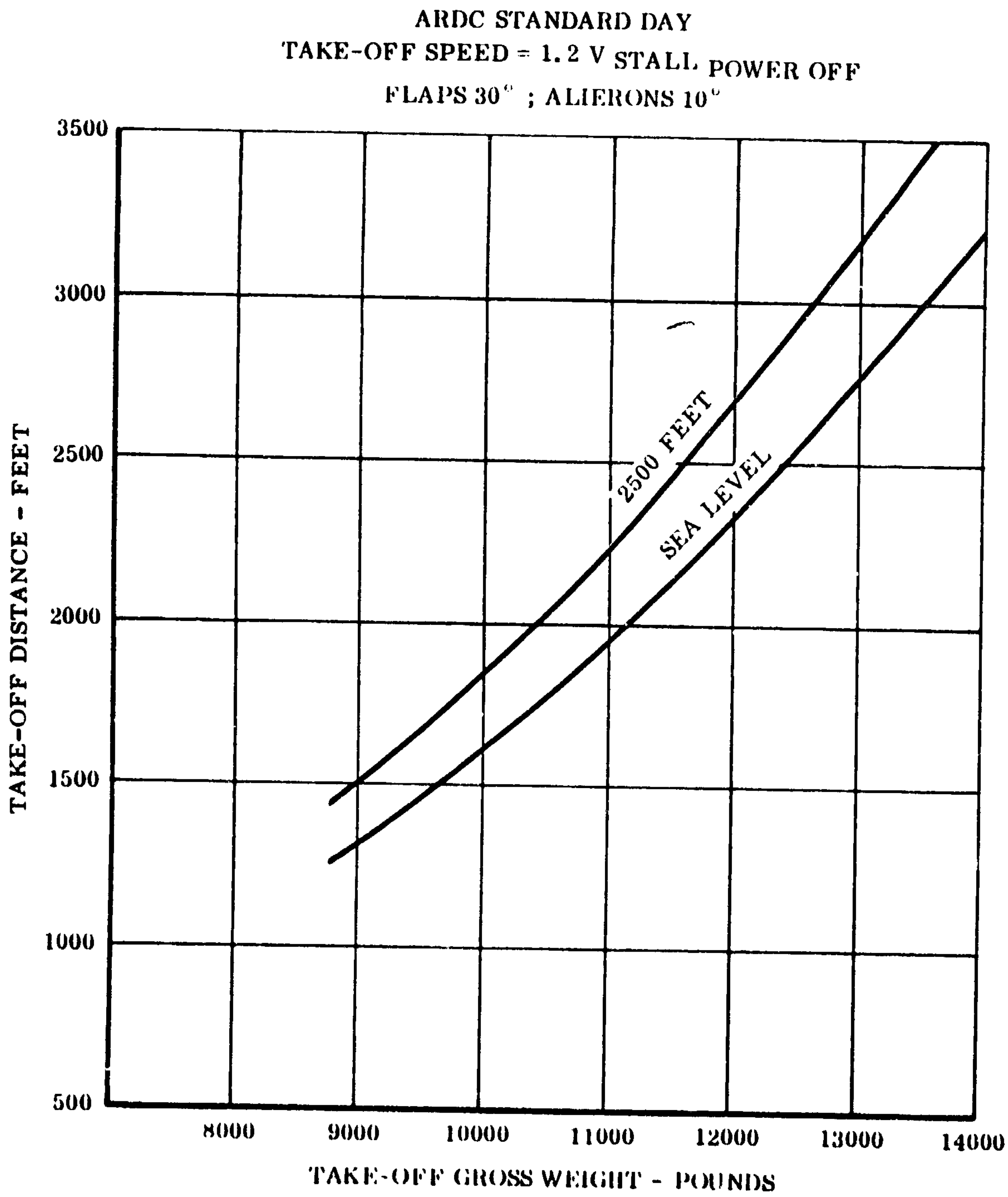


Figure 7 Conventional Takeoff Distance Over 50 Foot Obstacle

100% RPM OR TEMPERATURE LIMITED
ARDC STANDARD DAY

NOTE:
BELOW 10,000 FT. ENGINES AT 100% RPM.
ABOVE 10,000 FT., ENGINES ARE TEMPERATURE LIMITED.

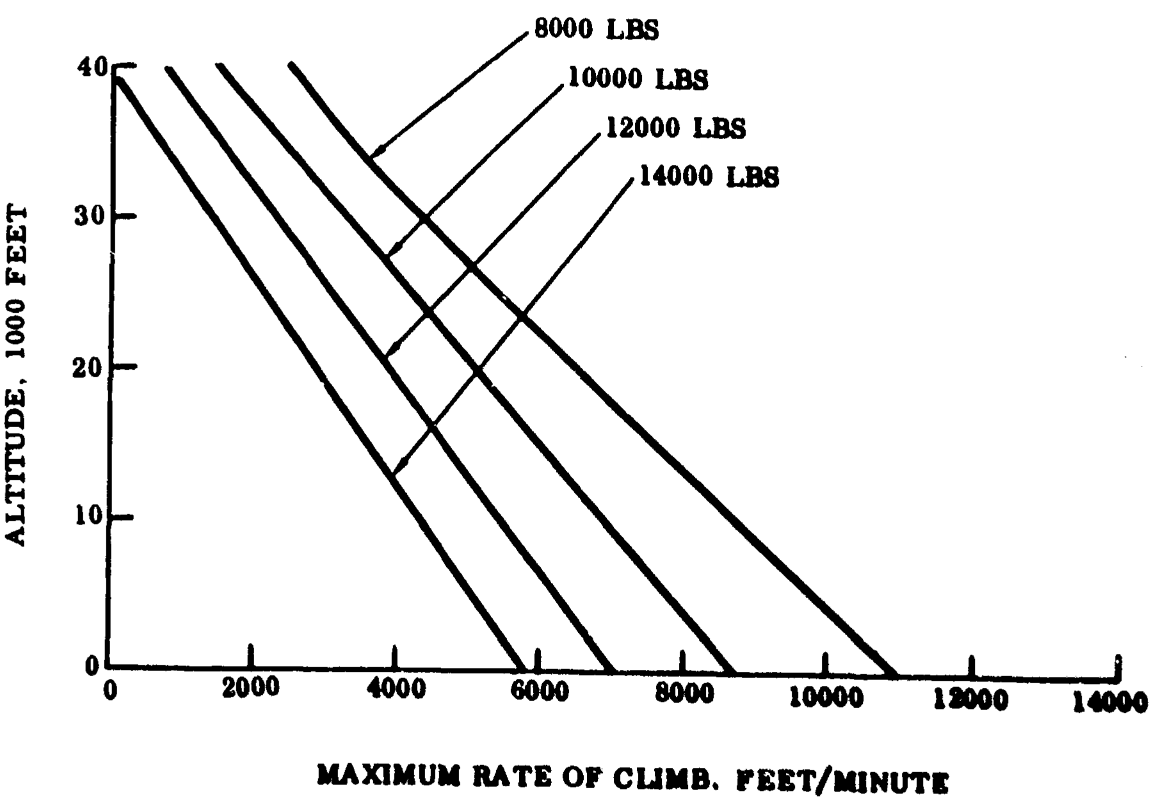


Figure 8 Altitude vs Maximum Rate of Climb

100% RPM OR TEMPERATURE LIMITED
ARDC STANDARD DAY

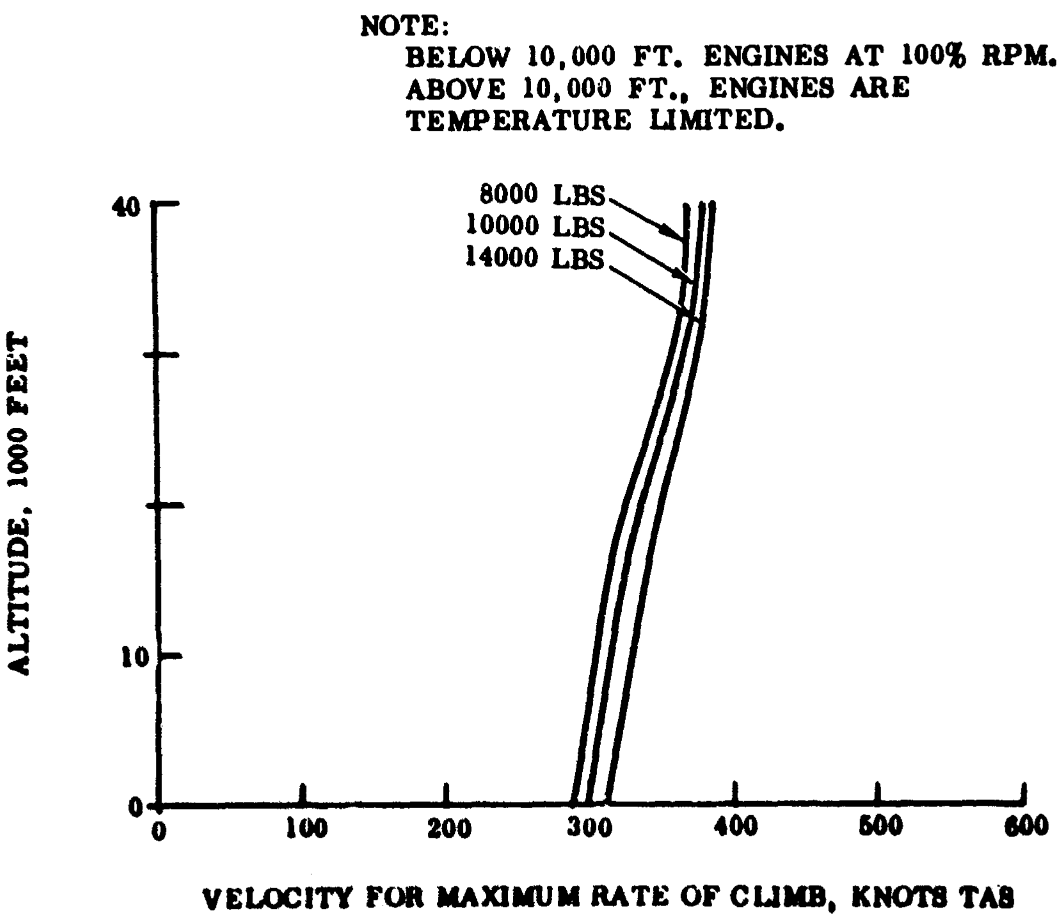
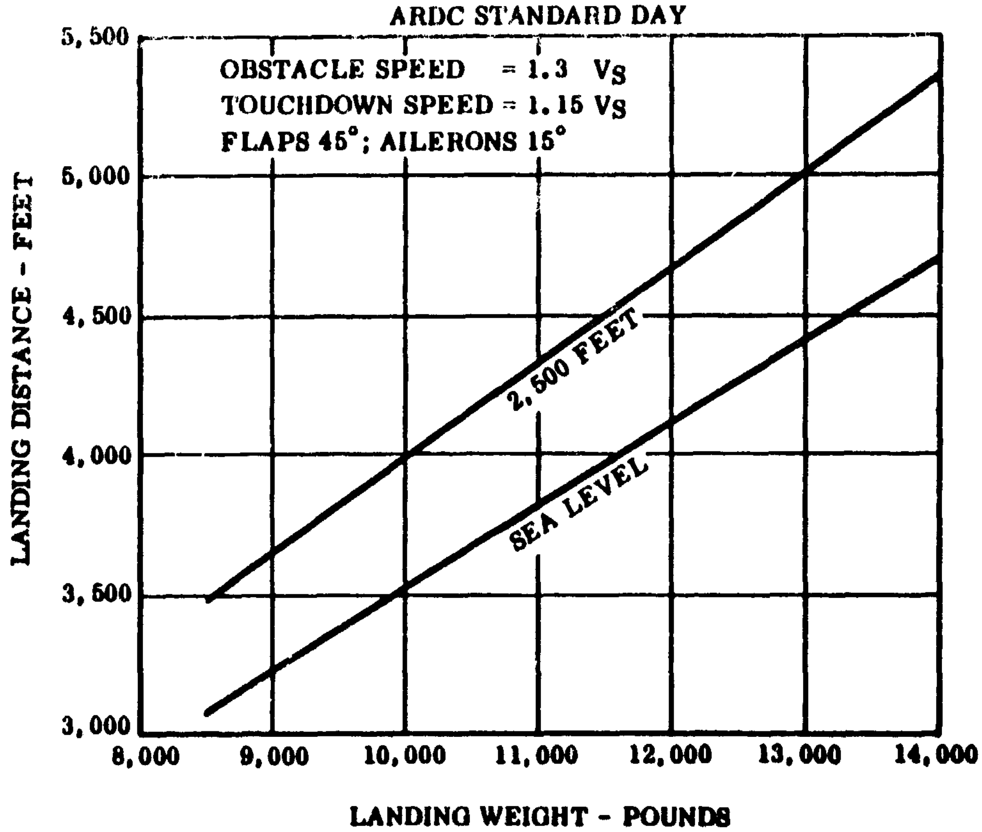


Figure 9 Altitude vs Velocity for Maximum Rate of Climb

ARDC STANDARD DAY



CONVENTIONAL LANDING GROUND ROLL DISTANCE
ARDC STANDARD DAY

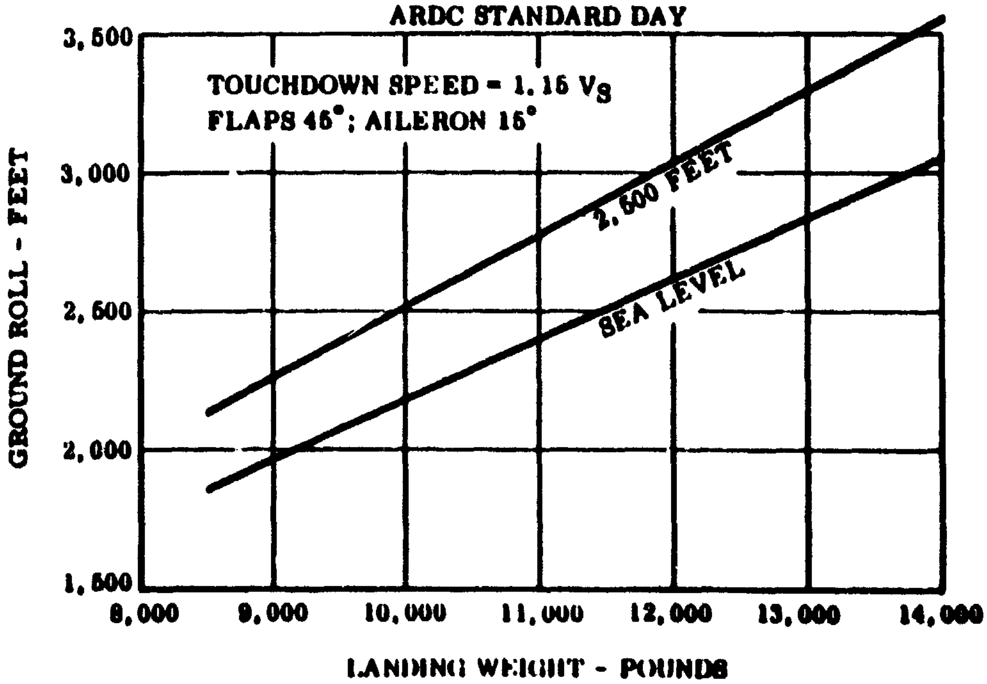


Figure 10 Conventional Landing Distance Over 50 Foot Obstacle

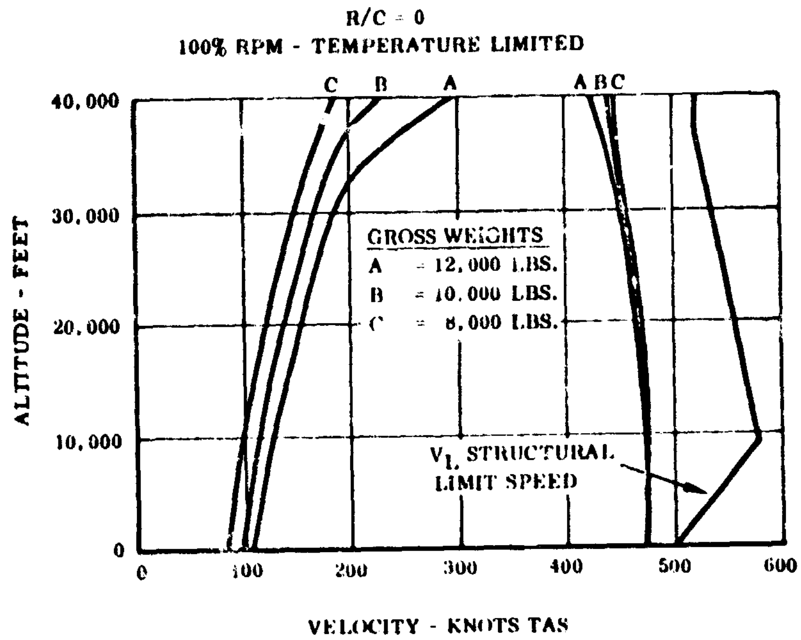


Figure 11 Speed-Altitude Envelope ARDC Standard Day

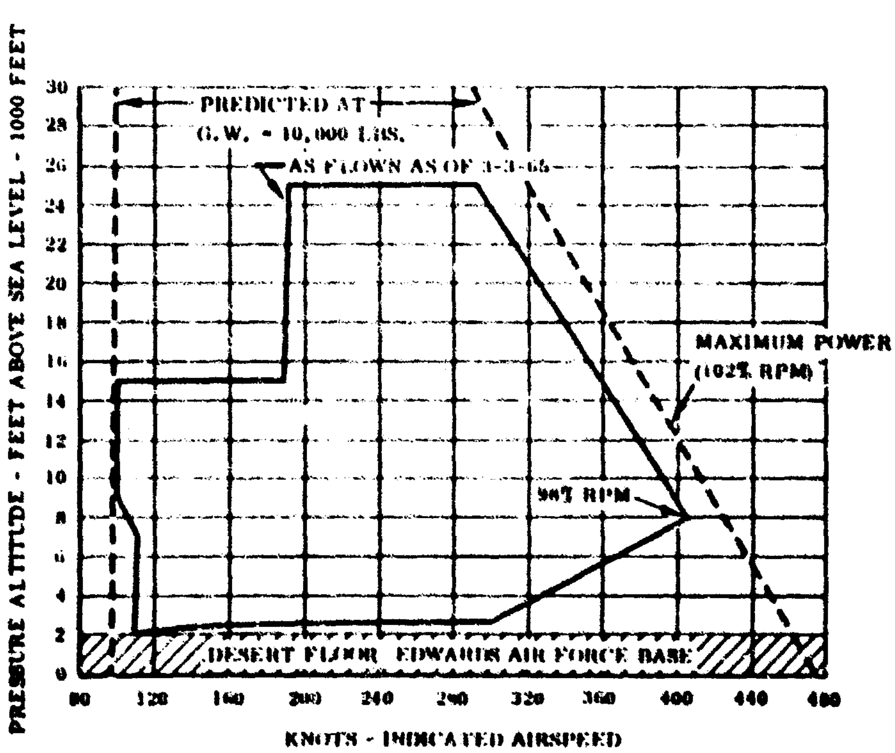


Figure 12 Flight Experience Envelope and Predicted Speed-Altitude Envelope Clean Configuration

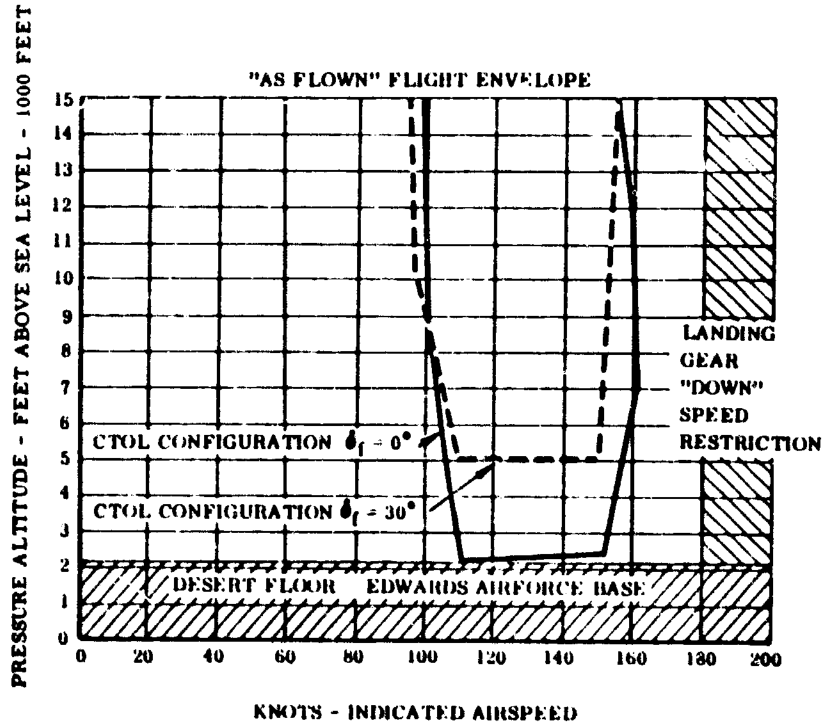


Figure 13 Flight Experience Envelope

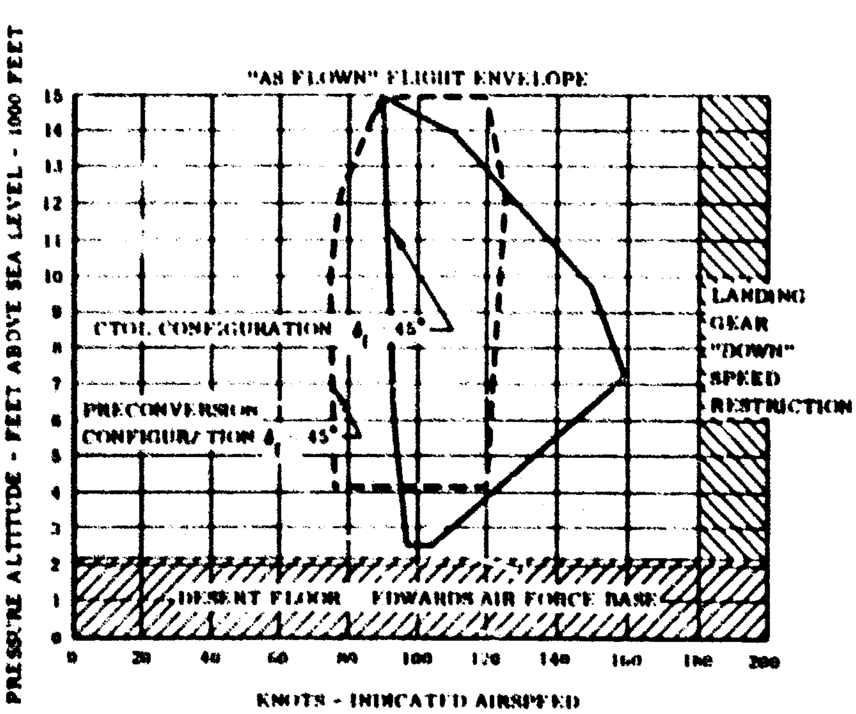


Figure 14 Flight Experience Envelope

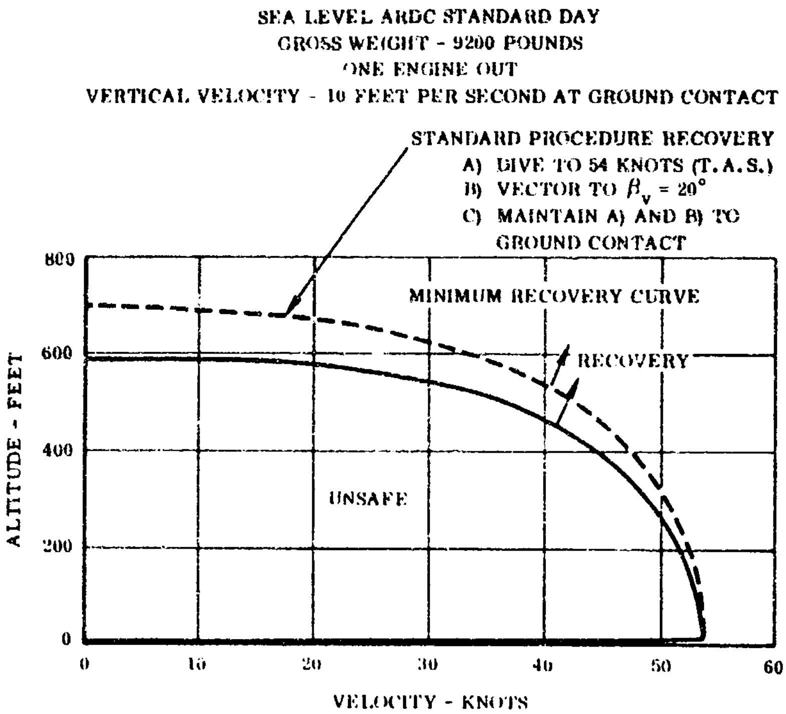


Figure 15 Single Engine Out Flight Recovery Envelope

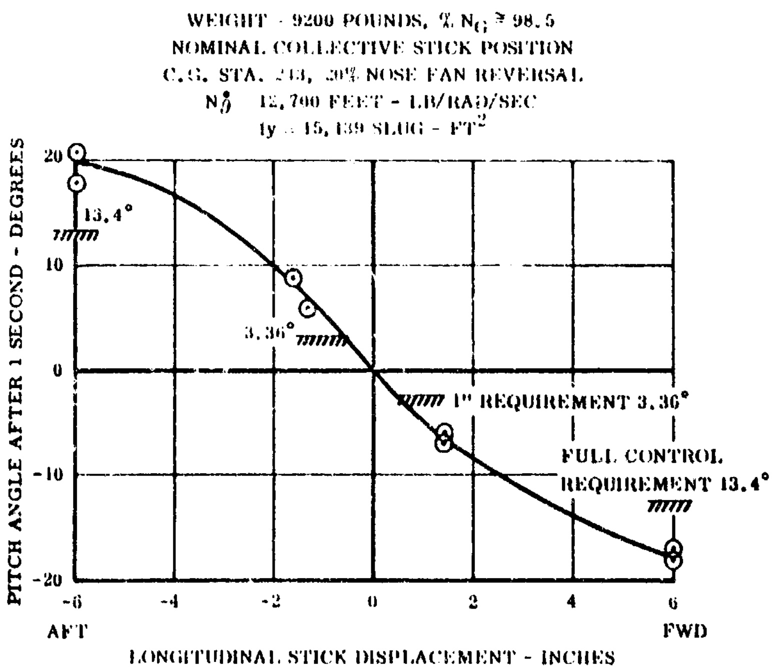


Figure 16 Pitch Angle Response

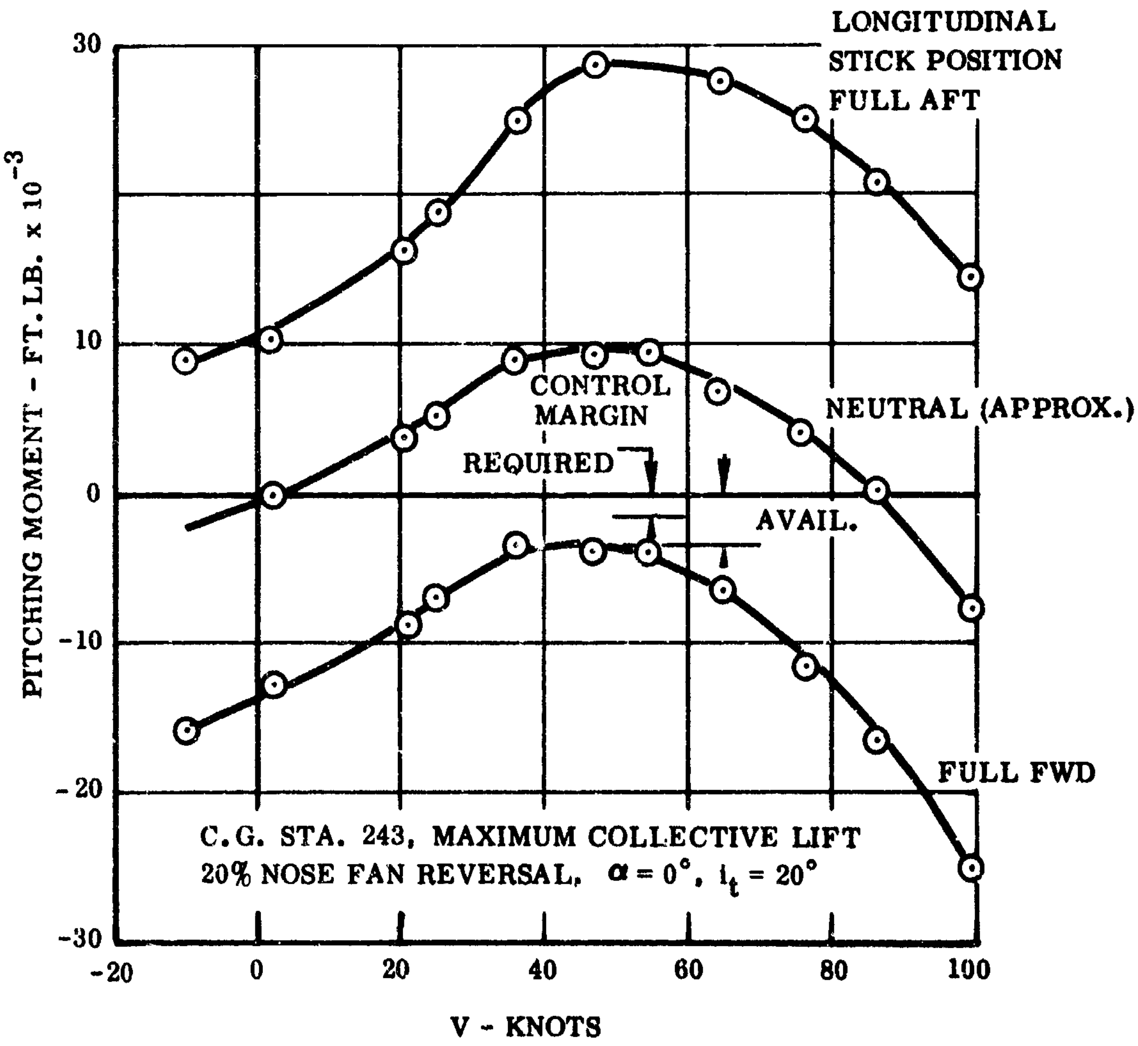


Figure 17 Total Pitching Moment

RESPONSE TO ROLL INPUTS

WEIGHT = 9200 LBS., I_x = 4254 SLUG FT^2
MID COLLECTIVE LIFT

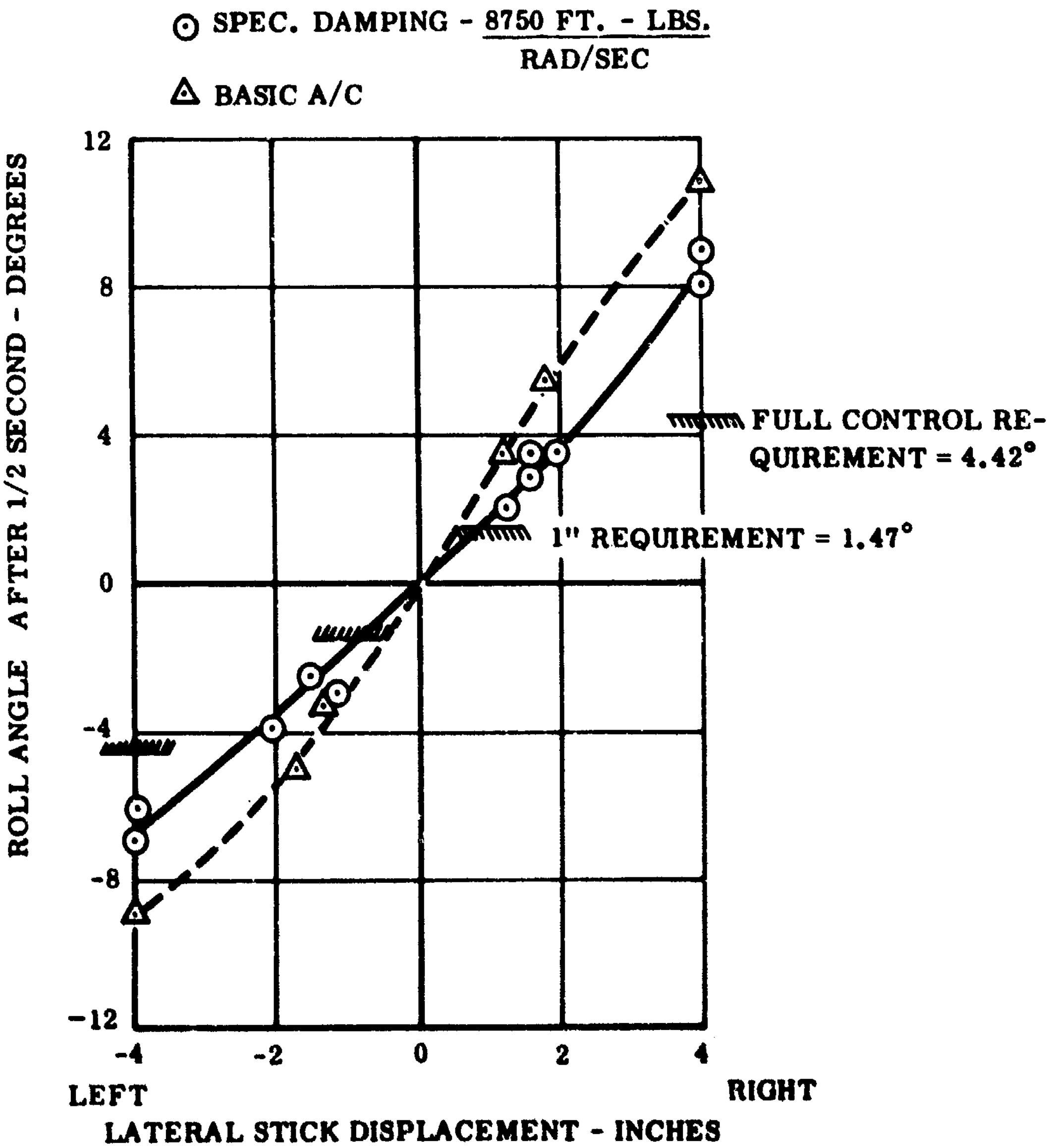


Figure 18 Roll Angle vs Lateral Stick Displacement

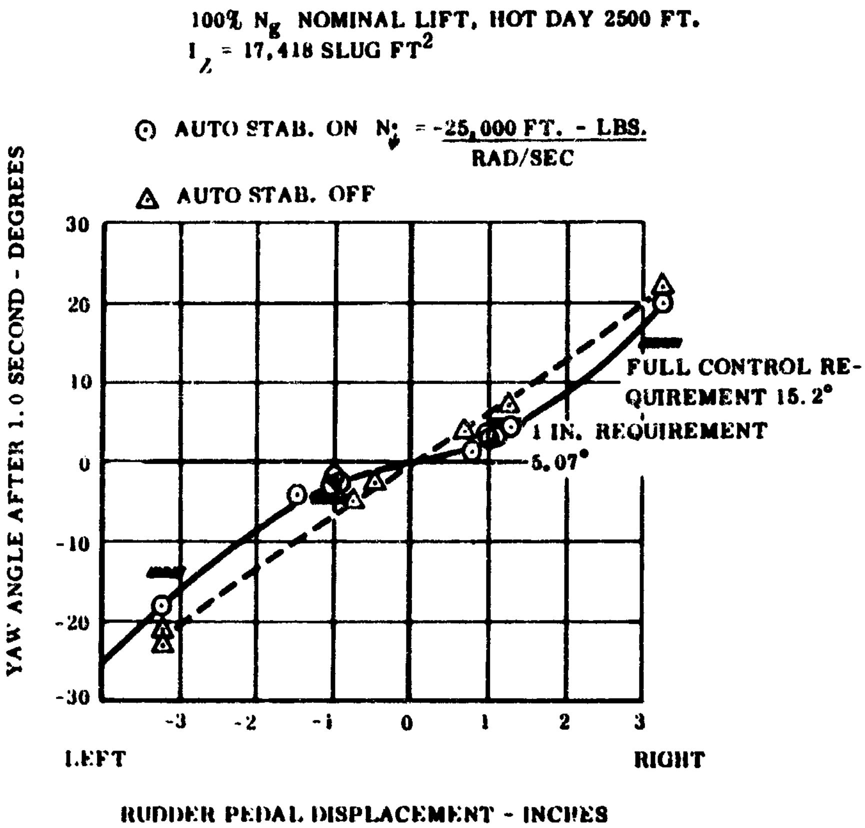


Figure 19 Hovering Directional Control Response

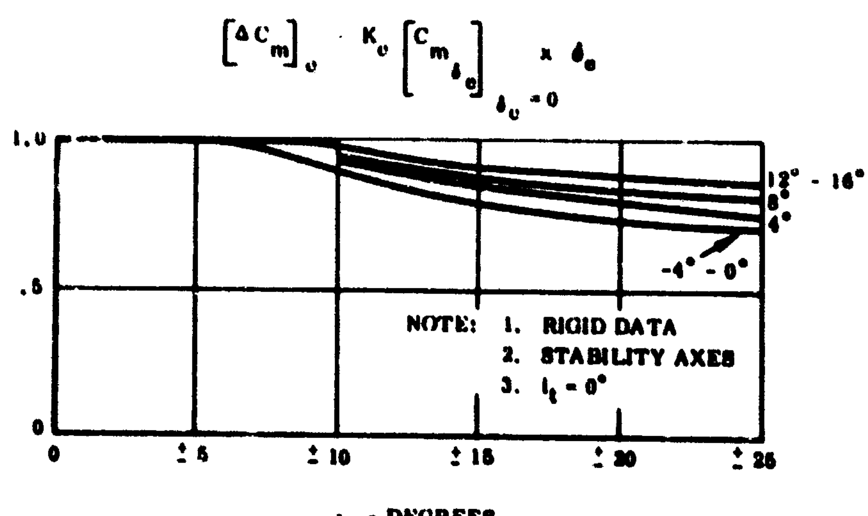


Figure 20 Elevator Effectiveness Parameter, Low Speed

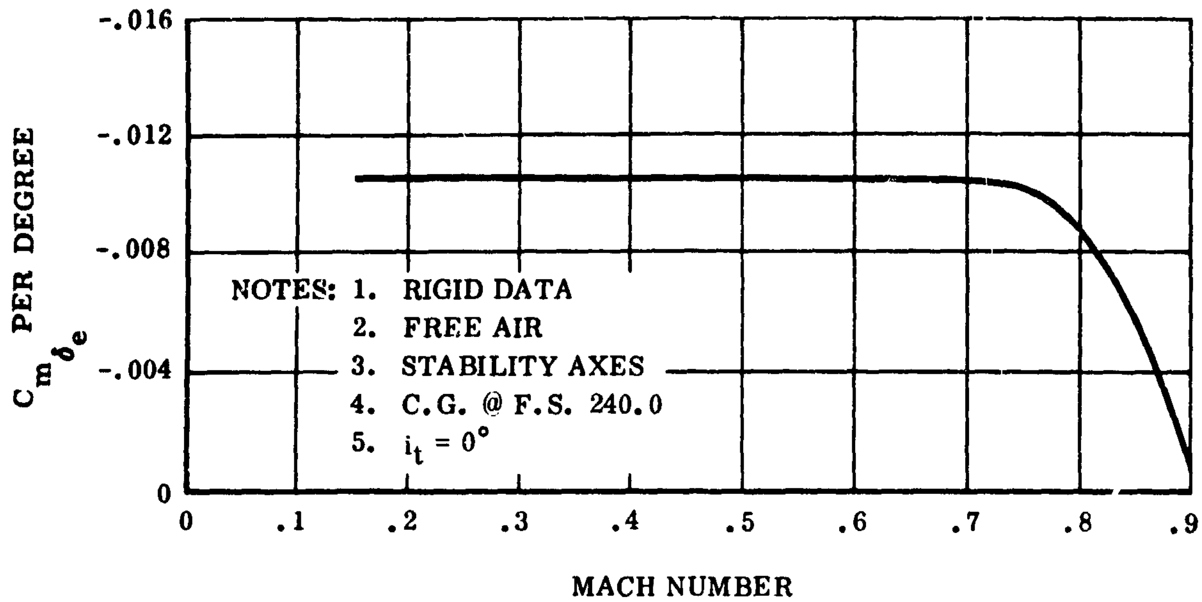


Figure 21 $C_{m_{\delta_e}}$ vs Mach Number

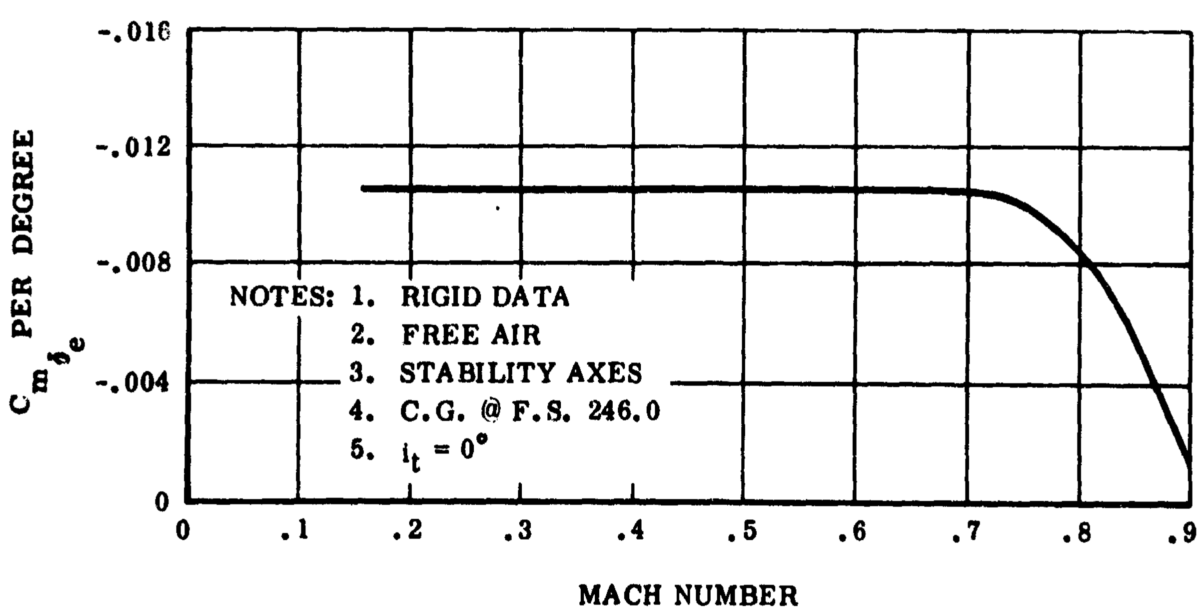


Figure 22 Elevator Effectiveness

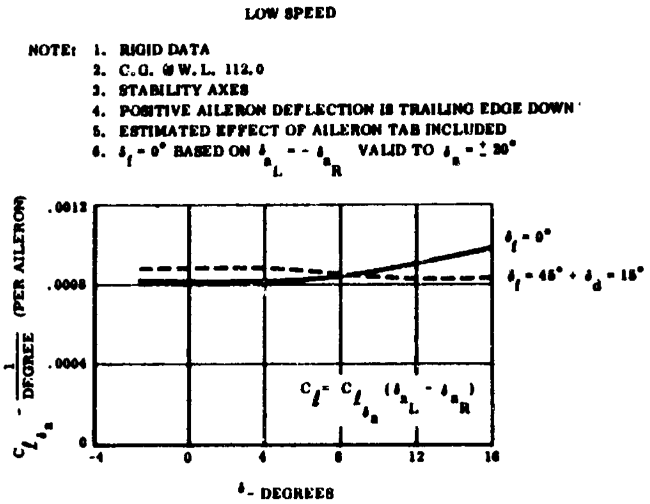


Figure 23 Rolling Moment Coefficient Due to Aileron Deflection

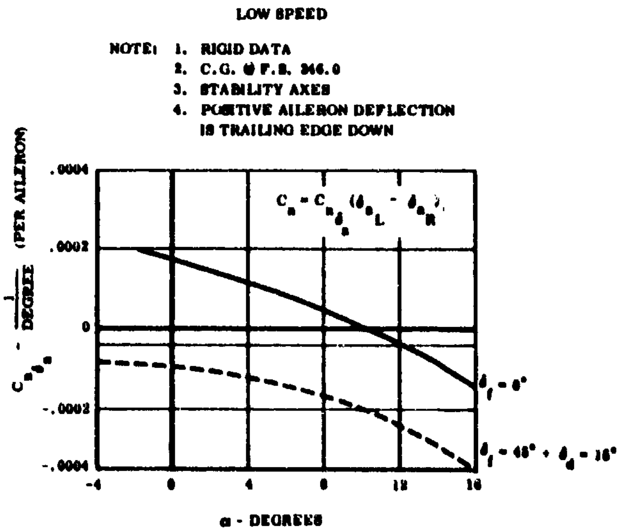


Figure 24 Yawing Moment Coefficient Due to Aileron Deflection

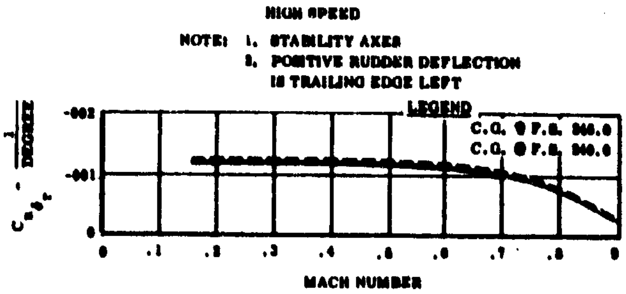


Figure 25 Yawing Moment Coefficient Due to Rudder Deflection - High Speed

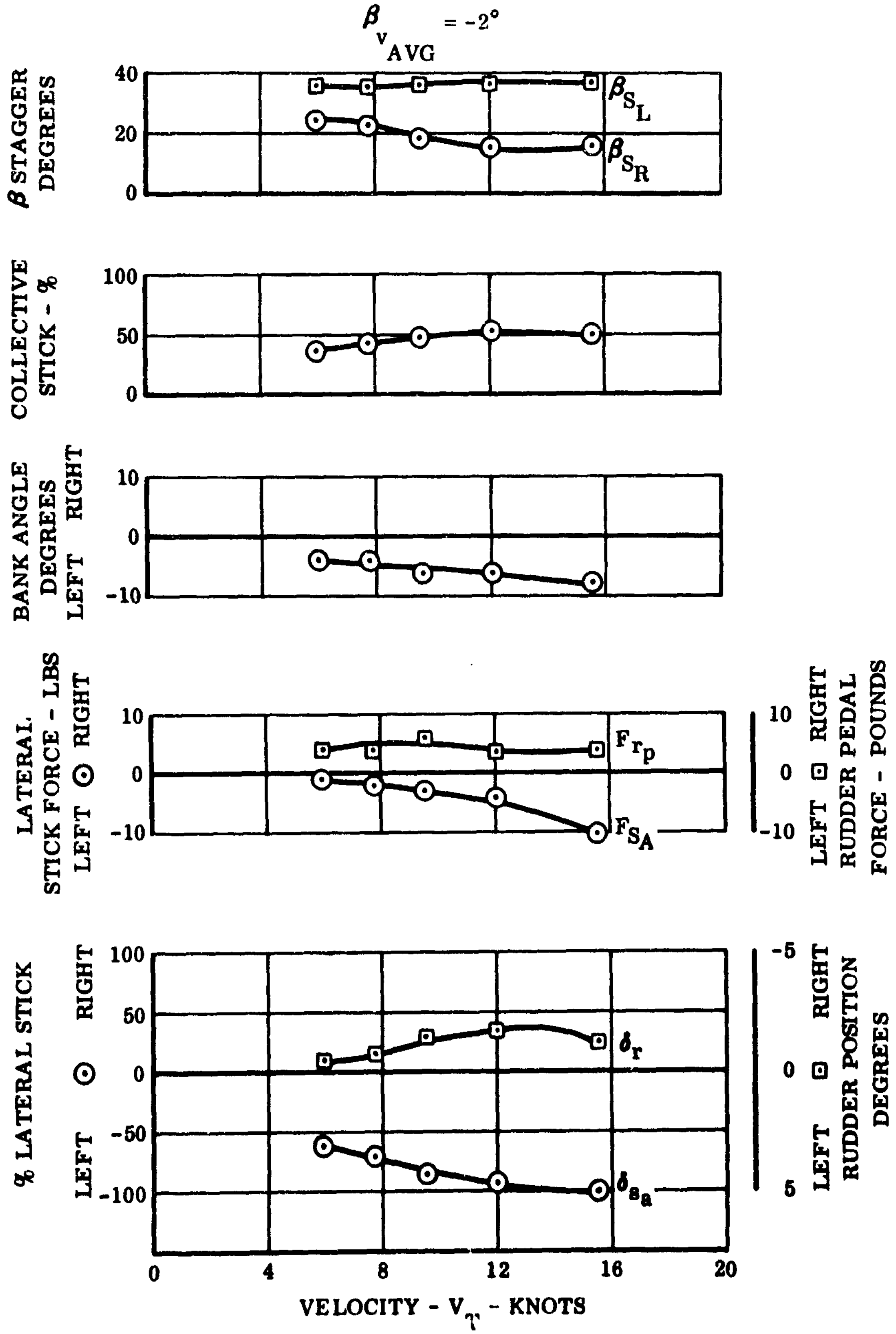


Figure 26 Hovering Lateral Translation

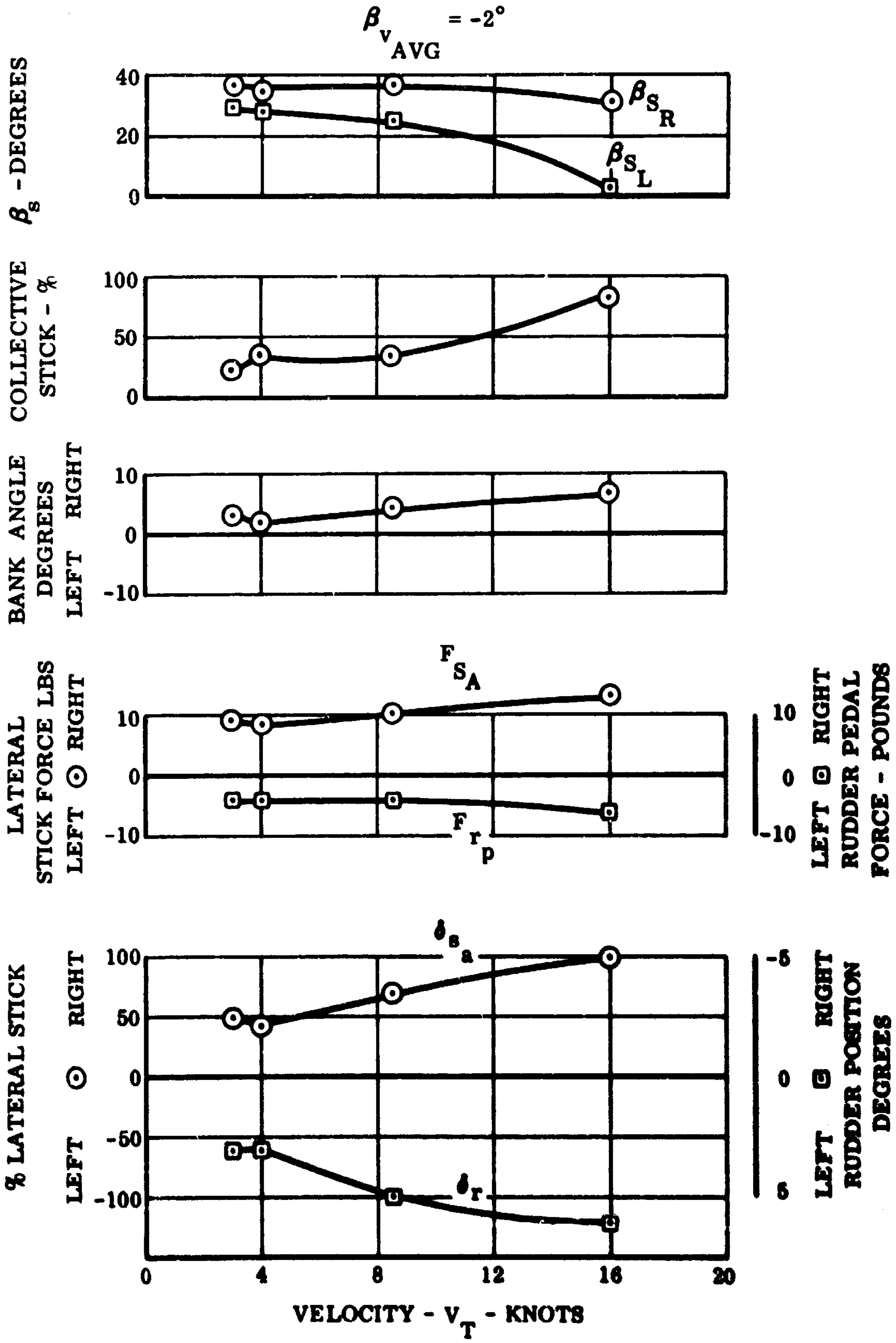


Figure 27 Hovering Lateral Translation

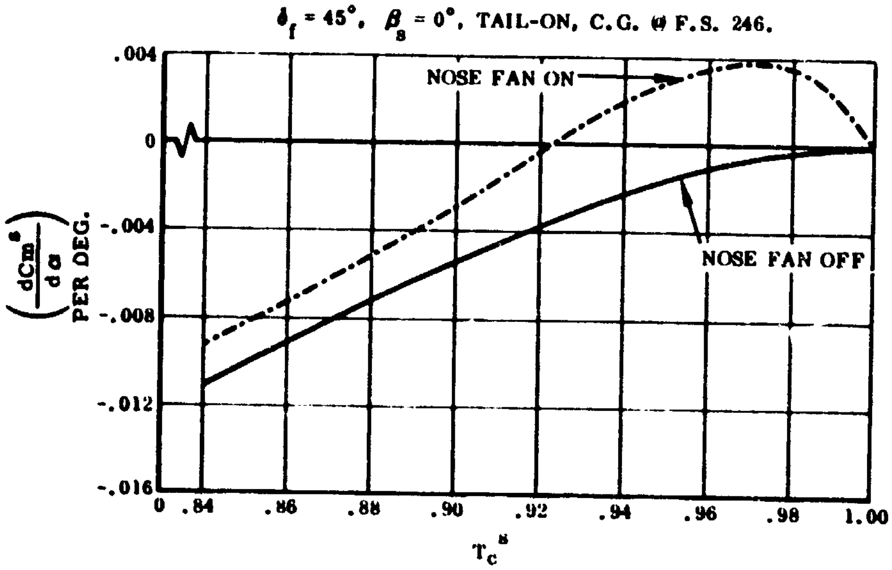


Figure 28 Estimated Longitudinal Static Stability in Transition Speed Range

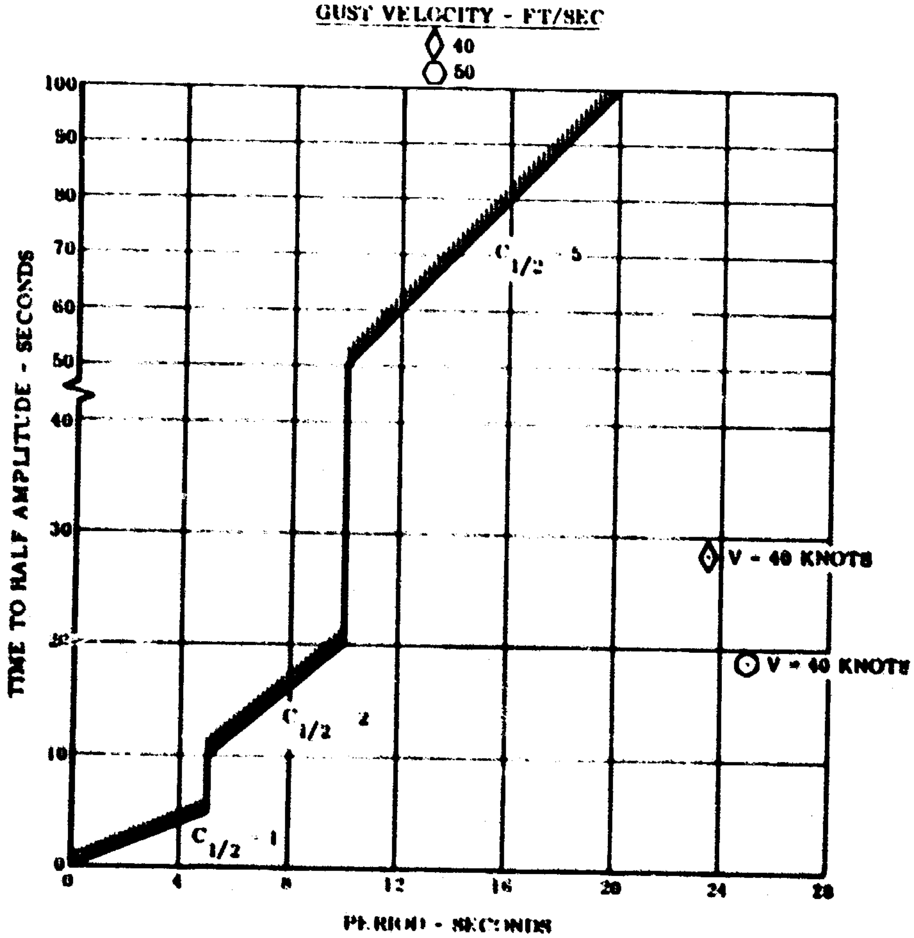


Figure 29 Longitudinal Damping Requirements

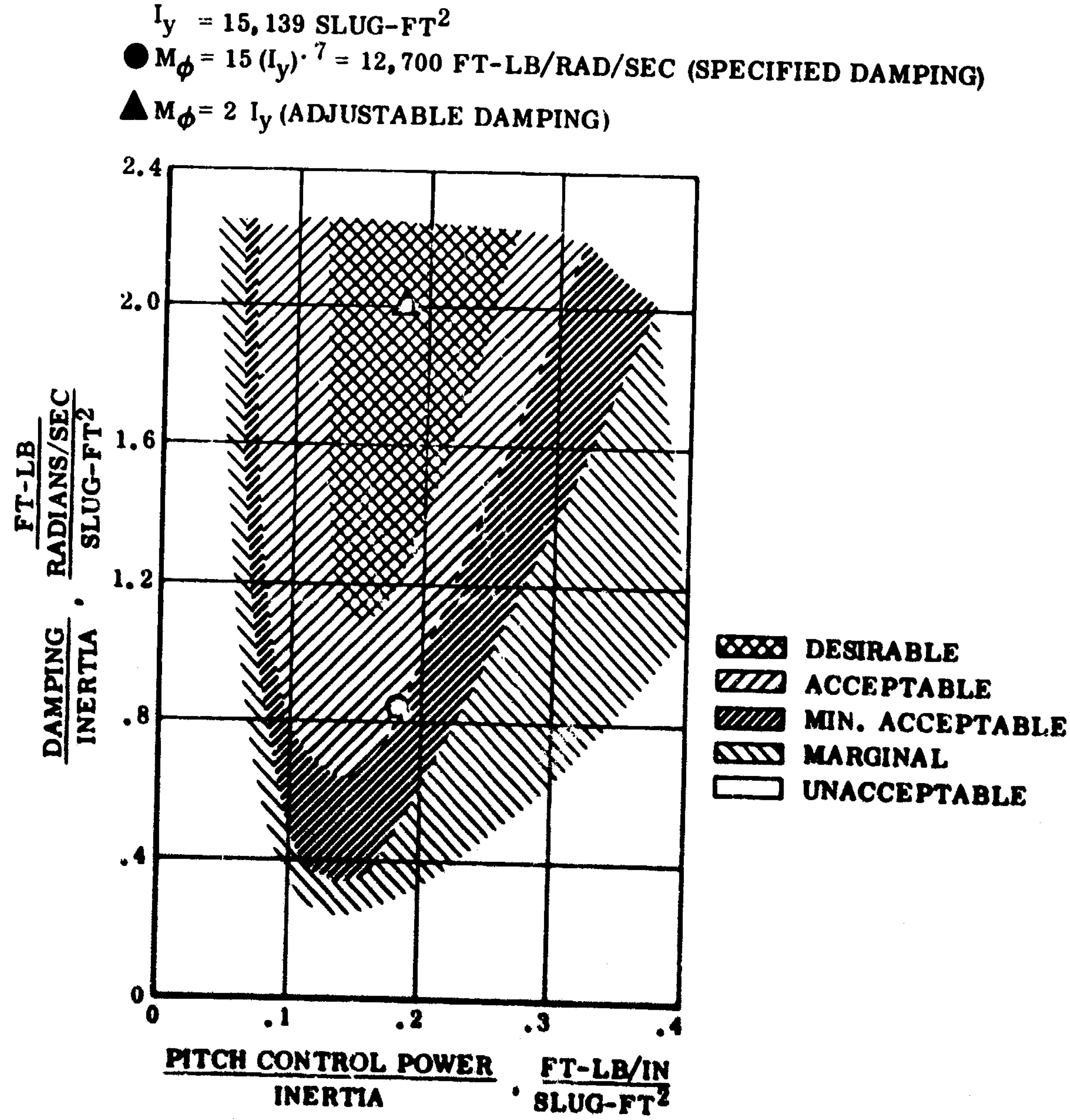


Figure 30 Control System Adaptability Characteristics

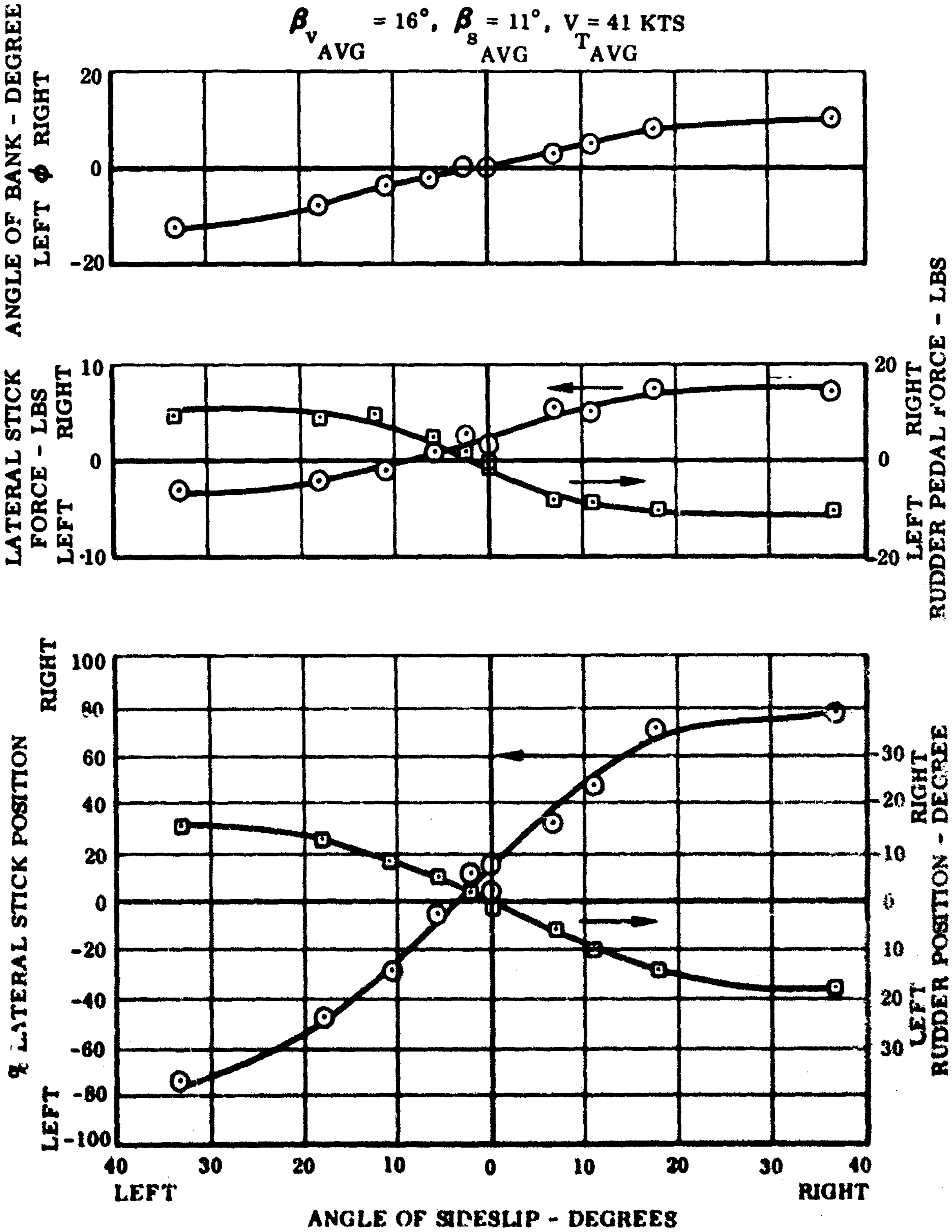


Figure 31 Steady State Sideslips

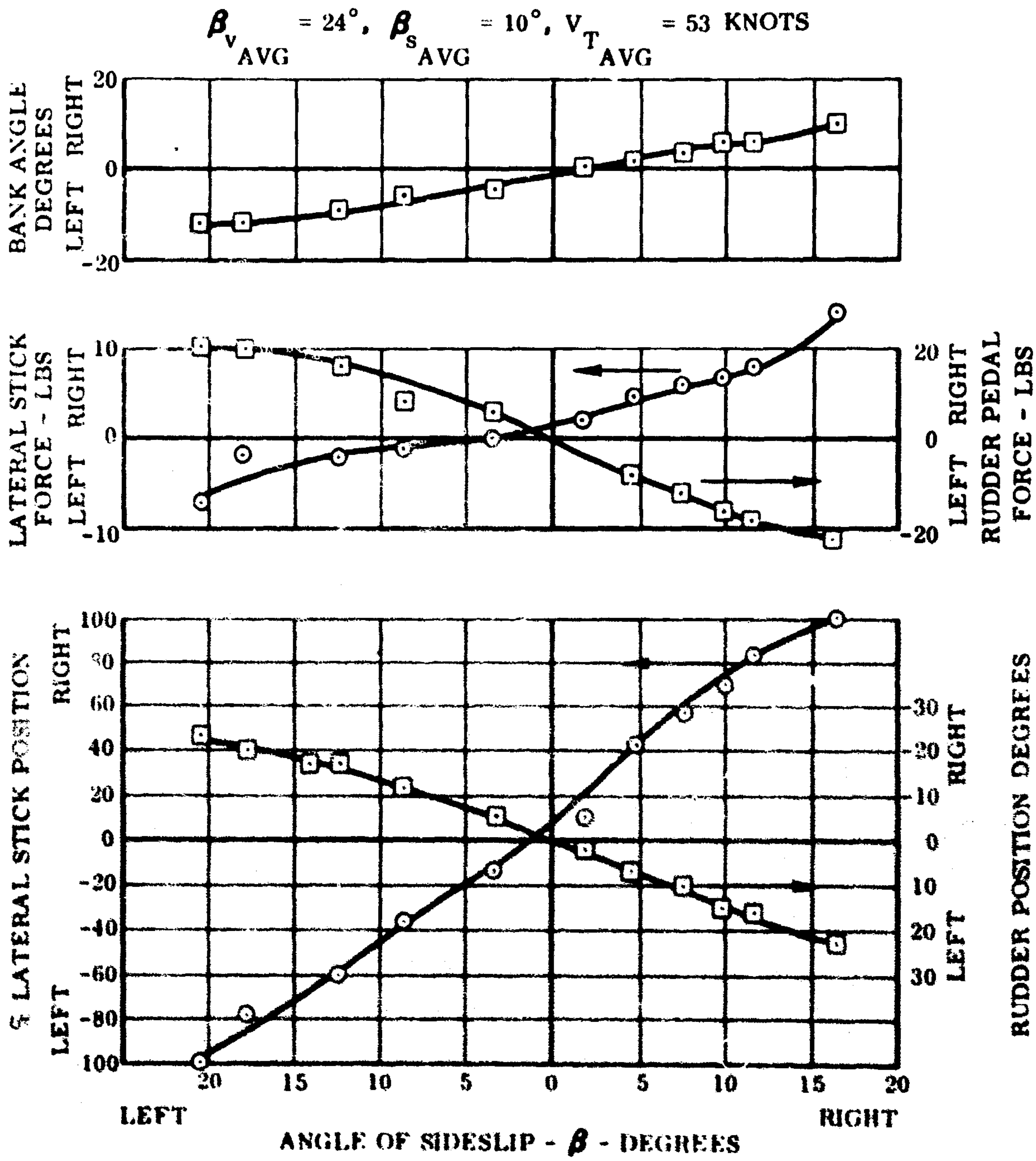


Figure 32 Steady State Sideslips

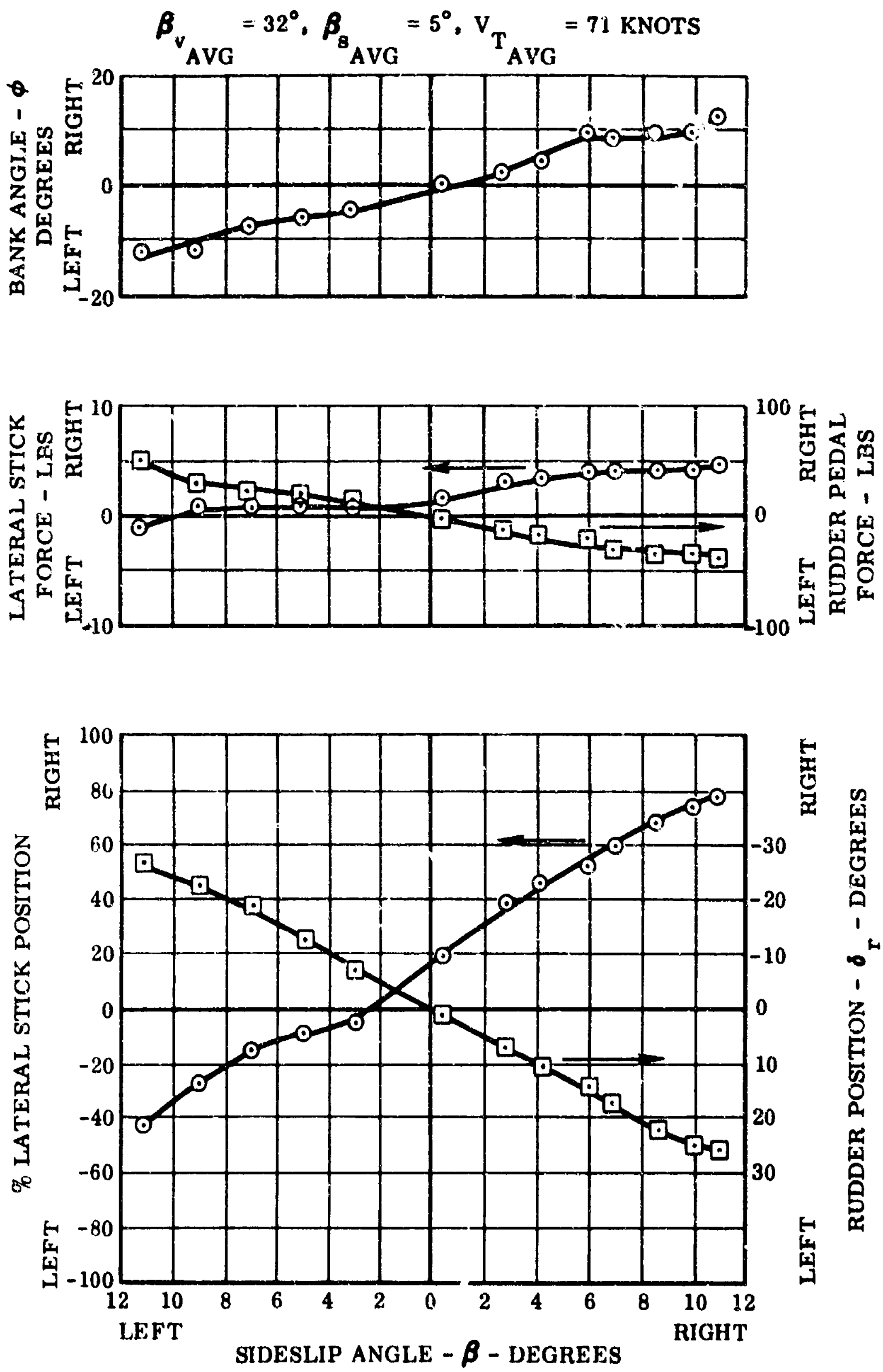


Figure 33 Steady State Sideslips

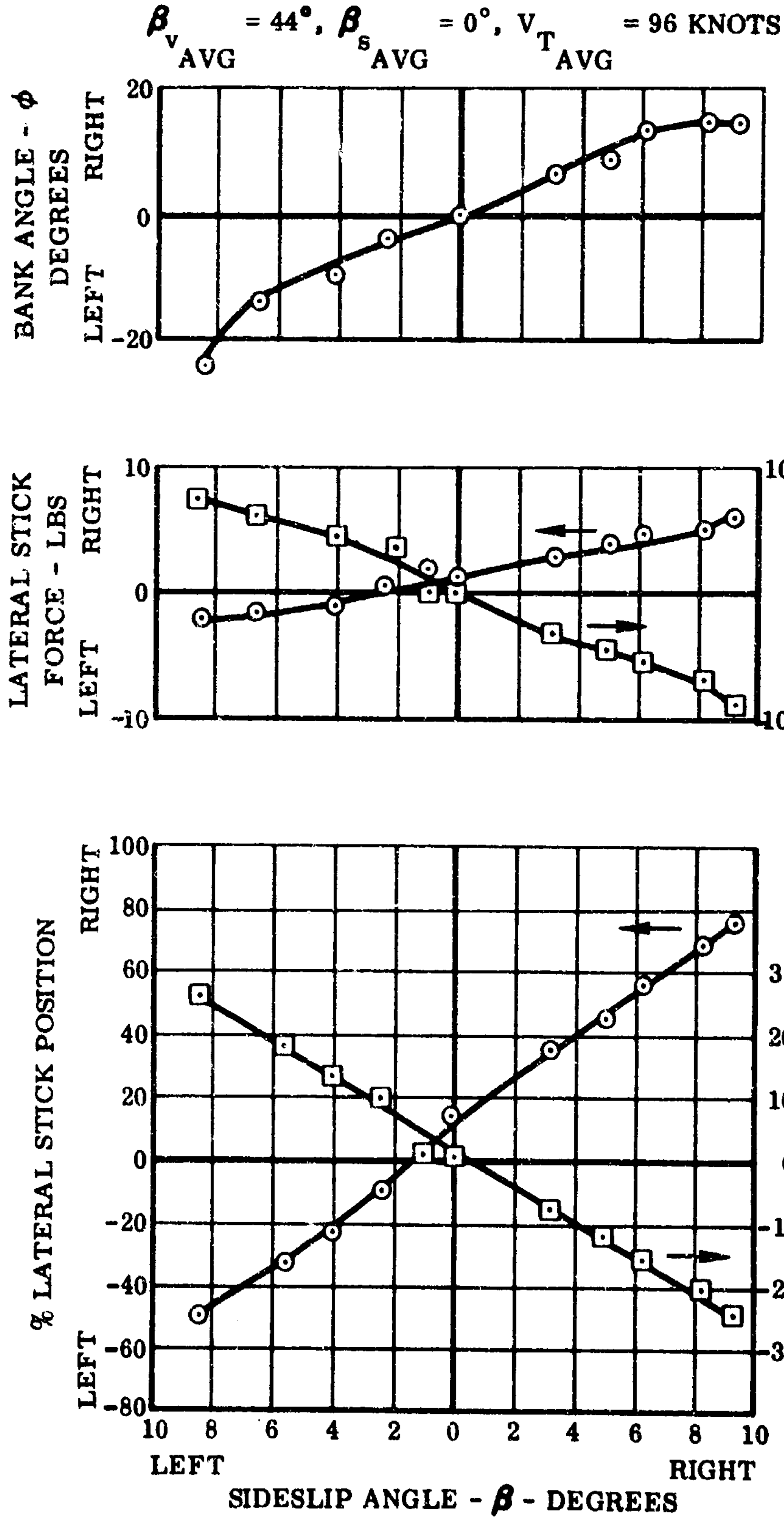


Figure 34 Steady State Sideslips

$Ix = 4,252$ SLUG-FT^2

● $L\phi = 25\ (I_x)^{.7} = 8,750$ FT-LB/RAD/SEC (SPECIFIED DAMPING)

▲ $L\phi = 3\ I_x = 12,800$ FT-LB/RAD/SEC (ADJUSTABLE DAMPING)

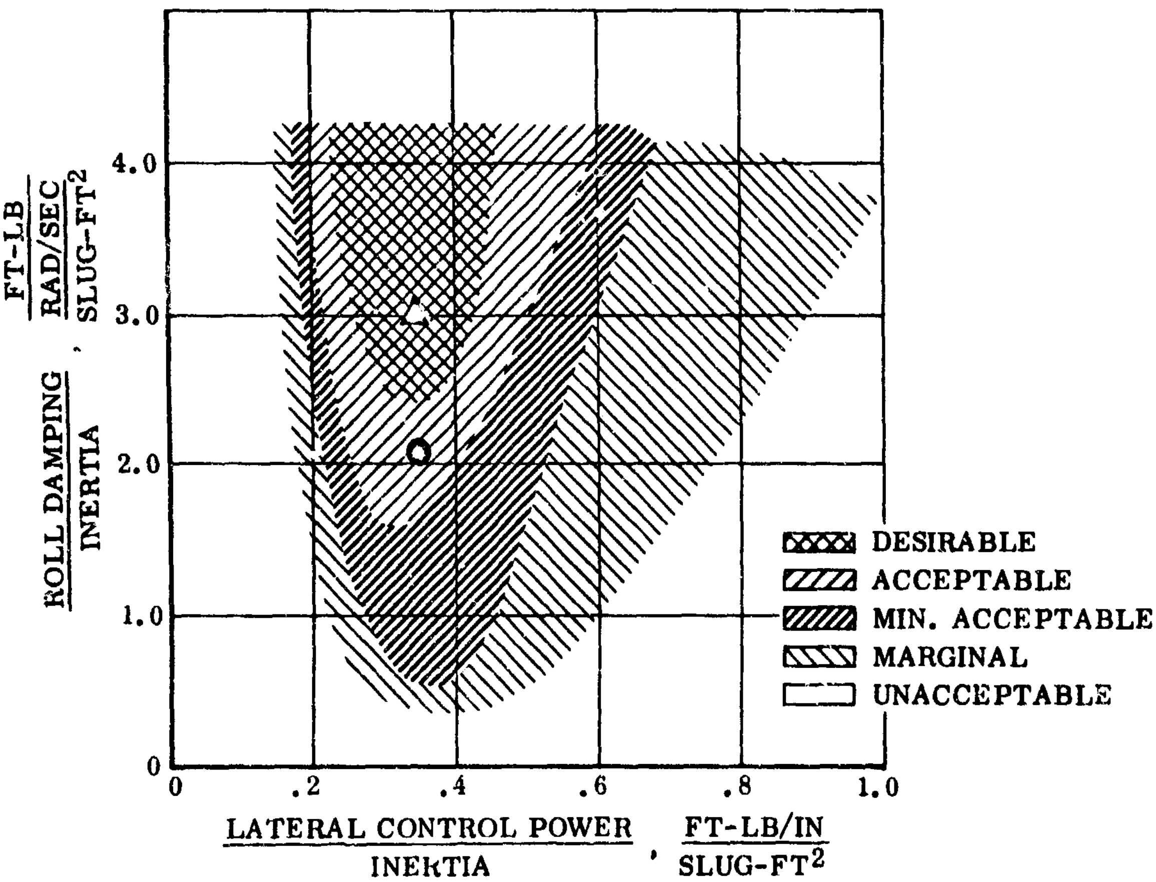


Figure 35 Lateral Control System Characteristics

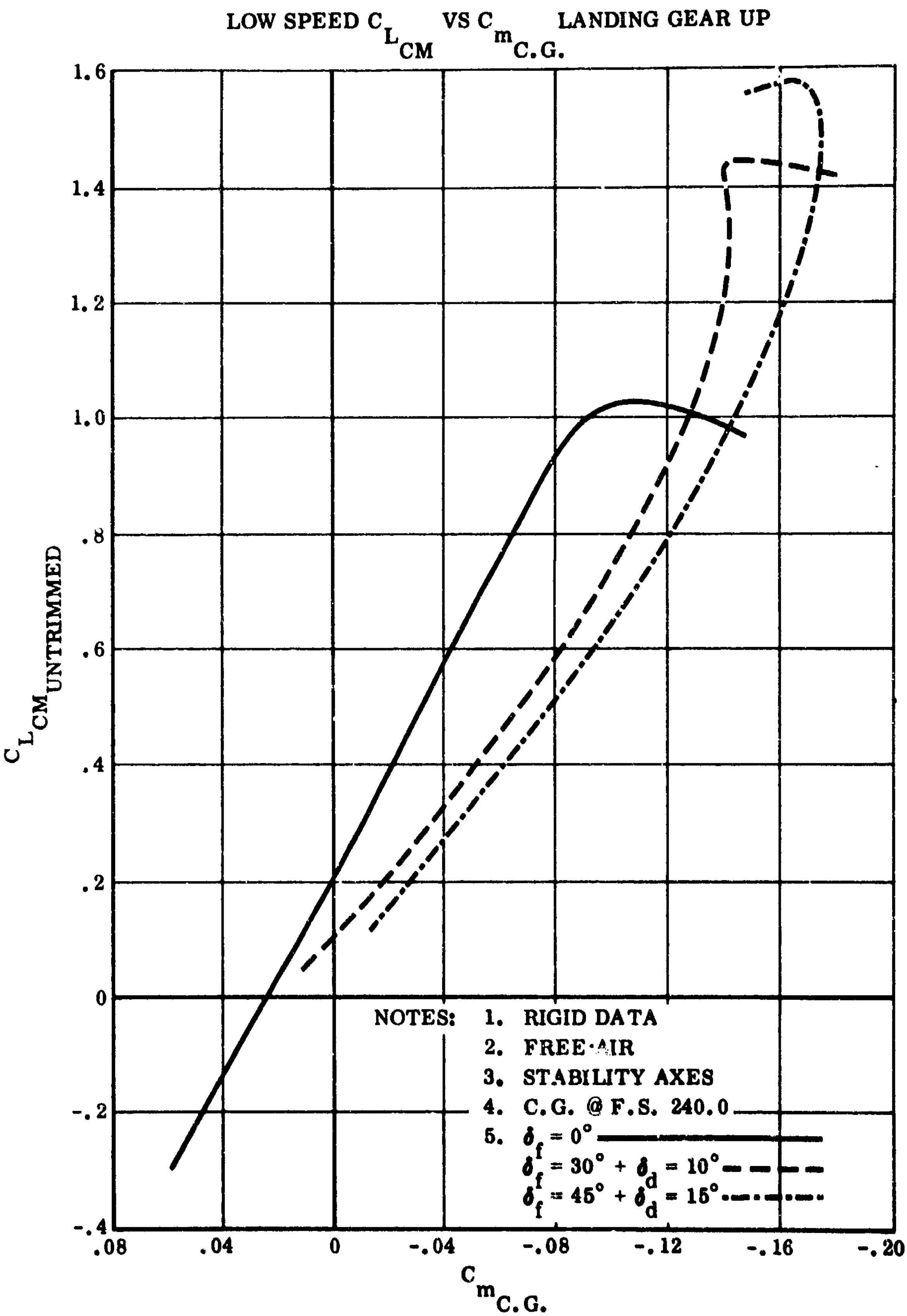


Figure 36 Static Longitudinal Characteristics

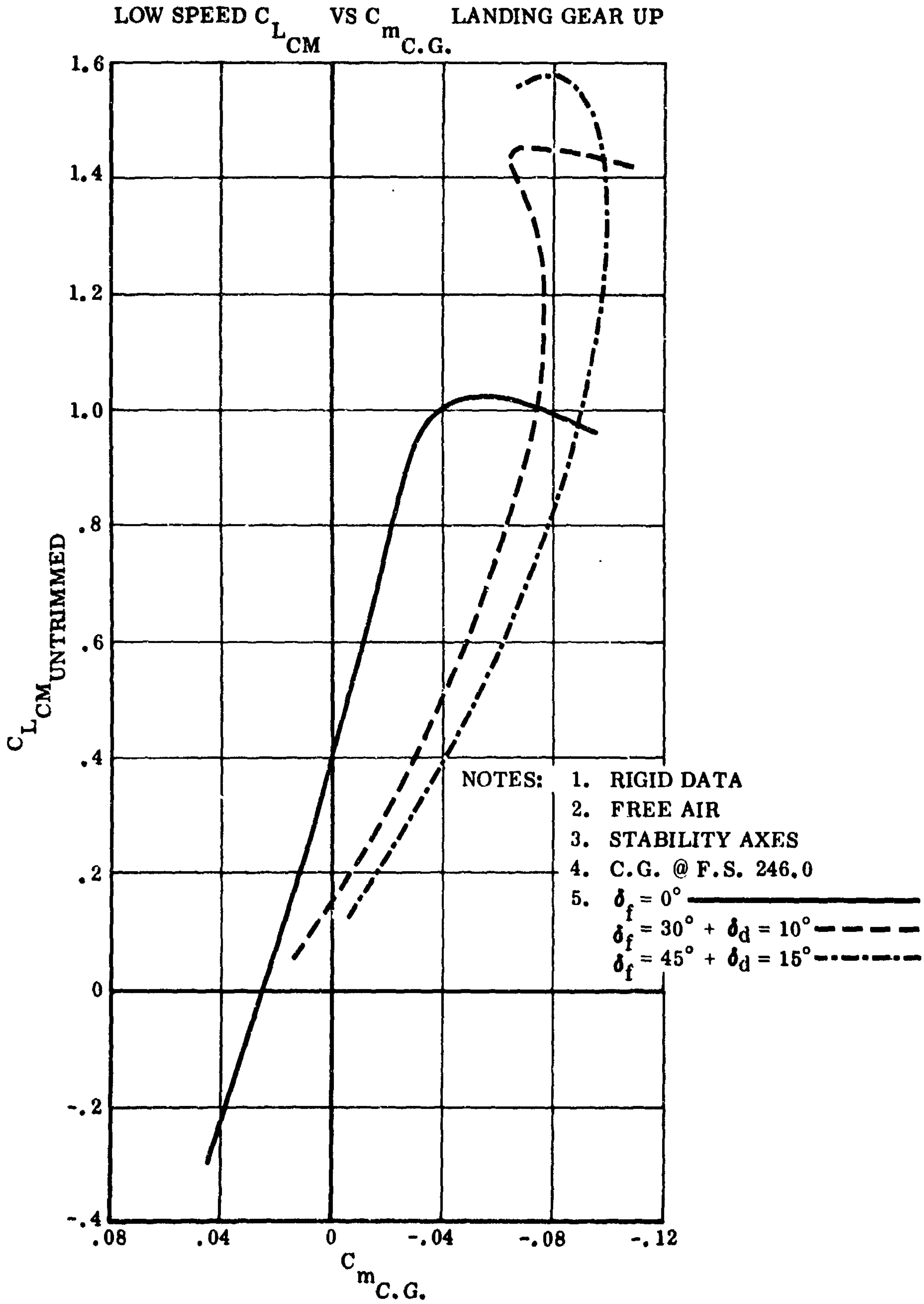
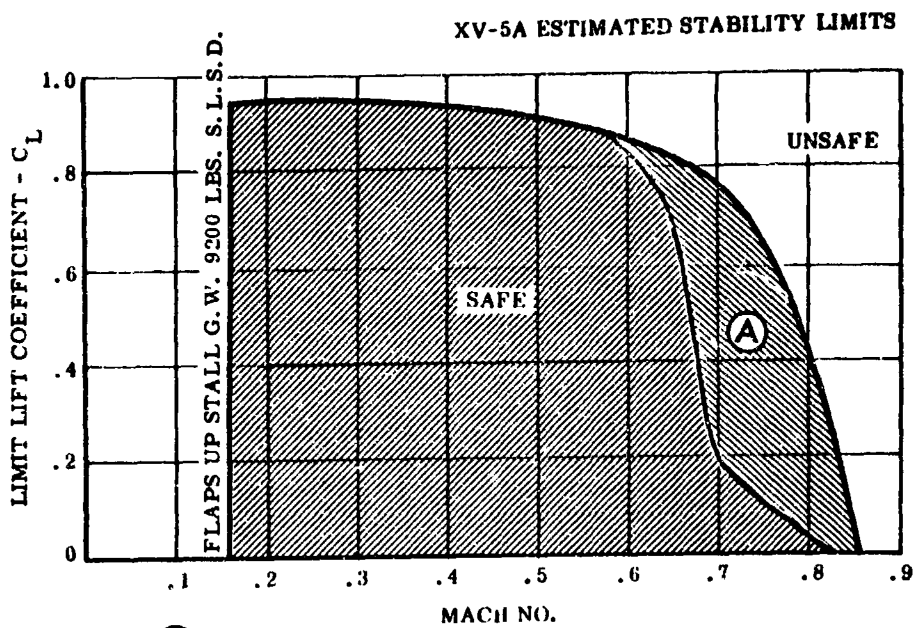


Figure 37 Static Longitudinal Characteristics



(A) QUESTIONABLE AREA, REF. RYAN REPORT 63B087

Figure 38 Static Longitudinal Characteristics

NOTES: 1. RIGID DATA
2. FREE AIR
3. STABILITY AXES
4. C.G. @ F.S. 240.0
5. $\delta_f = 0°$

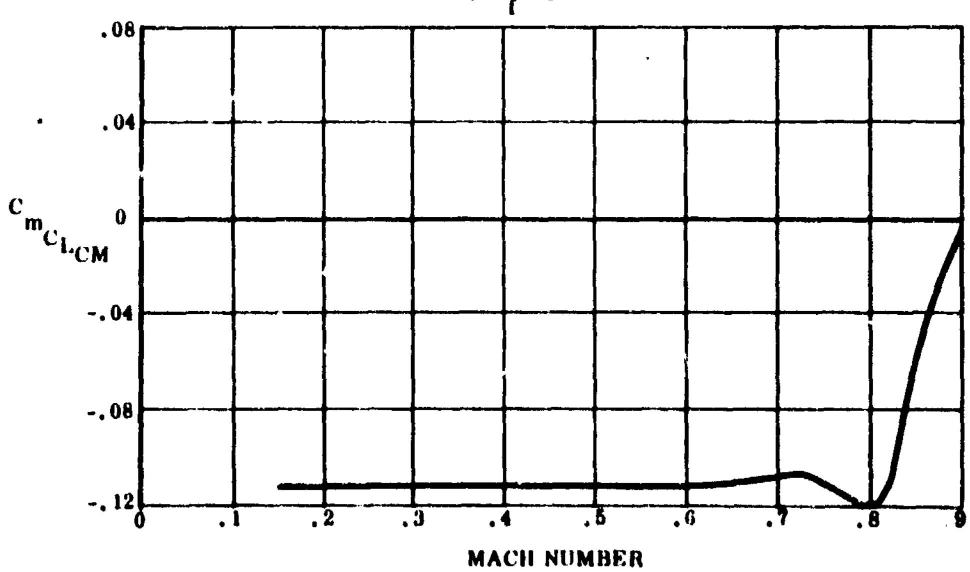


Figure 39 Static Longitudinal Characteristics

NOTES: 1. RIGID DATA
2. FREE AIR
3. STABILITY AXES
4. C.G. @ F.S. 246.0
5. $\delta_f = 0°$

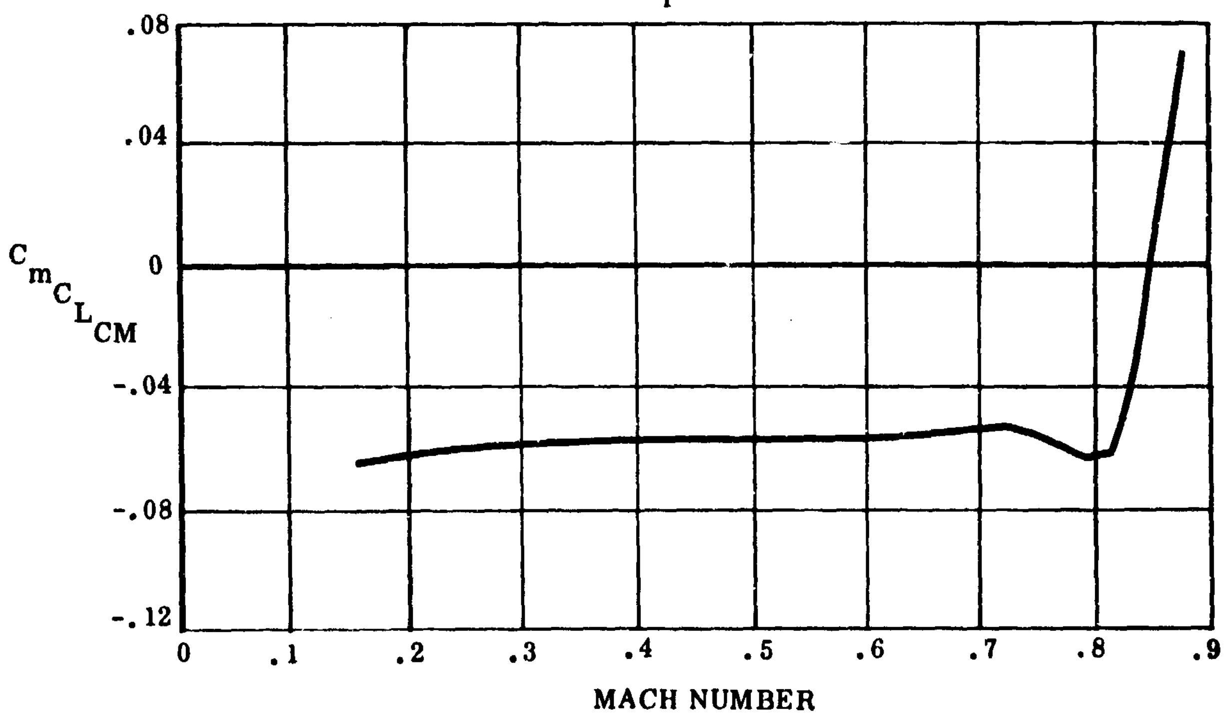


Figure 40 Static Longitudinal Characteristics

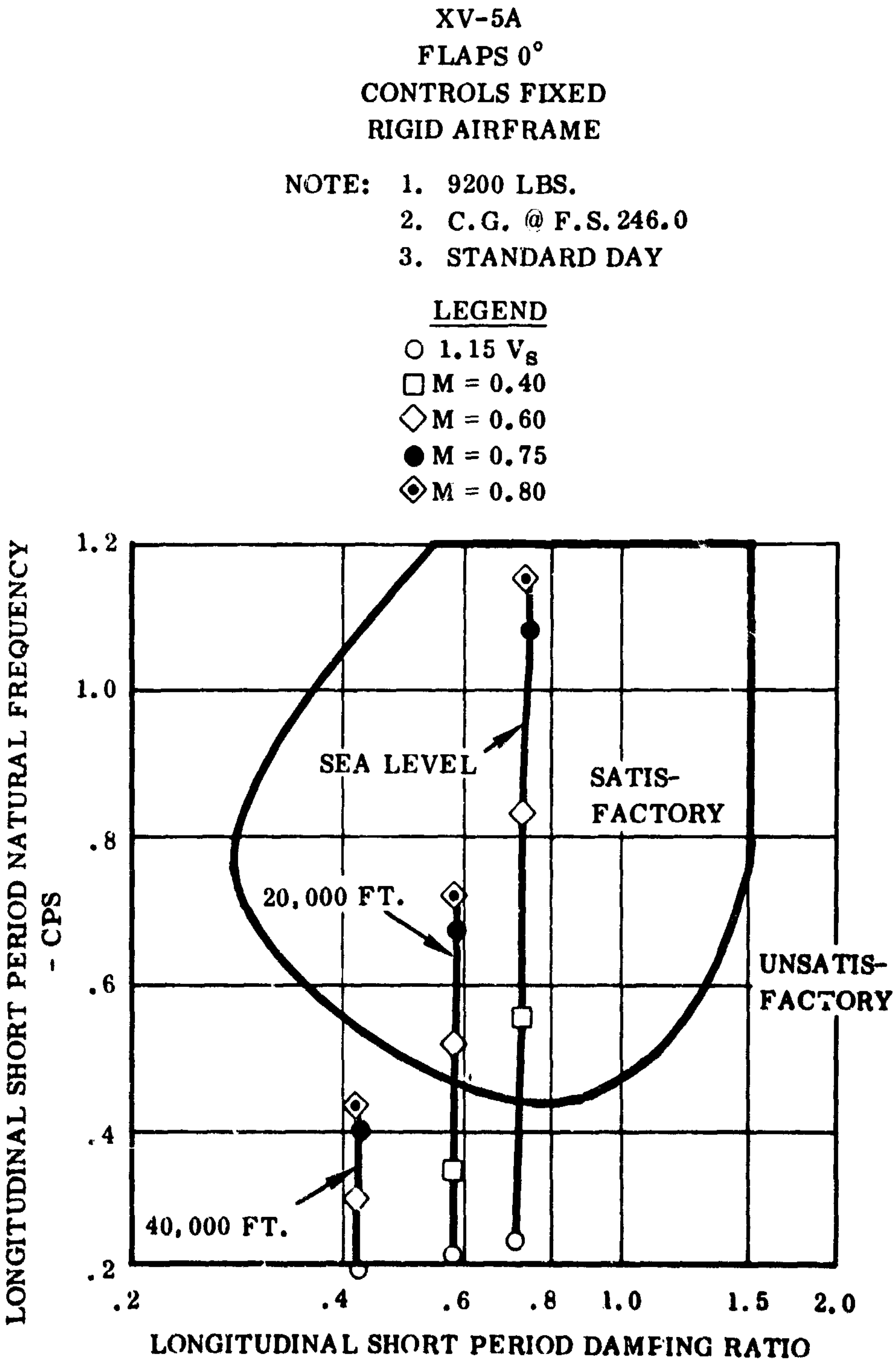


Figure 41 Longitudinal Dynamic Stability - Short Period Mode

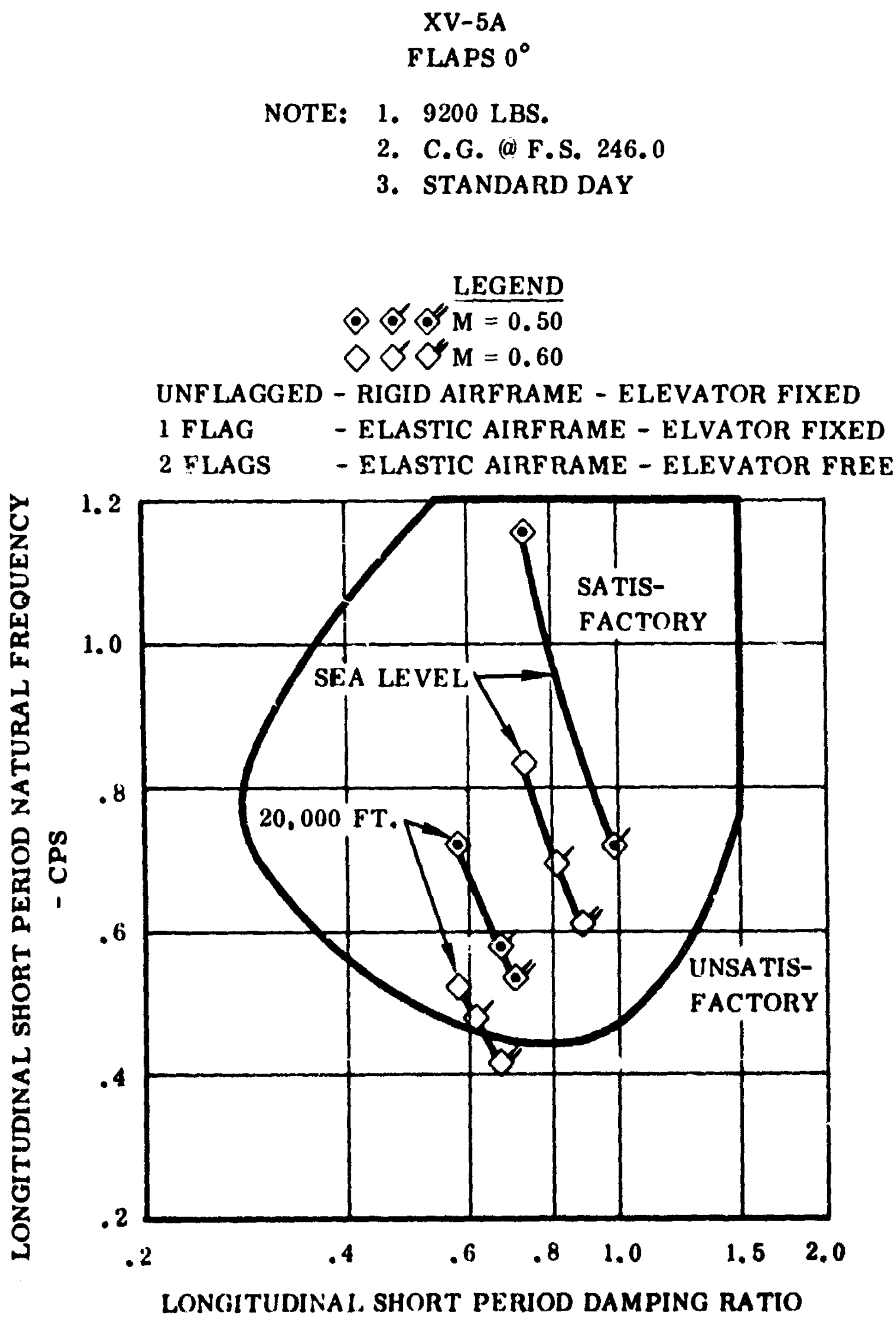


Figure 42 Effect of Aeroelasticity and Free Elevator on Longitudinal Short Period Dynamic Stability

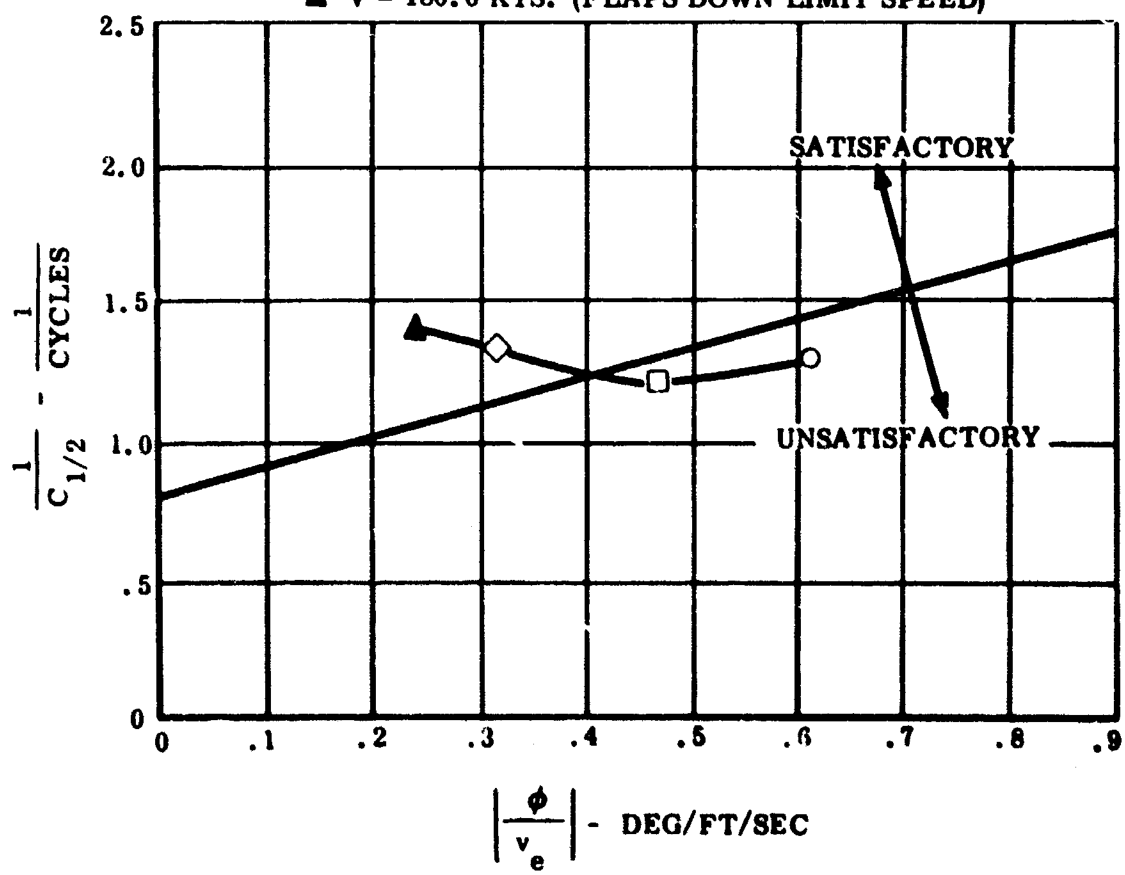
FLAPS 45° + 15°

9200 LBS.

NOTE: 1. STICK-FIXED
2. C.G. @ F.S. 246.0
3. SEA LEVEL STANDARD DAY
4. FREE AIR
5. RIGID DATA
6. CONVENTIONAL FLIGHT MODE

LEGEND

○ V = 94.6 KTS. (1.15 V_s)
□ V = 115.0 KTS. (1.40 V_s)
◇ V = 150.0 KTS.
▲ V = 180.0 KTS. (FLAPS DOWN LIMIT SPEED)



294-69-1

Figure 43 Lateral Directional Dynamic Stability - Oscillatory Mode

FLAPS 0°
9200 LBS.

NOTE:
1. STICK-FIXED
2. C.G. @ F.S. 246.0
3. STANDARD DAY
4. RIGID DATA

LEGEND
$V = 1.15\ V_s$
M = 0.40
M = 0.60
M = 0.75
M = 0.80

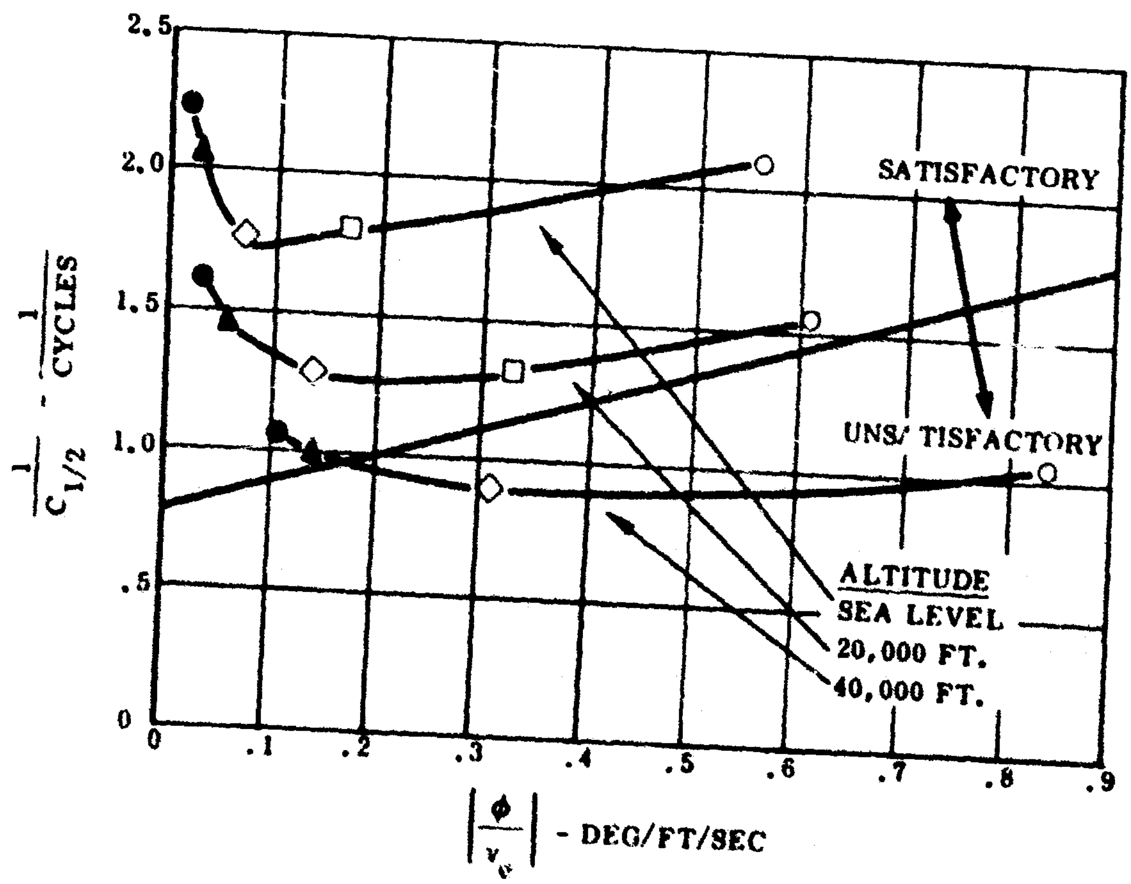


Figure 44 Lateral Directional Dynamic Stability - Oscillatory Mode

FLAPS 0°

NOTE: 1. 9200 LBS.
2. C.G. @ F.S. 246.0
3. STANDARD DAY

LEGEND

◇ ◇ ◇ M = 0.60
○ ○ ○ M = 0.80

UNFLAGGED RIGID AIRFRAME - CONTROLS FIXED
1 FLAG ELASTIC AIRFRAME - CONTROLS FIXED
2 FLAGS ELASTIC AIRFRAME - RUDDER FREE

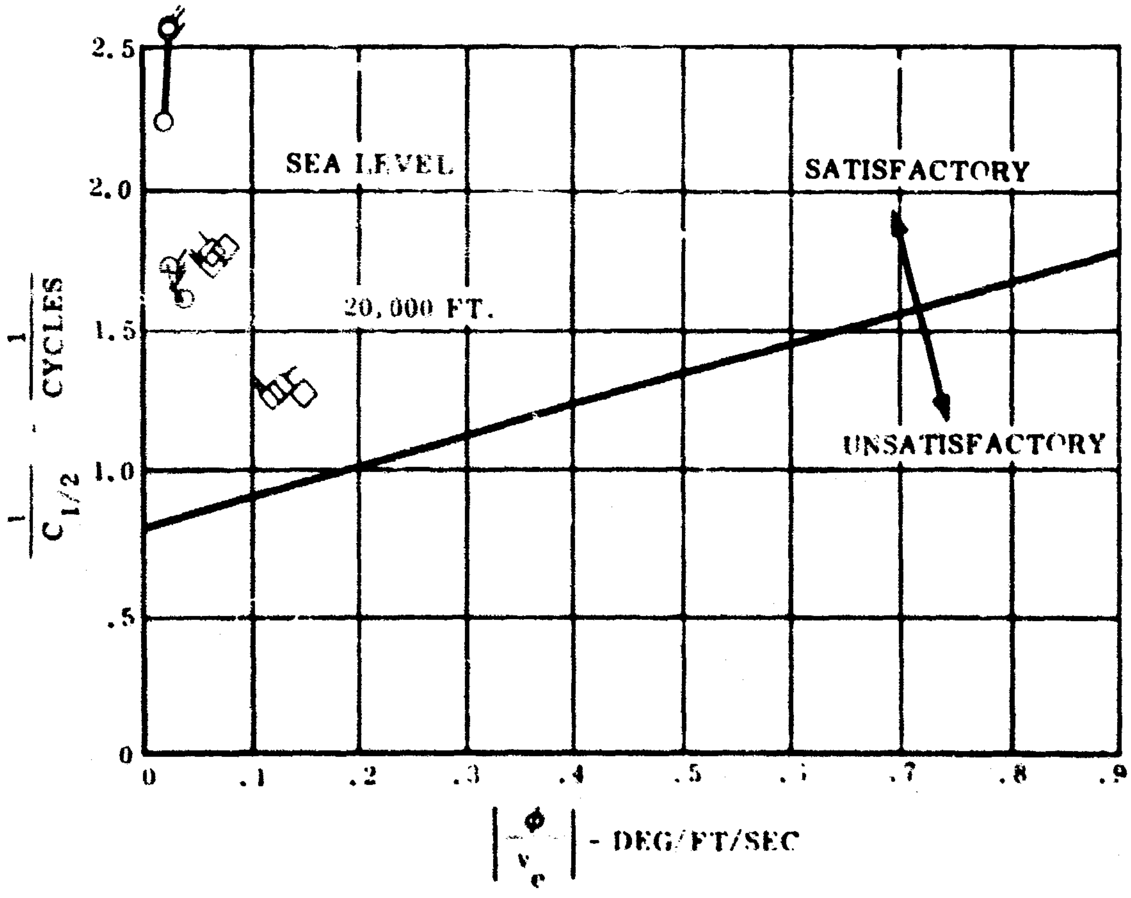


Figure 45 Effect of Aeroelasticity and Free Rudder on Lateral-Direction Dynamic Stability - Oscillatory Mode

FLAPS 0°

NOTE: 1. 9200 LBS.
2. C.G. @ F.S. 246.0
3. STANDARD DAY

LEGEND

◇ M = 0.60
○ M = 0.80
UNFLAGGED - RIGID AIRFRAME - CONTROLS FIXED
1 FLAG - ELASTIC AIRFRAME - CONTROLS FIXED
2 FLAGS - ELASTIC AIRFRAME - AILERONS FREE

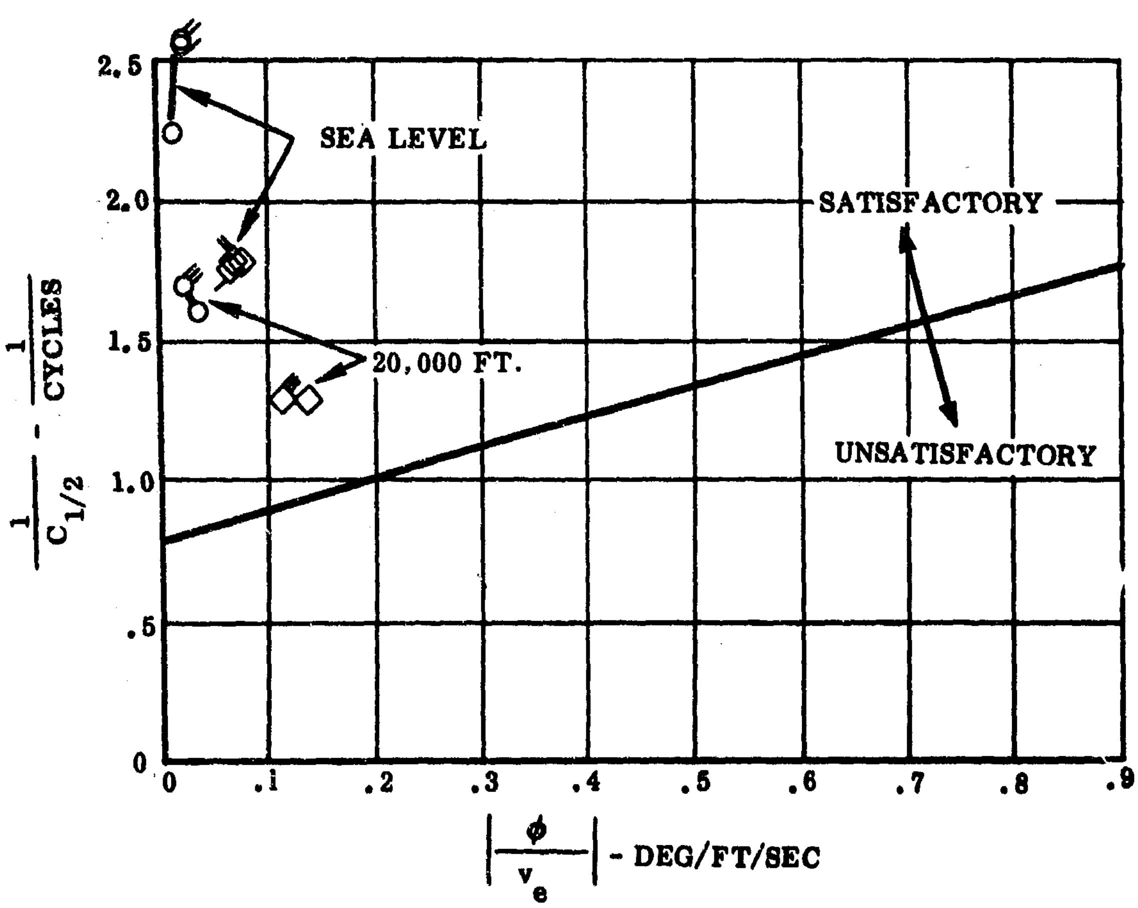


Figure 46 Effect of Aeroelasticity and Free Ailerons on Lateral -Directional Dynamic Stability - Oscillatory Mode

3.3 FLUTTER AND VIBRATION

The flutter and vibration support for the XV-5A aircraft design was organized so that optimum evaluation of the basic design could be effected through careful merging of theoretical analyses and experimental ground tests, prior to the final flight vibration tests for envelope expansion.

3.3.1 General

Support in the area of flutter and vibration was provided concurrently with design, manufacture and flight testing of the aircraft. Theoretical analyses of a preliminary nature initially provided the best results due to the ease with which aircraft design changes could be incorporated. Next, as the design was set, a wind tunnel model of the wing provided good evaluation of the design and also provided checks on the preliminary analyses. Final checks were provided by utilizing experimental results of ground tests in analytical investigations. In this way, insight was gained in the structural dynamic behavior of the aircraft, and provided a measure of confidence during final flight flutter testing.

3.3.2 Conclusions

The overall flutter analysis and experimental phases, both ground and flight tests of the XV-5A aircraft, have indicated that the aircraft is free of flutter within the prescribed flight envelope. Initial static and dynamic tests of the empennage indicated a low horizontal stabilizer pitching frequency, which, when compared to theoretical calculations, indicated that a potential flutter problem existed. Subsequent equivalent pitch restraint and dynamic tests indicated a flutter speed, based upon the initial calculations, to be above the limit dive speed of the flight envelope. Further calculations, based upon experimental shake test modes of the modified structure (after structural changes to the horizontal stabilizer pitch restraint), supported the earlier conclusions.

3.3.3 Criteria

The requirement of the flutter and vibration program was to determine adequately that the XV-5A aircraft was free of any flutter instability within the design flight envelope. Flutter margins were applied corresponding to MIL-A-8870, "Airplane Strength and Rigidity, Vibration, Flutter and Divergence", dated 18 May 1960.

3.3.4. Analytical Investigations

The analytical portion of the flutter and vibration program was in three parts. Each analysis could be achieved independently without altering the final analysis of the aircraft as a whole. The wing, empennage and finally, control surfaces were treated separately, but final results did not affect the flutter characteristics of the aircraft as a whole. Flutter analyses were restricted to the conventional flight mode.

3.3.4.1 Wing

The wing preliminary flutter analysis was performed on a passive analog computer with the aid of Computer Engineering Associates, Pasadena, California. Results of this investigation are presented in Reference 8, and the results indicate that the XV-5A wing is free of flutter within the specified flight envelope. The study was exhaustive in variations of wing bending material, aileron mass-balance, aileron spring restraint, aircraft simulation effects (fuselage and/or aircraft degrees of freedom) and the wing leading edge box stiffnesses which were evaluated from a flutter standpoint.

3.3.4.2 Empennage

The empennage analysis covered several phases continuing up to the actual flight testing of the aircraft. The analysis was aided by a Ryan digital computer program which incorporated both calculated and experimental vibration modes. Initial investigations showed a low empennage flutter speed in the anti-symmetric sense. Subsequent studies of the torsional stiffness distribution of the vertical stabilizer indicated the need for increased stiffness, which was incorporated into the design. In addition, symmetrical analysis indicated a need for increased pitch stiffness of the horizontal stabilizer. This was done while the aircraft was at EAFB. Final theoretical analysis of the empennage, utilizing experimentally-determined modes shapes, showed satisfactory results throughout the design flight envelope. Reference 9 details the complete analytical investigations of the empennage.

3.3.4.3 Control Surfaces

Preliminary analysis of the control surfaces was restricted to the basic control surfaces except for the longitudinal system, flight or trim tab if appropriate to the system, and to the control circuit with the cockpit controls. Two-dimensional aerodynamic theory with corrections for the internal aerodynamic balance were used throughout the analysis. Results of the preliminary analysis indicated possible flutter regions within the flight envelope for certain values of the aileron uncoupled rigid body frequency, for a given aileron flight tab restraint (control circuit), and for an uncoupled rigid body rudder trim tab rotational frequency of less than 50 cps. Subsequent analysis, based upon experimentally-determined mass properties and control surface - control circuit frequencies indicated a flutter-free system within the design envelope of the aircraft. Reference 10 presents, in detail, the above analysis.

3.3.5 Experimental Investigations

The experimental investigations, required to carry the flutter and vibration program of the XV-5A aircraft through to completion, included wind-tunnel testing of a high speed model of the wing with appropriate fuselage constraints and freedoms. Static and dynamic tests were also performed on a jig-mounted horizontal stabilizer. Full-scale ground vibration tests of the complete aircraft were made, and finally, in-flight vibration (flutter) tests were accomplished.

3.3.5.1 Wind Tunnel Tests

Wind tunnel testing of a flutter model was confined to the wing only, and in the conventional mode. The wing simulation followed the final actual wing construction of two-spars, and also simulated fan mass and inertia. Fuselage effects were included so that fuselage and/or aircraft degrees of freedom could be represented. Ailerons and flight tabs of the model were based upon analysis, and these components participated in the flutter mode of the wing tests. Results of this experimental program indicated that the wing is free of flutter within the design envelope of the aircraft. Adequate stiffness restraint is important since flutter characteristics were altered by variation of this parameter. The aileron differed from that analyzed in the preliminary analysis (Section 3.3.4.1) in that no mass-balance was included in the flutter model, due to a change to a powered system with flight tab from the initial manual system. Reference 11 depicts the aspects of this phase of the flutter investigations.

3.3.5.2 Static and Dynamic Ground Tests

Initial ground tests were restricted to the horizontal stabilizer in an effort to determine the equivalent pitch spring. Evaluation of the results indicated a low spring rate, and when compared to the results of the preliminary analysis (Section 3.3.4.2), indicated a low flutter speed. The next series of tests encompassed the complete aircraft in which aircraft mode shapes and frequencies were determined. In addition, component (control surfaces, flaps, fan doors, etc.) modal characteristics were determined. Upon stiffening of the horizontal stabilizer pitch restraint (as mentioned in Section 3.3.2) a second ground shake test was conducted at EAFB to evaluate these effects. These tests, covering both techniques and results, are discussed in Reference 12.

3.3.5.3 Flight Tests

Expansion of the flight envelope called for in the Phase I flight testing of the XV-5A aircraft resulted in a series of flights which evaluated the sub-critical response of the aircraft to external disturbances. The aircraft was excited by applying sharp control inputs in the appropriate axis, with the response being picked up by accelerometers. Selected signals, in turn, were telemetered to a ground station, where immediate evaluation of the overall damping was made. Between flights, magnetic tapes containing the response signals were partially analyzed for a more detailed analysis of the response. In all, fourteen test points were flown with the entire flight envelope showing satisfactory damping. Reference 16 presents the complete results of this phase of the experimental investigation of the XV-5A aircraft.

4.0 STRENGTH REQUIREMENTS AND COMPLIANCE

4.1 GENERAL

Strength requirements of the XV-5A airframe were specified in the Structural Design Criteria Report (Reference 17), submitted early in the program, and accepted as the official specification for all loading conditions and stress analyses. Although the MIL-A-8860 series specification served as a guide for this criteria, it was not followed exactly because of the special intended use of the airplane, and because it was agreed that a VTOL airplane intended for test and evaluation under ideal conditions should not be subject to the stringent military aircraft requirements capable of meeting broad handling and flight boundaries. In establishing the strength criteria, some of the provisions in the MIL specifications were omitted, some were simplified, and some were extended to cover unique characteristics, such as hovering flight, transition flight, and vertical landings.

An intended service life of 250 hours was specificed. This life requirement meant that fatigue problems would be relatively minor, and also that the probability of inadvertent loads would be lower than those for operational aircraft. Other items concerned with structural integrity were similar to conventional aircraft, including a 1.5 factor of safety and the usual specifications for allowables, deformations, vibrations, and thermal effects.

Loads were calculated in accordance with the structural design criteria, and a summary of design load together with methods of calculation, maneuvering time histories, aeroelastic characteristics, etc. were recorded in the Loads Report (Reference 18). Wind tunnel model data were used in the development of aerodynamic loads, and the balance of these with inertia was dependent upon extensive use of digital computer programs (IBM 704). The calculation of ground loads was based on MIL-A-8862. A summary of ground loads, plus internal landing-gear loads, may be found in Reference 19. Both static tests and structural analysis were used as a proof of adequate structural strength for these loads. Prior to the development of a static test program (Reference 20), sufficient preliminary structural analysis determined which load conditions would be critical for the major structural items. The detailed static test procedures are described in Reference 21. The tests were satisfactory and the results are recorded in Reference 22. Structural analysis reports (References 26 through 35) constitute proof of the structure. The publication of these reports, which are mainly summaries of critical analyses, followed lengthy analyses which continued throughout the design phase.

Since the static proof test program was conducted successfully, and positive margins of safety were found for all critical loads, it is concluded that the XV-5A airplane is structurally flightworthy.

The XV-5A program did not provide complete structural flight testing or flight load survey. However, operational limits beyond those required for normal mission performance were specified, and these limits, including envelopes for speed-altitude and speed-load factor (V-n), were approached during the Phase I flight testing without any structural, or other difficulty.

A few of the more noteworthy speed-load factor points were taken from the flight test data and superimposed on the maneuvering envelope - gust diagram (Figure 47). Note that the maximum normal load factor experienced in Phase I was approximately 80% of the 4.0 maximum design limit load factor, based on a 9200 pound basic design gross weight. This point, and the others (particularly those close to the more critical upper part of the operational or desired envelope) are added evidence of airframe airworthiness.

4.2 FLIGHT LOADS

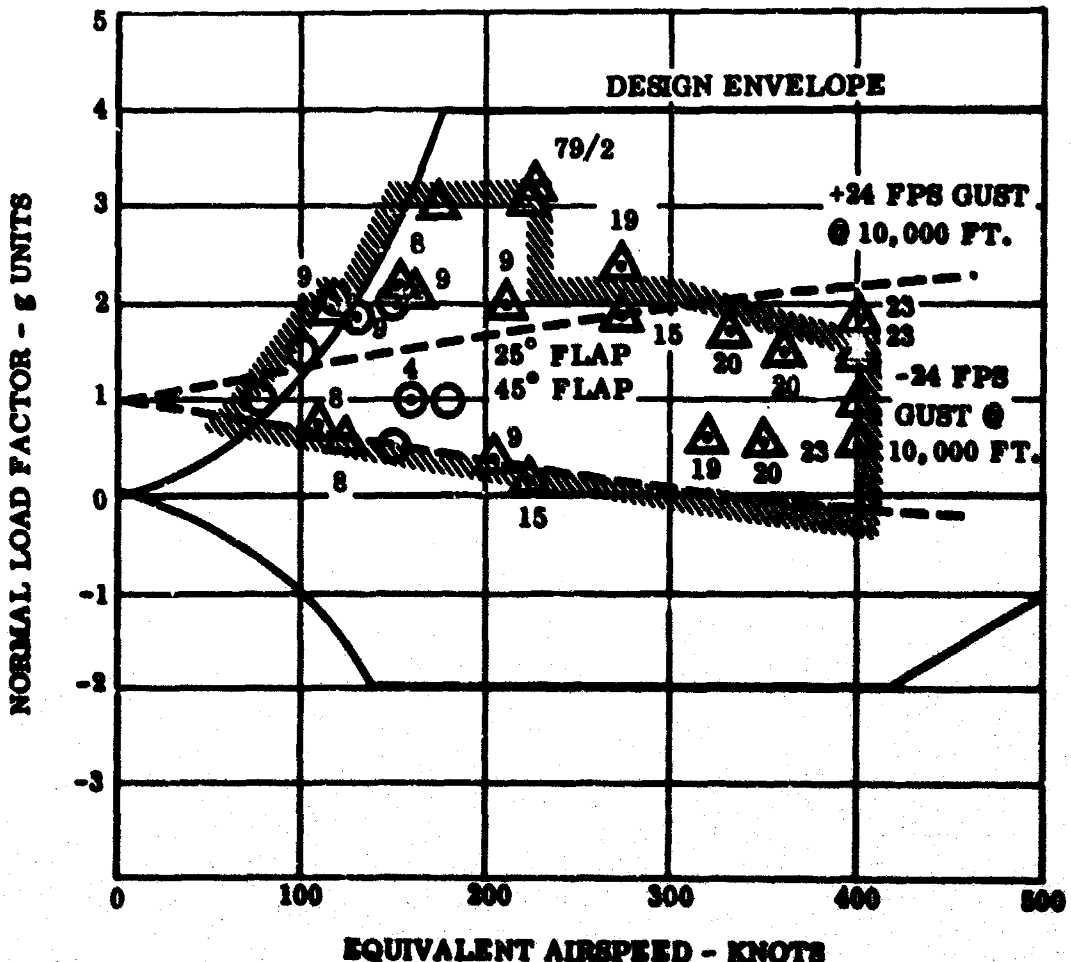
The following is a discussion of the load conditions considered, methods used in calculating loads, methods used in the stress analyses, and the particular proof tests conducted.

The structural design flight loading conditions (Reference 17) were defined to provide adequate limitations within which required maneuvers can be performed with the XV-5A. The analysis of these loading conditions consisted of evaluating them within specific speed, altitude, weight and c.g. restraints. This required investigation of aerodynamic, propulsive and inertia forces and their effect upon the loading of the various airplane components.

For some conditions, the total design load on the airplane was directly established by the structural criteria. For others, it was necessary to analyze the specific maneuvers to determine the design loads which occur during the dynamic motions of the maneuvers.

SYMBOL	L.G. AND FLAPS
O	EXTENDED
Δ	RETRACTED

BASIC DESIGN GROSS WEIGHT = 9, 200 POUNDS
DENOTES BOUNDARY OF DESIRED FLIGHT ENVELOPE



Δ¢⊙ INDICATE POINTS DEMONSTRATED IN PHASE I FLIGHT TESTS. INDEXED BY FLIGHT NO., FOR A/C NO. 1 UNLESS OTHERWISE NOTED. LOAD FACTORS WERE PROPORTIONAL TO BE CONSISTENT WITH 9200 POUND GROSS WEIGHT.

Figure 47 Demonstrated Maneuvering Envelope - Gust Diagram for Altitudes below 10,000 Feet

Wind tunnel test data (References 23 through 25) were utilized extensively throughout the analysis, together with calculated and/or actual distributions of aircraft weight. A basic design gross weight of 9200 pounds was used throughout the analysis. For higher gross weights, adequate structural integrity was assumed when, in accordance with the design criteria, a constant product of load factor and weight (NW) is maintained.

4.2.1 Symmetrical Flight Conditions

Because of the unique capabilities of the XV-5A, investigation of symmetrical flight maneuvers included not only conventional flight, but also the fan-flight conditions of hovering and transition. The design symmetrical maneuvers are completely defined (Reference 17) in terms of angular-and-linear rates-and-accelerations. The gust conditions are defined in terms of a gust environment at various speeds.

The aircraft has been designed to sustain the loads produced by maximum fan lift, induced gyroscopic forces and attitude control capability at speeds of -10 to 125 knots, and at load factors up to 1.3 g's. Angular rates and accelerations based upon the maximum control system capabilities were combined with the vertical load factor to provide critical fan-flight loading conditions.

Conventional flight conditions have been investigated to speeds of 500 knots at sea level. The airplane has been designed to load factors of +4.0 to -2.0 with and without the effects of angular acceleration. The angular velocities and rates appropriate to various combinations combinations of velocity, altitude and load factor are shown in detail in Figure 7 of Reference 17. A system of equations was derived and solved in order to place the airplane in equilibrium for the various design conditions, and to determine the division of load between the wing, body and tail.

Design gust velocities of up to 24 ft/sec at all permissible aircraft speeds (up to V_L) and gust velocities of up to 40 ft/sec at aircraft speeds below 418 knots (V_H) were considered. The maximum calculated gust load factor of 3.6 occurred at sea level, as a result of the 40 ft/sec design gust at 418 knots.

4.2.2 Flaps-Extended Flight Conditions

Conventional flaps-extended flight conditions are identical in presentation to conventional flaps-up flight. The design symmetrical conditions are completely specified in terms of the maximum design load factor at 2.0 maximum design speed of 190 knots, and values of pitching velocity-and-acceleration at various combinations of load factors and velocity. The flight envelope for flaps-down flight is presented in Figure 7.0 of Reference 17.

4.2.3 Unsymmetrical Flight Conditions

Unsymmetrical flight conditions are defined (Reference 17) in terms of lateral gust velocities and of pilot forces applied to the lateral and directional controls. As opposed to the symmetrical flight conditions for which maneuvers were completely defined in terms of load factor, angular acceleration, etc., the unsymmetrical structural design maneuvers required analysis of the airplane motion from the specified pilot force applied to the controls. The resulting motions were then analysed for peak structural loads.

Two types of rolling maneuvers were considered. In the first type, the steady-state roll resulting from a 60 pound pilot force on the aileron control is combined with a vertical load factor of 1.0. The rudder remains neutral throughout the maneuver. A second type of maneuver has been called the rolling-pull-out. The airplane is initially in a constant-altitude turn at a bank angle commensurate with the particular vertical load factor (1.0 to 2.5). The maneuver is executed by application of a 60 pound force to the lateral control system in not more than 0.1 second. This force is maintained until the airplane has rolled out of the turn through an angle equal to twice the initial bank angle. The roll is then checked by full reversal of the control force.

Rudder induced yawing maneuvers have been investigated by considering four conditions during the maneuver: (1) an abrupt rudder deflection, (2) the dynamic overswing, (3) the steady-state sideslip, and (4) an abrupt return of the rudder to neutral from the steady-state sideslip. At speeds up to 250.8 knots, (.6 V_H at S.L.), the rudder deflection is that which results from a 300 pound pilot force. At greater speeds, a 200 pound force was assumed.

Design lateral gust conditions are identical to the vertical gust conditions of Section 4.3.

4.3 WING LOADS

Critical wing loads occur as a result of symmetrical and unsymmetrical flight conditions. Symmetrical maneuvers are characterized by aircraft loadings produced by displacement of the cockpit longitudinal control to attain a pre-established vertical load factor. Since the dynamic state of the airplane was defined, it was then necessary to place the applied force in equilibrium with inertia forces and parametrically evaluate the effects of speed, altitude, c.g., power, etc. Therefore, to place the airplane in equilibrium and to determine the primary subdivision of loading between wing, body and tail, a system of equations was derived to determine:

1. Trim angle of attack for unaccelerated level flight assuming zero elevator deflection with trim achieved by tail incidence.

2. Equilibrium angle of attack which produces specific linear and angular accelerations and angular rates.

3. Subdivision of loading among the primary aircraft components.

The equations are discussed in detail on Page 7 of Reference 18.

To facilitate solution of the equations and thereby afford broad parameter investigations, a digital computer was employed. Although the equations were developed on the basis of a stability-axis system which assumes a negligible variance from an ideal body axis system, artificial derivatives were utilized to provide realistic solutions for the high-speed stall conditions. Iterative calculations were required for the solution of the high-speed stall conditions because of nonlinear aerodynamic derivatives. Aerodynamic $C_{L_{max}}$ of 1.25 times the static value was considered for the high-speed stall conditions.

For most of the calculations, a rigid airframe was assumed. However, for selected critical symmetrical flight conditions, the effects of an elastic wing were also investigated. No appreciable change in loads resulted from the investigation.

The maximum calculated wing lift of 33,476 pounds results from a high-speed 4.0g maneuver with flaps up. The maximum wing load with flaps down was calculated to be 19,820 pounds. A summary of wing loads for numerous selected symmetrical flight conditions is presented in Table 4.1 of Reference 18.

Critical unsymmetrical wing loads occur during rolling maneuvers. Roll maneuvers were analytically investigated through impulsing the airplane by rapid displacement of the aileron control in accordance with the design criteria (Reference 17). Wing loads are primarily dependent upon angle-of-attack, roll rate, roll acceleration, and aileron deflection. Since load factor, and therefore angle-of-attack, were held constant, a simplified one-degree-of-freedom analysis was employed for the wing. In addition to those describing aircraft motion, equations were formulated to define the response of the lateral control system to finite pilot forces. The equations are summarized on Page 13 of Reference 18. Wing loads for various time points throughout the maneuver were combined with the appropriate symmetrical loads to define the overall wing loads.

Elastic loads calculations of the roll maneuver reflected consideration only of wing flexibility, which was found to be relatively stiff in the symmetrical mode and relatively flexible in the anti-symmetrical mode. For this reason, the unsymmetrical wing loads from the rolling maneuver were calculated on the basis of an elastic wing and the symmetrical contributions assumed a rigid structure.

The wing loads, as presented for structural analysis, were represented by concentrated forces at a discrete number and location of panel points as depicted in Figure 3.8 of Reference 18. The distributed load included the effects of inertia, aerodynamics and aeroelasticity. The distribution of airloads were determined from wind-tunnel data (Reference 18 and 23 thru 25). The calculations for distribution of the loads to the panel points were performed to a large extent by a digital computer.

Wing aileron loads were determined on the basis of maximum pilot effort inputs (Reference 17). The critical aileron design load of 3125 pounds occurs at the maximum sea level flight velocity (Reference 18). The design flap load in terms of the maximum hinge moment is 9420 in.-pounds per flap (Reference 18). This moment occurs at 180 knots with full flaps.

Wing-fan closure door loads occur during both conventional flight with the doors closed and in fan flight with the doors open. The maximum door loads during conventional flight were calculated to be 5000 pounds for both doors on one fan (Reference 18). These occur during a high-speed 4.0g symmetrical flight condition. The maximum open door load occurs at 110 knots during a 40 f.p.s. lateral gust. This was calculated to be a door load of 800 pounds (Reference 18).

4.3.1 Structural Analysis and Test

Since lift fans in the wing accounted for significant torque-box structure loss, a simple unit-beam method of stress analysis was not applicable. The basic wing structure consists of a conventional torque-box outboard of the lift fan, two full-span spars bolted to the carry-through structure at the fuselage, and an inboard leading-edge torque-box. This basic structure was idealized into a system of bars and webs and analyzed as a redundant problem by use of a general method programmed for the IBM 704 Computer. In the solution, internal loads were found as functions of externally-applied unit panel point loads. Deflection influence coefficients were also found, and these were used in flutter analysis. As noted above, symmetrical and unsymmetrical flight loads were found in terms of the same panel point forces, so that a considerable number of load conditions could be run through the stress and deflection analysis program. The results of sixteen symmetrical and twenty unsymmetrical conditions are given in the stress report (Reference 26). All stresses and deflections were within allowable limits. The condition most critical for the rear spar and its attachment (Symm. Flt., Pos. Low Angle of Attack, Zero Pitching Acceleration) was simulated with satisfactory results in the static proof test (References 20, 21 and 22). Internal strains and external deflections from test compared favorably with those from the analysis.

Stress analysis of the flap was based on a loading corresponding to the maximum hinge moment. The chordwise pressure distribution considered was rectangular from the leading edge to 60% chord, and triangular from 60% chord to the trailing edge. The flap was conservatively analyzed and proved satisfactory (Reference 28). The flap was also static tested satisfactorily to the same limit load (References 20, 21 and 22).

The aileron was stress analyzed as a continuous beam on three supports for a pressure distribution which produced the critical load noted above. The total hinge moment resulting from the airload used in the analysis is greater than the maximum input hinge moment based on actuator capacity, because the reduction in torque due to the tab airload was conservatively neglected. Stress analysis of aileron and tab indicated adequate strength. The aileron and hinge fittings were also proof-tested satisfactorily to the critical load (References 20, 21 and 22). Since the wing fan doors serve as part of the upper wing surface in conventional flight, they had to be analyzed for critical pressures resulting from conventional flight maneuvers. In addition, the doors were analyzed for fan flight conditions with the doors in the open position (Reference 28). Requirements for high rigidity resulted in fairly thick fiber glass skins and correspondingly low stresses throughout the door. Developmental static tests were done during the preliminary design of the doors, and these tests were relied upon to meet rigidity requirements. For additional proof of strength and rigidity, the final doors were installed on the wing fan and tested to critical conventional and fan-flight loads. Various combinations of actuator power were simulated. The tests showed that the doors, support structure, and actuators were adequate (References 20, 21 and 22).

The wing spar-fuselage joints were analysed for the critical shears and moments resulting from a comparison of all conditions analyzed. Ample margins of safety were found (Reference 28). The rear spar joint, which was the more critical of the two, was also proof tested in the basic wing test (References 20, 21 and 22).

The wing fan mount critical loads were taken from 16 load conditions, which were different combinations of thrust vector angle, engine power, linear load factors, and angular rates producing gyroscopic effects. The analysis indicated adequate strength (Reference 28). The mounts were also satisfactorily proof tested (References 20, 21 and 22).

4.4 FUSELAGE LOADS

Fuselage loading results from the combined effects of inertia and external aerodynamic forces. The inertial forces depend entirely upon the load factors specified, or those calculated for the structural design conditions. The external airloads are a function of the flight velocity and altitude, and the angles of attack and sideslip which occur during the design maneuvers. The load factors and angles for symmetrical maneuvers are discussed in Section 4.3, and the unsymmetrical maneuvers in Section 4.5.

The fuselage loads from gound conditions are primarily from inertia. For landing conditions, however, wing lift equal to airplane weight was assumed to act at the wing spar locations. The maximum vertical landing load factor used for design was 3.82 g's (Reference 18).

Two distributions of fuselage weight were used in the analysis (Reference 18) and both were appropriate to a 9200 pound airplane. One distribution results in an airplane c.g. at Station 240 and the other at Station 246.

Fuselage wind-tunnel pressure data were available for Mach numbers of .4 to .9 (Reference 25). Fuselage vertical and lateral airload distributions were determined by fairing through the available data points considering also the fuselage profile and the aerodynamic forces and moments indicated by wind-tunnel force measurements (Reference 18).

To combine all distributed and concentrated loads in the many combinations required to define fuselage loading, a digital computer routine was devised. Basically, the program combines the effects of (a) fuselage vertical and lateral distributed airloads, (b) fuselage distributed inertia loads produced by linear and angular accelerations, (c) concentrated loads and moments at the landing gear and parachute attachments, (d) wing inertia and airloads, (e) empennage inertia and airloads, and (f) engine thrust and ram drag. The program provides fuselage loading in the form of vertical and lateral shear, bending moment and torsional moment (Reference 18).

4.4.1 Structural Analysis and Test

Primary structure of the center fuselage is composed of a space frame consisting of tubular steel members gusseted and welded at the joints. This space frame was idealized as a system of two-force members having 14 redundants, and it was therefore readily adaptable to the computer-programmed method outlined for the wing basic structure. Complete stress and deflection analysis (Reference 32) included loads due to 3 landing conditions, 4 fan-powered conditions, and 9 conventionally-powered conditions. The engine mounts, which are a part of the space frame, were analyzed for critical landing, fan-flight, and conventional-flight conditions. Critical center fuselage and engine mount loads were simulated with satisfactory results in the static tests (References 20, 21 and 22). The conditions included 2-Wheel Tail Down Landing (Spring Back), Drift Landing, Rolling Pull-Out, and Hover.

The forward and aft sections of the fuselage are conventional semi-monocoque structures. Longitudinal bending members together with skins and webs were stress analyzed by means of a box-beam method programmed for the IBM 704. This analysis (Reference 30) considered all critical load conditions: There were (4) for symmetrical flight, (7) for unsymmetrical flight, and (3) for landing. The most severe conditions for forward and aft fuselage were also simulated with satisfactory results in static tests (References 20, 21 and 22).

Detailed stress analysis was accomplished on fuselage frames, bulkheads, fittings, and miscellaneous items, and was summarized in Reference 31. Brief analyses for canopy, pitch fan mounts, pilot seat support structure, fuel tanks, thrust spoiler, and parachute support structure were included in this Reference 31 report. The canopy was tested satisfactorily to ultimate load, simulating the critical pressure distribution due to 500 k at sea level, with 5 degrees sideslip (References 20, 21 and 22). The windshield failed during proof test. The thickness was then increased by 75%, which was shown to be adequate by stress analysis based on the earlier test data (Reference 22).

Analysis of the engine air inlet, the thrust spoiler installation, and the pitch fan louver installation and the results are summarized in Reference 35. The thrust spoiler installation and pitch fan doors were tested satisfactorily during tie-down ground tests with engines at full power. Strength and rigidity of both nose and main landing gear doors were proved adequate by static tests to limit loads corresponding to V = 500 k at sea level.

4.5 HORIZONTAL TAIL LOADS

The majority of the loads critical for design of the horizontal tail result from symmetrical flight maneuvers. The airplane balance methods discussed in Section 4.6.1 provide the overall horizontal tail loads due to angle of attack and to elevator deflection. A maximum load of 7100 pounds was calculated by use of the methods.

Unsymmetrical loading on the horizontal tail is produced during rolling maneuvers, yawing maneuvers and lateral gust conditions. These unsymmetrical maneuvers are discussed in Sections 4.5 and 4.6. Critical horizontal tail unsymmetrical loads result from the dynamic-overswing of the rudder induced yawing maneuvers.

In the calculation of horizontal tail loads, local inertial contributions were conservatively omitted. The distribution of the aerodynamic contribution was determined through application of the well-known Lifting Line Theory. This theory, together with a simplified method of solution, may be found in Reference 36. For the XV-5A, however, an expanded version was formulated and mechanized for solution by digital computer. The expanded method provided greater accuracy and solution of all forms of symmetric and anti-symmetric loadings. An elevator design load of 1270 pounds total has been calculated. This load was based on a maximum pilot effort of 200 pounds being applied to the cockpit longitudinal control.

4.5.1 Structural Analysis and Test

The horizontal stabilizer, a three-spar semi-monocoque structure, was stress analyzed for three critical flight conditions using a box-beam method programmed for the IBM 704 (Reference 27). Elevator stress analysis for a conservative loading corresponding to maximum pilot effort was included in the same report. The horizontal stabilizer was proof tested satisfactorily to a composite condition simulating maximum total load and maximum torsion (References 20, 21, and 22). The elevator was satisfactorily proof tested to a load corresponding to maximum pilot effort.

4.6 VERTICAL TAIL LOADS

The design conditions of rolling maneuvers, rudder induced yawing maneuvers and lateral gust conditions are responsible for loading on the vertical tail. Solution of all of these conditions for structural loads and the distribution of the airloads upon the vertical tail was accomplished through use of a digital computer.

The analysis of the rolling maneuvers determined the motion in the antisymmetrical or lateral-directional mode separately from the symmetrical or longitudinal mode. The results were subsequently superimposed for representation of the net unsymmetrical loading condition. Vertical load factors during the maneuver were considered constant at initial values from 1.0 to 2.5.

Because of the significance of cross-coupling effects on fuselage and empennage loading, a three-degrees-of-freedom solution was used (Reference 18). These correspond to interacted aircraft motions in roll, yaw and lateral displacement. In addition to the equations defining the motion, auxiliary equations were derived to simulate pilot/control system response characteristics. Although this method primarily served as a means of evaluating the rolling pull-out maneuver, it also enabled examination of the inherent characteristic lateral motion during "steady-state" rolls.

The rolling pull-out maneuver investigated consisted of rolling the airplane out of a constant altitude turn through an angle equal to twice the initial bank angle, maintaining zero rudder deflection and assuming the vertical load factor to remain constant. Aileron deflection and rate were the maximum attainable, commensurate with a 60-pound stick force and pilot application time of 0.1 second. Elastic values of aileron effectiveness and wing contribution to roll damping were used for the calculations. Four distinct rudder-induced yawing conditions were analyzed and are:

1. A rudder kick maneuver which assumes an instantaneous rudder deflection to the maximum mechanical limits or as limited by pilot pedal force.

2. A steady-state sideslip maneuver which results from a rudder deflection to the mechanical stops or as limited by a pilot effort of 300 pounds.

3. A dynamic-overswing sideslip condition which assumes that during a rudder-induced yawing maneuver, the airplane will attain an overswing sideslip angle 50% larger than the steady-state value.

4. A rudder deflection reversal maneuver which assumes that the rudder is instantaneously returned to neutral with the airplane in the steadstate sideslip condition resulting from specified values of pilot pedal force.

The equations defining these four static conditions were programmed for solution by a digital computer. Other equations for solution of airplane component loading were also programmed.

For the lateral gust conditions, the airplane was assumed instantaneously exposed to the effects of the sideslip angle resulting from the lateral gust. A simple lateral/directional static balance of the airplane was performed to determine the lateral gust loading. The vertical tail design load of 3527 pounds, resulted from the calculated effects of a lateral gust.

4.6.1 Structural Analysis and Test

The vertical stabilizer, a three-spar semi-monocoque structure, was stress analyzed for two critical flight conditions, one which produced maximum shear and bending moment and one which produced maximum torsion. A box-beam method was used which had been programmed for the IBM 704 (Reference 27). A rudder stress analysis was included in the same report. Margins of safety for the rudder were high, since high stiffness requirements had been introduced to prevent flutter. The vertical stabilizer was proof tested satisfactorily for the condition producing the critical shear and bending moment (References 20, 21, and 22). A component static proof test was conducted satisfactorily on the rudder for a load corresponding to maximum (300 pounds.) pilot effort.

4.7 LANDING GEAR

Conventional landing and taxiing loads were calculated in accordance with MIL-A-8862 for 9200 pounds and 12,500 pounds gross weights, with 10 ft/sec. and 6 ft/sec. sinking speeds, respectively. In addition, vertical landing loads were calculated for 9200 pounds gross weight with 10 ft/sec. sinking speed.

A general computer program was developed for the main gear which yielded internal loads in all members including reactions at the fuselage. A summary of these loads for all landing and taxiing conditions may be found in Reference 19. Both nose and main landing gears were stress analyzed. Results may be found in References 29 and 34.

The nose gear and its support structure were static tested satisfactorily on the airplane for the two critical conditions: 3-Point Landing (Spring-Back) and Ground Turning (W = 12,500 pounds). The main gear and its support structure were static tested satisfactorily on the airplane for the two critical conditions: 2-Wheel Tail Down Ldg. (Spring-Back) and Drift Landing. Test program requirements, procedures, and results may be found in References 20, 21 and 22.

4.8 CONTROL SYSTEMS

The primary flight control systems consist of conventional stick and rudder pedals mechanically connected to rudder, elevator, and to servo actuators which control the ailerons, wing-fan exit louvers and nose fan thrust modulator. The limit pilot stick/pedal forces specified in Reference 17 were as follows: 100, 200, and 300 pounds, respectively, for lateral, longitudinal, and directional control. The various methods of reacting these pilot forces were also specified in the criteria.

Internal load distributions and stress analyses were summarized in Reference 33 for the conventional flight control systems, which were also satisfactorily tested in the airplane by applying a limit load to the cockpit controls and reacting the load by locking the surfaces (References 20, 21, and 22).

The fan-powered flight primary control system is a fully powered, irreversible system consisting of a collective (lift) stick in addition to the conventional cockpit controls, which mechanically control hydraulic servo valve tandem actuators. The only significant forces applied to the mechanical systems from the pilot control to the servo valves result from the pilot-feel spring packages. Since these forces were relatively small, conventional flight internal load stress analyses defined design requirements. The wing-fan louver and nose-fan modulator actuating mechanisms were satisfactorily proof tested on the simulator.

The collective control stick was proof tested satisfactorily to 150 pounds in both up and down positions. Both throttles were also proof tested satisfactorily to 75 pounds aft load. These items were tested as installed in the airplane (Reference 22).

5.0 DESIGN AND CONSTRUCTION

5.1 GENERAL

The XV-5A was designed and constructed to applicable specifications and accepted aircraft standards. Ground and flight tests proved that the XV-5A was flightworthy.

5.2 MANUFACTURING

The XV-5A was fabricated according to good aircraft manufacturing practices. Welding, heat treating and the fabrication of Fiberglass parts were controlled by Ryan manufacturing process specifications, which meet military requirements.

5.3 FASTENERS

The XV-5A fasteners are commonly used, standard types. Special fasteners and unusual applications of standard fasteners are eliminated.

5.4 FINISH AND PROTECTION FROM CORROSION

Finish and protection from corrosion was accomplished according to Ryan Specification 14359-1, Finish Specification XV-5A. This document specified methods to protect the parts from weather, corrosion, erosion and contact with dissimilar metal.

5.5 QUALITY CONTROL

Inspection and quality control was accomplished under the requirements of MIL-Q-9858 and Ryan Aeronautical Company Quality Control Procedures. Inspection records for Ryan made parts, test reports and test data for purchased part are filed by Ryan or Ryan's vendors. These records are available for examination. For any part that deviates from engineering specifications, an MRB action report is filed with the Quality Control Department.

5.6 MATERIAL STRENGTHS

Material strength properties and design values for the materials used in the XV-5A were taken from MIL-HDBK-5 and MIL-HDBK-17.

5.7 FATIGUE

Since the design life of the XV-5A was 250 hours, few fatigue problems appeared to exist. An exception was the center fuselage section which is somewhat more highly stressed and is constructed of welded, high-strength steel. However, no problems are anticipated since ample fatigue allowance was incorporated in the space frame design. See Reference 32, Section IV, Structural Analysis of Center Fuselage and Engine Mounts. As normally expected, minor airframe repairs were necessary following the contractor's flight test program. These repairs (resulting from panel fatigue) were confined to the nonstructural canoe fairing under the wing fans.

6.0 PROPULSION SYSTEM

6.1 GENERAL

The General Electric supplied propulsion system was certified as flightworthy in the applicable reports noted below. Ryan-designed, purchased, and fabricated components conformed with applicable standards. The results of component tests, ground tests, wind-tunnel tests, and flight tests verify that this subsystem is flightworthy.

The Propulsion Installation (Ryan Drawing 143P004) consists of two G. E. J-85-5 (G. E. Drawing 4012028-411) axial flow turbojet engines fitted with diverter valves (Drawing G. E. 4012001-912). These valves permit diversion of the exhaust gas to the tailpipe (Ryan Drawing 143P008) or to the lift fans (G. E. Drawing 4012001-941 and -942) and the pitch fan (G. E. Drawing 4012001-940). The divider and pitch fan ducting permits balanced operation of the fans with gas flow from either engine.

Power Plant Installation

The J-85 engine is qualified to MIL-E-5007. The fans were subjected to a 50-hour qualification test program (G. E. Report X353-5B Propulsion System Flightworthiness Test Report).

6.1.1 Operating Characteristics

Operating characteristics of the propulsion installation are considered normal and satisfactory. Test results obtained thus far indicate minor restrictions are necessary on the present installation. Internal engine compartments are maintained well below temperature limits. In CTOL jet mode continuous operation within engine limitations is without restriction.

In the VTOL fan mode, the compartment temperatures are somewhat higher partly due to the reduced cooling airflow and partly due to the increased ambient air temperature caused by fan diffused exhaust gases, under certain wind conditions and wing fan louver angles. Initially, external temperatures caused severe restriction. By the addition of insulation where the aluminum structure could not be replaced with titanium or steel, the safe operating times have been significantly increased.

In general, the aircraft may be operated in either mode restricted by indication of overheating from a continuous loop overheat warning system which encompasses the structure surrounding the hot components throughout the aircraft, such as the fan scrolls, ducting, and tailpipe. However, at present, some time limitations are imposed for various configurations which would normally be considered transient. Fan cavities are limited to 120°C for fan flight and 150°C for turbojet flight.

Installed thrust appears to be better than design estimates.

In the initial stages of Phase I testing, engine compressor stalls were experienced; however, no compressor stalls have occurred after several modifications and engine adjustments. For additional details and discussion of the engine stall margins of the installation, refer to General Electric XV-5A published memorandums entitled Datem Sheets No. 2, 4, 18 and 20.

6.2 EXHAUST GAS DUCTING

The divider ducts (Ryan Drawing 143P013) and the pitch fan ducting (Ryan Drawing 143P029) were used during the qualification tests of all of the fans; they are still in satisfactory condition after approximately 130 hours of operation. The tailpipe (Ryan Drawing 143P012) and its flexible section (Ryan Drawing 143P032) were tested in conjunction with the above tests approximately 30 hours operation with no signs of deterioration. In addition, a considerable amount of ground operation and flight testing has been accomplished without incident. On this basis the hot gas ducting and tailpipe installation are considered flightworthy for the XV-5A.

The engine and ducting mounts were accepted by structural analysis (see Reference 32) and verified by ground and flight tests.

6.3 ENGINE INLET

The fiber glass engine air inlet (Ryan Drawing 143P006) was accepted by structural analysis (see Reference 32) and verified by ground and flight test.

6.4 ACCESSORIES

The accessory installation (Ryan Drawing 143P007) components have been qualified by individual testing (Table Components Qualification Data) as well as complete installation tests in conjunction with operation of XV-5A simulator program, approximately 400 hrs to date. The cooling fans, which are a part of the gear box-fan assembly, supply the cooling air requirements for static and hovering operations. The smaller fans cool the generators, the hydraulic oil coolers, the electrical compartments, the pitch fan ducting, and the pitch fan scroll areas. The larger fans cool the engine compartments, the divider ducts, and the wing fan scroll areas. Ground and flight tests indicate satisfactory temperature limits are maintained when aircraft is operated within design limits.

6.5 FUEL SYSTEM

The fuel system (Ryan Drawing 143P009) can supply fuel directly to each engine from its tank, with cross feed provisions. The engine pumps can draw fuel from the tanks up to 6000 feet without booster pumps. Over 6,000 feet the booster pumps are required; each pump is capable of supplying both engines. A capacitance-type fuel quantity gauge indicates fuel directly in pounds. A float switch-operated warning light indicates low fuel level. Booster pump inoperative and low fuel pressure are indicated by pressure switch-operated warning lights.

The fuel tank vent installation (Ryan Drawing 143P070) vents all fuel tanks to atmosphere and maintains a positive relative pressure on the surface of the fuel between 0 to 3 psi under all conditions of flight. Float valves prevent siphoning at extreme attitudes.

Fuel booster pumps are powered by engine compressor 8th stage bleed air which is controlled by a normally open solenoid valve (Ryan Drawing 143P059).

6.6 FIRE PROTECTION

The fire extinguisher system (Ryan Drawing 143P017) consists of two CF-2D twin valve pressure vessels using CF_3B_R (MIL-B-13318) pressurized to 600 psi. The CF_3B_R can be discharged into the forward and aft engine compartments of either engine selected at a high discharge rate. The concentration of agent in the protected compartments exceeded the requirement as measured by FAA conducted test.

Firewall installation (Ryan Drawing 143P036) provides barrier between each engine, between engine and fuel tanks, and between divider ducts and fuel tank. A lateral firewall (Ryan Drawing 143P049) was constructed between the engine compressor and turbine compartments.

A drain system (Ryan Drawing 143P052) carries away combustible fluid from the engine and ducting and safely disposes of them below the aircraft.

6.7 PITCH CONTROL DOORS

The pitch control doors (Ryan Drawing 143P034) control the direction of discharge of the air accelerated by the pitch fan, thereby controlling the longitudinal attitude of the aircraft. Tests at the G. E. Evendale test facility in conjunction with pitch fan acceptance and qualification and subsequent flight tests, have verified design integrity.

7.0 EQUIPMENT

7.1 GENERAL

All equipment items installed on the XV-5A have been qualified individually or by similarity to the applicable government specifications.

Compatibility of these equipment items with the intended use was assured during design and testing of the XV-5A aircraft.

Ryan Engineering and Quality Control Departments feel all efforts have been implemented to establish that the equipment installed in the XV-5A are flightworthy.

7.2 INSTRUMENTS

The choice of instruments was dictated by the flight requirements as set forth in Reference 2. The display is suitable for experimental, ferry and for VFR conditions.

7.2.1 Flight Instruments

Installed are the basic flight instruments (airspeed, altimeter, and magnetic compass), instruments required for special flying, instruments normal to medium performance jet aircraft (vertical speed, turn and bank, attitude, and acceleration).

In near hovering flight, the angle-of-attack indicator becomes a primary indicator. Other instruments required are position indicators which provide position readings for louver vect angle, flaps, thrust spoiler and trim in longitudinal, lateral, and yaw directions for both fan mode and jet mode flight (Ryan Drawing 143K005, sheet 2).

All flight instruments are qualified under military or FAA specifications.

The pitot-static system is installed according to MIL-I-6115A and has been calibrated with the instruments installed. This system comprises a boom mounted at the end of the left hand wing, piping (with moisture drains), and manifold piping in the cockpit with hoses for instrument attachment. The pitot line has a pneumatic solenoid valve so it can be shut off from the low airspeed instrument when the aircraft exceeds the speed limit of the low airspeed instrument.

7.2.2 Power Plant System Instruments

The power plant instruments are typical of instruments normally found in dual engine jet aircraft as to mounting and type. The instruments are mounted from top to bottom with left engine instruments on left and right engine instruments to the right. The powerplant instruments comprise: tachometer, exhaust gas temperature, fuel flow, fuel quantity and oil pressure. The instruments have been qualified in other aircraft with equivalent or more stringent requirements (see component qualification data Table 16).

Power levels during fan flight are monitored by fan RPM indicators. Fan cavity temperatures also are indicated for CTOL flight.

7.2.3 Instrument Arrangement

The instrument arrangement minimizes pilot effort for both flight modes, and the arrangement follows the standard display for military aircraft (MS33634 and MS33635). The instrument panel is not shock mounted, for flight demonstration has proven that shock mounting increases the need for panel vibrators to decrease hysteresis inaccuracies in instrument readings. Design is in accordance with applicable sections of MIL-I-5997B.

7.3 ELECTRICAL SYSTEM

7.3.1 General

The design and installation of the aircraft electrical system complies with MIL-E-25499 (General Specification For) and the following additional specifications:

MIL-B-5087	Bonding Electrical, For Aircraft
MIL-D-7006	Detecting Systems, Fire, Aircraft
MIL-E-5400	Electronic Equipment, Aircraft
MIL-E-7017	Elec. Load Analysis, Method For
MIL-STD-704	Electrical Power, Aircraft
MIL-I-7032	Inverter, Aircraft
MIL-W-5088	Wiring, Aircraft, Installation of
MIL-E-5272	Environmental Testing
MIL-A-8064	Actuators and Actuating Systems, Aircraft

Specification MIL-E-7080 was used as a guide for the selection of applicable electrical equipment and installation.

7.3.2 Generating

Primary power on the aircraft is 28 vdc. The 115 vac, 400 cps power is provided by two 250 va Mil Std rotary inverters. A 28 vdc (nominal) silver-zinc secondary battery is a source of emergency (primary) power. The battery is small, rated at 25 ampere-hours, but provides more than adequate power yield for "generators out" emergency (see Navy Specification 143E011 Load Analysis), as well as adequate capacitance for good bus regulation (ripple). Electrical power characteristics are within the confines of Mil-Std-704 (exhibit "Component Qualification Data").

Two "brushless" type d-c generators are used in an equalizing circuit. Together at normal power plant speeds they provide a 9 kw power source; actual steady state power required is approximately 2.8 kw. Systems growth as well as good circuit clearing capability is therefore obtained. The installed system is much simpler and lighter in weight than an equivalent conventional "brush" type generating system.

7.3.3 Distribution

Power is controlled and distributed through a closely coupled control module and circuit breaker panel. The largest circuit breaker is rated 25 amperes. All dc bus control contactors are Mil Std type. The generator controller modules and the generator housings incorporate "feeder fault" protection relays. The sensitivity of the feeder fault relays and the circuit breakers have been sized to the capacity of the distribution leads. Distribution to all systems is by "open wire" harness constructed with Mil Std wiring. The number of connectors used has been minimized for best reliability. For the most part harnesses are separate with respect to discrete function: power, primary signal, standby signal, etc. with exceptions to best attain system neatness, compactness, integrity, and reliability.

7.3.4 Conversion Control Interlock System

The conversion control interlock system comprises the bulk of the aircraft's electrical distribution network. The system distributes the cockpit signals/pilot commands to the electrical mixer/integrating, and subsequently to the electrically powered control devices/interlock switches, actuators, solenoid valves, etc. The Electrical Mixer

Ryan Drawing 143E012-1 is the center of the control network. A Tester, Ryan Drawing 143G021-1 permits functional inspection of the electrical mixer before aircraft installation. This tester is now used periodically as ground support maintenance and inspection equipment. Compliance to operational specifications has been established, and detailed. Refer to Reference 40, paragraph 3.1.

7.3.5. Stability Augmentation System

Requirement

The stability augmentation system requirement is presented in paragraph 3.3.2.1.1 of the Detail Aircraft Specification, Report Number 62B125A, dated 30 December 1964. The system shall stabilize the aircraft in pitch, roll and yaw during the fan mode of flight. This is to be accomplished through the use of rate gyros to electrically control the wing fan exit louvers and pitch fan door actuators.

Compliance

Satisfactory compliance with the requirement has been demonstrated both in the flight simulator program (Reference 38), and in actual flight test operation. Flights have been made to determine handling characteristics with the system inoperative. The simulator program indicated the possibility of handling difficulties with the roll axis stabilization inoperative. On this basis roll axis testing was limited to gain variation.

7.3.6. Components

All components are standard AN/MS parts, except those listed under Components Qualification Data, Table 16.

7.4 HYDRAULIC SYSTEM

7.4.1. System Design

The hydraulic system was designed in accordance with MIL-H-544C, Type II, Class 3000 psi system. Two completely independent, engine driven systems are provided. Both systems operate continuously and each is capable of supplying full control load requirements in event of pressure loss of either system.

All primary and secondary flight control functions, that are hydraulically powered, are controlled through tandem hydraulic cylinders, except the thrust spoilers. The thrust spoiler actuator is powered by hydraulic system No. 1 only with internal locks provided to lock the spoilers in the retracted position in the event of system No. 1 pressure loss. The landing gear control actuators are also powered by hydraulic system No. 1 only, with emergency extension provisions from the emergency pneumatic system.

7.4.2 System Installation

Ryan drawings 143H001 through 143H010 show the hydraulic system installation. Ryan drawing 143H002 is the system schematic drawing. Compliance of the installed system to specification requirements has been demonstrated on the hydraulic and control simulator and during ground and flight tests of both aircraft.

7.4.3 Components

Wherever possible, AN or MS hydraulic components, or components previously qualified for another user, were selected for use in the system. Where the requirements did not lend themselves to this approach, specification control drawings were prepared and components were designed, fabricated, and qualified to the requirements of these drawings. A tabulation of the components falling into this category is contained in Table 16.

7.5 CONTROL SYSTEM

7.5.1 System Design

The control system was designed essentially in accordance with specifications MIL-F-8785, MIL-F-9490B, and MIL-S-8698. The control system was designed as simple, foolproof, and reliable as possible consistent with the intended mission of the aircraft.

The control system consists of conventional stick, rudder pedals and collective lift stick mechanically connected to aerodynamic flap type control surfaces, for conventional flight control; and connected to wing fan exit louver servos and a pitch fan thrust modulating door servo for fan flight mode control.

Operating loads of the fan flight mode control functions necessitated the use of powered subsystems. These subsystems were all designed with tandem cylinder actuation for reliability.

The conventional flight controls were designed for manual operation to enable a conventional landing even in event of failures in both hydraulic systems. (The ailerons required hydraulic boost for maximum maneuvering performance, but are still controllable without hydraulic power.)

7.5.2 System Installation

Installation of the control system is shown on Drawing 143C001, and the system schematic is shown on Drawing 143C002. Compliance of the installed system to specification requirements has been demonstrated on the hydraulic and controls simulator, during the aircraft installed systems tests (Reference 40) and during ground and flight tests of both aircraft.

7.5.3 Components

All control cables and pulleys used in the conventional control systems conform to military specifications. All bearings used in both the conventional and fan flight controls are precision, low-friction bearings manufactured to closer tolerances than their AN counterparts covered by specification MIL-B-7949. Cable tension regulators in the elevator and rudder systems are both qualified by similarity to Pacific Scientific Co. Regulator R75-11001-100-00, Qualification Test Report 352.

The remaining control system components are hydraulically powered (refer to paragraph 7.4).

7.6 COCKPIT

The arrangement of the cockpit was made in accordance with applicable sections of ARDCM 80-1 Vol. 1 Handbook of Instructions for Aircraft Designers and MIL-STD-803 Human Engineering Criteria. It was then studied in a mockup and the hydraulic and controls simulator to assure correct arrangement and distances for pilot minimum effort with maximum efficiency.

For safety reasons all equipment mounted in the cockpit is installed to withstand 30g crash conditions. All glass is of a nonsplintering type (windows are of Plex 55).

Vision is at a maximum with minimum distortion and glass areas have a luminous transmittance in excess of 70 percent. Precautions have been taken to reduce bothersome reflections.

Controls locations and actuations are in accord with applicable sections of MIL-STD-250A Cockpit Controls Location and Actuation, for Helicopters and MIL-STD-203 Cockpit Controls Location and Actuation for Fixed Wing Aircraft.

A diluter/demand low pressure gaseous oxygen system is incorporated and is sufficient to supply oxygen, up to 100 percent, for the pilot throughout any possible aircraft mission.

7.7 LANDING GEAR

7.7.1 Main Gear

The basic fuselage mounted configuration of the main landing gear was dictated by consideration of minimum wing weight. Considerations peculiar to the fan-in-wing concept led to the use of a two position shifting mechanism, providing two-station positions for the main wheels.

7.7.2 Nose Gear

The nose gear design concept is entirely conventional. Power steering is not provided.

7.7.3 Loads

Loads were calculated by Ryan Structures Group and computing facilities as outlined in Section 4.10.

7.7.4 Stress Analysis

Detail stress analysis of landing gear structural components was performed by landing gear vendor and checked by Ryan Structures Group.

7.7.5 Shock Absorption

Shock absorbers were designed in accordance with MIL-S-8552. The shock absorbers are of conventional oleo pneumatic configuration using metering pin orifice control. Satisfactory performance of nose and main shock absorbers was demonstrated by drop tests using vendor's test tower facilities. References giving detailed description of test equipment, instrumentation, and results are listed in the Component Qualification data, Table 16.

7.7.6 Nose Gear Shimmy Suppression

Early difficulties involving dynamic instability of the nose wheel installation were encountered during high speed taxi tests and required detail redesign of the nose wheel fork, torque links, and shimmy damper installation. An intensive development and testing program was launched and satisfactorily completed prior to first flight.

7.7.6.1 General

A group formed of Ryan/Republic dynamics and design engineers reviewed the shimmy suppression requirements, and assisted Loud Co., (landing gear vendor), with the necessary redesign. Detail design recommendations were made, chiefly in the design of the lower end of the forks to provide a higher spring rate, a more positively retained axle, and a higher damping ratio damper.

7.7.6.2 Testing of Original Configuration

(a) Loud Co. obtained the use of the Lockheed wheel spin test facility at Rye Canyon. The gear and shimmy damper from ship No. 1 were installed on the drum test machine. A series of runs were performed, witnessed by the Ryan/Republic engineers. These tests confirmed the existence of a self-sustaining shimmy tendency of the gear.

(b) Loud Co. made torsional stiffness measurements of the nose gear fork, torque linkage, and shock strut assembly to aid in redesign analysis.

(c) Loud Co. made shimmy damper dynamic test runs to establish the damping coefficient, as originally installed, to provide comparison data with the redesigned damper and to support Ryan's analysis effort.

7.7.6.3 Theoretical Investigation

This effort was conducted by the Ryan/Republic engineers. The investigation consisted of an application of Moreland's theory and analytical techniques to the XV-5A airframe/nose gear configuration. As data became available from the various phases of testing, it was incorporated in the analysis. First objective of the analysis was to obtain a correlation between the shimmy condition experienced with the aircraft using Lockheed spin test results, and theoretical predictions. The effect of variations in the major parameters such as gear torsional and lateral stiffness, damping coefficient load, tire elastic effects, caster and trail, taxi speed, shimmy frequency, mass distribution, etc., were then studied. The investigation originally made use of the Ryan digital computer. Work was transferred to the Ryan analog computer due to the rapidity with which the effect of varying parameters could be studied with this facility.

The first computer run gave an excellent correlation with actual experience. It also indicated that with the original gear configuration, virtually no amount of damping would provide stable operation at the nose gear maximum required taxi speed. Design recommendations regarding increase of nose wheel fork and torque linkage torsional stiffness and shimmy damper coefficient at various shimmy frequencies were transmitted to the Loud Co. and used for analysis of the redesign components. The most critical condition appeared to be the light load condition at point of take-off or immediately after touchdown. This showed the existence of an upper limit for the usable damping coefficient. Development testing on the shimmy drum, and frequency and stiffness measurements of the gear installed on the airplane subsequently showed this upper boundary to be imaginary, which removed the "suspect" critical condition.

7.7.6.4 Redesigned Nose Gear Development and Shimmy Testing

(a) Nose Gear Torsional Stiffness

New wheel fork and torque links were made available for testing at Loud Co. These items were designed in accordance with Ryan/Republic stiffness requirements. After one week's continuous testing and development the nose gear assembly achieved satisfactory stiffness measurements. Additional stiffening of the shimmy damper mounting brackets and the torque link/damper shaft connection was required.

(b) Shimmy Damper Development

Testing of the original shimmy damper with minor modifications proved that the shimmy damper could not achieve the damping coefficient values required by the Ryan/Republic recommendations.

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A new damper with improved vanes was designed, built, and tested. Sufficient testing was performed to indicate the damper would meet the predicted requirements.

(c) Shimmy Drum Test

The stiffened nose gear with the new damper was installed on the Rye Canyon shimmy test machine. Tests performed included:

1. Driven shimmy and damper performance.

2. Unrestrained shimmy up to maximum taxi speed with optimum damper adjustment.

3. Unrestrained shimmy with reduced damping and taxi speed.

4. Lateral bending stiffness, natural frequency response and decay rate measurements.

At the conclusion of these tests the gear/damper combination was regarded as satisfactory over the entire load/speed spectrum. Also confirmation of the lower boundary of acceptable damping coefficient was obtained.

(d) Airframe/Nose Gear Matching (Computer Study)

The shimmy drum testing did not confirm the existence of an upper limit of damping coefficient. The modified nose gear was installed on ship No. 2 at Edwards AFB and lateral stiffness and frequency response measurements of the complete system were obtained. The data from these tests and the Rye Canyon shimmy drum testing were incorporated in a refined analog computer study. This study indicated satisfactory compatibility between airframe and nose gear and removed the upper boundary limit on acceptable damping coefficient. To provide additional margin, an increased figure of minimum damping coefficient was recommended as a desirable target for the shimmy damper qualification test, which was met.

(e) High Speed Taxi and Flight Testing (Ship No. 2)

High speed taxi tests with the modified gear and shimmy damper tests were successfully completed 5-22-64.

7.7.7 Main Gear Two Position Mechanism

The portions of the mechanism designed and manufactured by Ryan were subjected to detail stress analysis by Ryan Structures Group, and successfully passed static proof tests.

7.7.8 Brakes

MIL-W-5013 was used as a general guide in the preparation of the Ryan drawing SCD L0003 lightweight wheel and brake specification. Considerations of minimum hovering take-off weight and the specialized mission of the aircraft, led to the acceptance of a limited number of stops between relines. Kinetic energy requirements were determined by Ryan Engineering Group and satisfactory compliance demonstrated on the vendor's dynamometer equipment. For detailed report references, see Component Qualification Data, Table 16.

7.7.9 Landing Gear Doors and Mechanisms

The landing gear doors and mechanisms were designed by Ryan Engineering to conform to inflight operating loads predicted by Ryan Aero dynamics Group. Detailed stress analysis was performed by Ryan Structures Group.

7.7.10 Actuators

All actuators in the landing gear retraction, two-position, door and uplatch mechanisms are of simple straight forward design. Restricted functional and proof testing were performed by vendors and repeated in Ryan Hydraulic Test Laboratory.

7.7.11 Landing Gear Operation

(a) Normal hydraulic system components, see Section 7.4.

(b) Emergency pneumatic system. The emergency system provides a completely independent means of extending the landing gear. This system is manually initiated and will lower the gear even in the event of complete electro/hydraulic system failure.

(c) Hydraulic and pneumatic system components are listed in the Hydraulics and Controls Component Qualification Data Tables.

7.7.12 Functional Testing

Complete system testing of installed landing gear and auxiliaries was performed prior to commencement of flight testing. Procedures and results are contained in References 39 and 40.

7.7.13 Structural Integrity

Static testing by Ryan Test Group demonstrated satisfactory structural strength and stiffness of landing gear and auxiliary components. Procedures and results are given in References 20, 21, and 22.

7.8 SAFETY PROVISIONS

7.8.1 Pilot Ejection Seat

The pilot's ejection seat was chosen for its capability at zero speed and zero altitude. These characteristics are important to an aircraft which would spend much time near the ground at speeds between 0 and 50 knots. The seat, an LW-2 North American Aviation, Columbus, Ohio, ejection seat system was developed by NAA, under U.S. Army contract, for aircraft such as the XV-5A and was the best available escape system to be found, relative to the XV-5A aircraft flight envelope. This system was test fired twice under simulated XV-5A conditions over and above the development and demonstration test program. Further discussion of these tests and the development program can be found in North American Aviation report NA63H-817.

7.8.2 Provisions for Exterior Canopy Opening

Provisions for opening the pilot's canopy by the ground crew from the outside have been made. From either side of the fuselage, with one continuous motion, the latch mechanism may be operated and the canopy opened by a person standing on the ground. This provision follows the recommended procedures found in HIAD.

7.8.3 Emergency Egress by the Pilot

If for any reason, the pilot is unable to unlock the canopy and he does not choose to eject through it, a heavy knife has been provided so that he can break the canopy glass to climb through it. (Plexiglass 55 will not shatter and cause injury from sharp pieces.)

7.8.4 Fire Protection

7.8.4.1 Overheat Detection System And Firewarning

A temperature sensitive two-wire system is installed in those compartments where overheat may occur, to set off a flashing light warning in case of overheat. In the case of fire, with its attendant higher temperatures, the steady light warning is given. These warning lights are located on the instrument panel, one for left, and one for right engine compartment.

7.8.4.2 Fire Extinguishing System

The fire extinguishing system is comprised of two bottles of two pounds each bromotrifluoromethane (MIL-B-12218) plus valves and piping to each engine compartment with selection of one or both to either compartment. This system is controlled by the pilot from a standard arrangement on the instrument panel.

7.8.5 Spin/Drag Chute

A spin chute or high speed chute has been included for flight test program. This feature is not used for normal braking but is reserved for emergencies. The choice of high speed chute or spin chute installed must be made prior to takeoff according to the planned test program.

7.8.6 System Emergency Shut-Off Switches

To preclude catastrophic runaway on the horizontal stabilizer and on the louver vector angle a warning light has been installed and shutoff switches have been included. The switches open the power circuit at the actuator to shut off any possible system failure. These two possibilities were discovered as potentially uncontrollable during the simulator flying test program.

7.8.7 System Redundancy

The control systems have been designed redundant so that the aircraft could fly with any one power system out. The loss of one system in hydraulic, electrical control, or stability augmentation systems will not be noticeable to the pilot except as indicated by condition lights on the main instrument panel.

7.8.7.1 The hydraulic system is dual throughout and either side is capable of complete control (see 7.4.1).

7.8.7.2 The electrical system is, in the control areas, dual redundant, and in the power source, is triple, so that even if two generators fail, the emergency buss is still powered by a battery, (see Section 7.3).

7.8.7.3 The SAS (Stability Augmentation System) is dual with one system variable by the pilot and a backup system previously set to a safe control mode. System changeover is automatic (see Section 7.3.6).

8.0 OPERATING LIMITATIONS AND INFORMATION

8.1 OPERATING DEMONSTRATED LIMITATIONS

8.1.1 Flight Limitations

The following paragraphs present a summary of the limiting flying speeds. For a more detailed coverage of all limitations, see Ryan Report No. 64B150, Pilot's Operating Manual.

8.1.1.1 Maximum Flight Speed 400 KEAS

The aircraft has been flight tested to this condition.

8.1.1.2 Maximum Approach Speed 180 KEAS

This speed applies to the extension of flaps, landing gear, nose fan pitch doors, inlet louvers, and exit louvers.

8.1.1.3 Conversion Speed

Fans to Turbojet	85-95 KIAS
Turbojet to Fans	95-105 KIAS

8.1.2 Power Plant Limitations

For the ground crew and others responsible for the aircraft or power plant, refer to T.O.-2J-J85-56.

OPERATING LIMITS

Item	Limits	Remarks
	NOTE! This figure supplies the maximum or minimum values for Standard Day Sea Level Static Conditions.	
1. STARTING TIME	20 to 50 Seconds (Nominal)	Starting time shall be measured from the initial tachometer indication of engine RPM to stabilize idle speed.

Item	Limits	Remarks
2. ENGINE SPEED IDLE SETTING	48(±1)% RPM 70% RPM Fan Mode	Conventional Mode
IDLE SPEED FLUCTUATION	3% RPM (peak to peak)	Within the 46.5-49.5% RPM range.
MILITARY SETTING	102(±1)% RPM for 30 minutes	

NOTE!

The military speed setting (102 (±1)% should be repeatable at any given condition. A change in condition will affect repeatability. Approximately 30 minutes continuous operation at Military setting is allowable.

MILITARY FLUCTUATION	±1% RPM	
OVERSPEED	104% RPM Transient	See Maintenance T.O. for Engine Rotor Disposition if speed exceeds 104%.

NOTE!

Normal max during start 800°C.

Pilot should chop throttle if EGT goes to 890°C on fire-up.

3. EXHAUST GAS TEMPERATURE (EGT) STARTING	980°C MAX (1800°F MAX)	

Item	Limits	Remarks

CAUTION

If EGT is consistently high during starting, check AIS diverter valve for full opening.

Item	Limits	Remarks
IDLE	550 to 600°C (1202°F) MAX	
MILITARY	680 $\left(^{+5}_{-10}\right)$°C 1256$\left(^{+9}_{-18}\right)$°F	Limits for steady-state operation. For overtemperature limits during start and other than start, see Maintenance T.O.
4. OIL PRESSURE		
IDLE	5 psig (MIN) 20 psig (MAX)	On cold starts, up to 185 psig is allowed.
MILITARY	20 psig (MIN) 50 psig (MAX)	Not more than 10 psig (MAX) change from Normal is allowed.
FLUCTUATION	±2 psig	10 psig (MAX) change.
5. IGNITION GENERATOR DUTY CYCLE	A. 2 minutes on, 3 minutes off. 2 minutes on, 23 minutes off. B. (alternate) 5 minutes on, 55 minutes off.	Select either cycle.

FAN LIMITATIONS

Nose Fan - Limited to 105% RPM Steady State

Wing Fans - Limited to 100% RPM Steady State

NOTE!

See Section I, Subsection 2.0 for overspeed protection system.

8.1.3 Weight Range and Center of Gravity

8.1.3.1 Weight and Balance

The ranges of weight and center of gravity within which the airplane may be safely operated are presented in the Reference 37.

Low fuel does not adversely affect the balance or stability of the airplane.

8.1.3.2 Use of Ballast

Provisions for removable ballast have been incorporated in the aircraft. If the airplane c.g. should exceed the forward limit, lead ballast may be installed on the platform and retaining bolts provided in the aft electrical compartment at approximately Fuselage Station 481. A maximum of 200 pounds may be installed, as shown on Ryan Drawing 143D068.

Should the aft c.g. limit be exceeded, ballast up to 600 pounds may be added at the observer's or instrumentation location.

8.1.3.3 Empty Weight

The empty weight and corresponding center of gravity location shall include all fixed ballast, the unusable fuel, undrainable oil and hydraulic fluid. The weight and location of the above items (not including ballast) and items of equipment may be found in Reference 37

8.1.3.4 Maximum Weight

The maximum design weight of the airplane is 9200 pounds at the limit load factor of 4.0 (ultimate 6.0). The weight may be increased above 9200 pounds if the load factor is proportionally reduced so the η W product remains constant.

8.1.3.5 <u>Minimum Weight</u>

The minimum weight for the airplane is 8236 pounds.

8.1.3.6 <u>Center of Gravity Position</u>

With no fuel in the extended range belly tank, it is not possible to exceed the c.g. limits if the empty fuel - gear down, and full fuel - gear up, conditions are within limits, and providing fuel is consumed equally from the forward and aft tanks.

With the extended range belly tank fuel aboard, it is possible to exceed the center of gravity limits. Proper loading and recommended fuel consumption for this condition may be derived from the Center of Gravity Travel Graph on Page 151 of Reference 37.

9.0 RELIABILITY DATA

9.1 SIMULATOR COMPONENT FAILURE STUDY

9.1.1 Scope

This section of the flightworthiness report presents the results of an investigation of the effects of simulated component failure in the XV-5A Flight Research Aircraft. The purposes of this investigation include the effects of component failures on system performance and aircraft behavior to establish envelopes of recovery capabilities, and development of recovery techniques and familiarization of pilots with failure symptoms and recovery procedures. (Refer to Reference 38 for a complete description of simulator configuration and data limitations).

9.1.2 Component Failure Mode Analysis

The first part of the failure mode analysis was to review each airplane system to identify those component/system failure modes most likely to occur, and most likely to adversely affect airplane operation. This review identified 85 component/system failure modes. Further analysis of the expected effects of failure, and suitability of simulation, resulted in selection of 45 significant component/system failure modes to be investigated on the simulator. In general, those failure modes not simulated were for one of the following reasons:

1. The failure mode was beyond the scope of the simulator to accomplish a valid simulation due to configuration limitations, or the range of validity of the simulator controlling data.

2. The effect on the airplane of failure modes of several different components could be duplicated by simulating one particular component failure mode.

3. Some component failure modes were found to be not hazardous after the effects of failure analysis was completed.

The results of the component/system failure mode analysis are summarized in Table 1 following. The effect of failure, for each failure mode, for each component/system is presented. Codes were assigned to each failure mode to simplify annotating data tapes.

TABLE 1
COMPONENT FAILURE MODES

SYSTEM OR COMPONENT	FAILURE MODE	CODE	EFFECT OF FAILURE	REMARKS
Gas Generator	1) Ream out	E12	Instant gas power loss including hydraulic, electrical and cooling blower on that engine.	Modes 1, 2, and 3 look the same on simulator. (Air restarting cannot be simulated.)
	2) Flame out	E12	Instant gas power loss, hydraulic, electrical and cooling blower run down as function of windmilling. (Air restart may be possible.)	
	3) No accessory power from PTO pad	E12	Same as 1 above plus no engine driven fuel pump (air restart not possible).	
	4) Fuel control oscillation	F1	Oscillating gas generator RPM's and power output	
Hydraulic System No. 1	1) No mechanical power power input, or no hydraulic power out due to pump failure or fluid loss.	*	Loss of following functions: a. Pitch axis stability augmentation b. Normal landing gear operation c. Thrust spoiler actuation d. Slightly increased time constants for remaining hydraulic system functions.	* Not simulated because previous simulation work indicated no problems and pressure loss always part of single engine failure. Note: 1. Stab. Aug. failure done separately (see run codes A3, B3 and B4).
Hydraulic System No. 2	1) No mechanical power input, or no hydraulic power out due to pump failure or fluid loss.	*	Slightly increased time constants for hydraulic functions in tandem with Hydraulic System No. 1.	* Same as Hydraulic System No. 1 except does not affect pitch SAS.
Horizontal Stabilizer Control	1) Immobile actuator (mechanical)		Fixed pitch trim condition and conversions not possible.	
	2) Directional control valve coil circuit open. (Either coil - i.e. up or down - in either valve - i.e. No. 1 Hyd. or No. 2 Hyd. System) or either valve jammed in neutral.	E13* E14*	Stabilizer travels at slower (single system) rate in direction(s) affected.	* (E13 up coil open) * (E14 down coil open)
	3) Control valve hard over (electrical short or mechanically jammed)	E1* E2*	Runaway stabilizer in direction affected.	* (E1 hard down) * (E2 hard up)
	4a) Bypass valve (-5 restrictor) failed open.	E4	a.) CTOL trim rate faster than nominal rate, VTOL and conversion trim rates normal.	No high speed CTOL simulations to test effect of failure.
	4b) Failed closed	*	b.) CTOL trim rate normal, VTOL and conversion trim rates less than nominal (may or may not affect conversion capability)	* (See E19 and E22)

TABLE 1 (Continued)

SYSTEM OR COMPONENT	FAILURE MODE	CODE	EFFECT OF FAILURE	REMARKS
Horizontal Stabilizer Control (Cont'd.)	5a) Bypass valve (-3 restrictor) failed open.	*	a.) Conversion trim rate normal, CTOL VTOL trim rates faster than nominal.	* Similar to E4
	5b) Failed closed	*	b.) CTOL and VTOL trim rates normal, conversion trim rate less than nominal (may or may not affect conversions).	* Similar to E4
	6a) *Both bypass control valves failed open (in one hydraulic system only).	-	a.) Conversion trim rate normal, both CTOL, VTOL trim rates faster than nominal. (Greatest effect in high speed flight.)	* Double failure. Note: High speed (CTOL) flight failures not simulated.
	6b) Failed closed (in one hydraulic system only).	E22	b.) CTOL trim rate normal, both VTOL and conversion rates less than nominal.	
	7a) Stabilizer rate switch failed short.	E17	a.) Permits conversion regardless of stabilizer trim rate.	
	7b) Failed open.	E16	b.) Interrupts conversion regardless of stabilizer trim rate.	
	8a) Conversion position limit switch failed open.	*	a.) No stabilizer driver from the affected hydraulic system.	* Failure effect same as directional control valve open coil (see E13 and E14).
	8b) Failed short	*	b.) Overrun programmed conversion cycle end point. Significant only during CTOL to VTOL conversions.	* Not simulated because of potential damage to actuator.
	9a) Stick grip trim switch failed open.	*	a.) No pitch trim change possible in direction affected.	* Not considered catastrophic failure.
	9b) Failed short	*	b.) Runaway stabilizer in direction affected.	Similar to control valve hardover or jammed (E1 and E2).
	10) Conversion trim rate too slow	*	Interrupted conversion (diverter valves cannot change due to low rate then stabilizer program is stopped because diverter valves do not change).	* Similar to E16
	11) Diverter Valves NO GO	*	Interrupted conversion (stabilizer program stopped because diverter valves do not change).	* Similar to E16

TABLE 1 (Continued)

SYSTEM OR COMPONENT	FAILURE MODE	CODE	EFFECT OF FAILURE	REMARKS
Stability Augmentation System	1) Single axis no output	B1) B2) B3)	Loss of stabilization in that axis.	Overall effect on airplane flying qualities uncertain.
	2) Single axis hard over	A1) A2) A3)	Apparent trim change in axis affected followed by no stabilization in that axis.	Overall effect on airplane flying qualities uncertain.
	3) All axes dead	B4	No stability augmentation (most critical at lower VTOL speeds.	Overall effect on airplane flying qualities uncertain.
	4a) Maneuvering switch failed open.	D2*	a) No change from holding gain to maneuvering gain in 1/2 axis affected.	Aircraft is expected to feel "stiffer" in half axis affected. *(All SAS channels in hold was simulated fail.)
	4b) Maneuvering switch failed closed.	D1*	b) No change from maneuvering gain to holding gain in half axis affected.	Aircraft is expected to feel looser in half axis affected. *(All SAS channels in maneuvering was failure simulated.)
	5a) Electro/Hydraulic servo-valve coil failed open.	C1	a and b) No appreciable effects due to bridge circuit in roll/yaw axes and paralleled coils in pitch axis.	(Roll/Yaw only)
	5b) Coil failed short	C2		
	6a) Single electro/hydraulic servo-valve hard over (Electrical, hydraulic, or mechanical) in roll/yaw axes.	*	a) Small trim change followed by small reduction of SAS gain in roll/yaw axes.	* See single axis
	6b) Pitch axis.	*	b) Trim change in direction affected followed by no pitch axis stabilization.	*See single axis hard over.
	7a) No hydraulic power from No. 1 Hydraulic System.	*	a) Loss of pitch axis stabilization and reduced gain in roll/yaw axes.	*No special run. See single engine recoveries.
	7b) From No. 2 Hydraulic System.	*	b) Reduced stabilization gain in roll/yaw axes.	*Same
	Both roll/yaw coils (on one louver servo-actuator) Open	C3	Test on simulator for flight effects.	
	Both roll/yaw coils (on one louver servo-actuator) shorted.	C4	Test on simulator for flight effects.	

TABLE 1 (Continued)

SYSTEM OR COMPONENT	FAILURE MODE	CODE	EFFECT OF FAILURE	REMARKS
Stability Augmentation System (Cont'd.)	One roll/yaw coil open and one coil short (on one servo-actuator).	C5	Test on simulator for effect.	
Throttle Cut-Back System	1) Pilot permits fan overspeed condition to develop.	F2	Throttle setting automatically reduced to 70% power position on both engines.	
	2) False overspeed signal.	E10	Throttle setting automatically reduced to 70% power position on both engines.	
	3) Overspeeding fan(s) not slowed down sufficiently to accomplish reset.	*	Both throttles cut-back to 70% power setting when reset button released.	*Not simulated
Diverter Valves	1) CTOL to VTOL first motion interlock switch failed short.	F3	Stabilizer control circuit sees diverter valve in VTOL position if true or not (and would continue programmed trim change <u>without</u> diverter valve operation).	Requires double failure to be a problem (e.g. diverter valve "NO GO").
	2) VTOL to CTOL first motion interlock switch failed short.	F4	Same as F3 except control circuit sees diverter valve in CTOL positon.	Same as above
	3) Diverter valve sequencing time delay relay failed short (CTOL to VTOL only)	F5	Diverter valve operates 0.3 second early (concurrent with stabilizer "programmed rate" signal, i.e. No time delay).	
	4) Diverter valve time delay relay fails open.	F6	Interrupted conversion from conventional mode to fan mode due to no diverter valve operation.	
Wing Fan Doors	1) Door(s) fail to open (CTOL to VTOL)	*	Interrupted conversion. Possible adverse roll, pitch or yaw moments during attempted conversion maneuver (15 possible combinations of events)	*Not able to simulate.
	2) Door(s) fail to close (VTOL)	F7*	Possible adverse roll, pitch or yaw moments after completion of conversion maneuver (15 possible combinations of events).	*Not able to simulate.
Thrust Spoiler Doors	One door fails to deploy or deployment angle greatly reduces.	*	Asymmetric thrust, resulting in uncontrollable yawing moment.	*Unable to simulate.
Thrust Vector Actuator	1a) Fan mode interlock switch failed open.	F10	Interrupts CTOL to VTOL conversion and "interlocks No-Go" annunciator lights (between β_v = 45° - 90°; no effect at β_v < 45°).	(Programmer switch A or C)
	1b) Failed short	*	No effect during normal two step conversion (because louvers are always opened to 45° before conversion is commanded).	*Not simulated. Note: Conversion could occur at β_v > 45° if commanded (i.e. auto convert).

TABLE 1 (Continued)

SYSTEM OR COMPONENT	FAILURE MODE	CODE	EFFECT OF FAILURE	REMARKS
Thrust Vector Actuator (Cont'd.)	2a) Conventional mode interlock switch failed short (7° to 45°).	*	a) Permits inadvertent VTOL to CTOL conversions at less than $\beta_v = 45°$.	(Programmer switch E or G) *Not simulated - no direct malfunction results from failure.
	2b) Failed open (45° to 90° β_v)	F11	b) Prevents conversion to CTOL (closed circuit is first condition required for conversion).	
	3a) Vector louver control "open" failed open (-7° to +45°β_v).	L1	Pilot unable to command reduced vector angle (sets minimum fan mode forward speed).	(Programmer switch M)
	3b) Failed short (45° to 90°)	*	Permits beep switch to open louvers in CTOL.	*Did not simulate.
	4a) Auto open switch failed short (-7° to + 45°β_v).	*	a) Permits louvers to devector to -7°, and at end of stroke actuator could burn up or blow out circuit breaker.	This failure would not cause flight problem as a single failure. (Programmer switch K) *Not simulated (simulator always in preconversion configuration.
	4b) Failed open (45° to 90°β_v)	*	b) Cannot put aircraft in preconversion configuration because louvers (vector actuator) cannot be moved to β_v - 45° position to close 45° interlock.	
	5a) Auto-Close switch failed open (-7° to +90°β_v).	*	a) Louvers not close after VTOL to CTOL conversion.	(Programmer switch J) *Not simulated - same reason as 4b and because not major problem in flight.
	5b) Failed short.	K7*	b) Permits louvers to vector to 90°, and at end of stroke actuator could burn up or blow circuit breaker.	(*K7 Check Switch or relay.
	6a) Vector louver control closed switch failed open (-7° to +45°β_v).	*	a) Pilot unable to command increased vector angle (sets maximum forward speed).	(Programmer switch L) *Not simulated.
	6b) Failed short (+45° to +90°β_v).	F12	Permits vector trim switch (stick grip) to drive louvers beyond 45°.	
	7a) Throttle cutback interlock switch failed short (-7° to +30°β_v).	E11	Throttle cutback could occur (not interlocked out) at low vector angles.	(Programmer switch R)
	7b) Failed open (+30° to +45°β_v).	*	Throttle cutback not available if required.	*Not simulated

TABLE 1 (Continued)

SYSTEM OR COMPONENT	FAILURE MODE	CODE	EFFECT OF FAILURE	REMARKS
Thrust Vector Actuator (Cont'd.)	7c) Failed short (45° to 90° β_v).	*	c) Throttle cutback possible in both CTOL and VTOL (not interlocked out).	*Not simulated
	8a) Trim control interlock switch failed short (-7° to +20° β_v).	*	a) Fan trim and automatic stick trim functions disabled. Stick grip trim switch operates aerodynamic trim functions.	*Not simulated
	8b) Failed open (20° to 45° β_v).	*	b) Stick grip trim functions disabled. Stick grip fan trims and automatic stick trim functions operate.	*Not simulated
	8c) Failed short (45° to 90° β_v).	*	c) Same as 8a, but only when mode select switch is in "FAN" position.	*Not simulated
	Increasing vector angle runaway (-7° to +45° β_v).	L5	Pilot may not be able to maintain altitude as vector angle increases.	
	Decreasing vector angle runaway (-7° to +45° β_v).	L6	No adverse effect expected, verify on simulator	
Flaps	Asymmetric Deployment	*	Adverse roll and yaw moments proportional to the magnitude of asymmetry (aerodynamic data indicates adverse roll moment greater than aileron roll moment).	*Unable to simulate
Controls	1) Complete loss of aileron boost power (CTOL only).	*	1) Roll power reduced to aerodynamic servo-tab capability only.	*Unable to simulate
	2) Forward louver torque tube "open".	K1	2) Reduced roll/yaw control power in fan mode.	
	3) Aft louver torque tube "open".	K2	3) Reduced roll/yaw control power in fan mode.	
	4) "Open" lift system	K3	4) No altitude control with lift stick (must depend on throttle control).	
	5) Main to pitch mixer interconnect "open".	K4	5) No pitch control at low fan mode speeds.	

9.1.3 Failure Simulating Controls

Failures were induced in three ways: by removing bolts from the simulator hardware, by manipulating the analog computer controls, and by installing control panels to simulate various electrical and hydraulic failures. Electrical failure control boxes are shown in Figures 48, 50, 53 and 54. Figure 48 shows the stabilizer control panel. This panel permitted simulation of all stabilizer directional and rate control valve, and rate sensing transducer failures. Figure 50 shows the control panel used to simulate switch failures in the thrust vector actuator programmer. Figures 53 and 54 show the two stability augmentation system failure control panels. Simulated louver servo-valve coil open and short circuits, and integrator cutout switch open and short circuits were introduced by these panels.

9.1.4 Flight Plans

Three basic flight plans were utilized for this study. They were designed to accomplish pilot and airplane exposure to random failures throughout the fan-powered flight regime; conversions in both directions and in the preconversion configuration, or to produce special conditions for certain specific failures.

9.1.4.1 General Flight Plan

A general flight plan was utilized for most of the failure study. This flight plan started with fan-powered flight at 300 feet altitude and 0 knots velocity. This was followed by a standard routine of climbing and descending 360° turns, vectoring and devectoring, and conversion in both directions. The flight plan was divided into 18 phases, and was flown continuously until a failure forced a change, or deviations were called for to further randomize the routine.

To further simulate an actual flight situation, the pilot was required to regularly state his flight condition, the aircraft position with respect to the terrain (i.e. visual display) and his intentions. The pilots were also instructed to indicate all suspected trouble when first detected, and then follow up with a description of his diagnosis and corrective action. Failures were introduced randomly and without notice to the pilot.

9.1.4.2 Single Engine Recovery Envelope

Before the actual simulated failure program was run on the simulator, both the fan mode single engine recovery procedure and recovery envelope were developed. This was accomplished by using the following flight plan. Each flight was initiated with altitude fixed at 1000 feet with single engine power, and the airplane trimmed for balanced flight at that specific speed point. Single engine failures could not be simulated for trimmed speeds less than $\beta_v = 10°$ due to computer data limitations.

9.1.4.3 Special Maneuvers

Two special maneuvers were flown at $\beta_v = 45°$ in order to deliberately approach fan stall and induce fan overspeed cutbacks. One maneuver was phugoid to simulate high speed, high angle of attack conditions. The other was maximum straight and level speed in fan mode.

9.1.5 Failure Recovery Criteria

Recovery criteria for this test program was established on the basis of airplane characteristics. An unsuccessful recovery (crash) was defined as ground contact with a sink rate greater than 10-feet per second and/or an unrecoverable stall. Stall was defined as occurring at 15° angle of attack. All other failure recoveries were considered successful. The 10-feet per second sink rate is the design limit load for the landing gear at a 9200 pound gross weight.

9.1.6 Fan Mode Single Engine Recovery Envelope and Recovery Procedure

The XV-5A is unable to sustain flight in the fan mode with only one engine operating. The objective of this part of the simulated failure program was to determine the boundary of velocity vs. altitude region. In this condition, if one engine should fail, the pilot would be unable to accomplish a safe landing within the limits as defined in Paragraph 9.1.5.

The speed range from 23.3 knots (10° β_v) through 73 knots (40°β_v) was investigated. A variety of flight paths and piloting techniques were tried at each speed point. Sink rate versus altitude data was plotted on an X-Y plotter for each test point. From this data, an optimized recovery procedure was developed and the minimum recovery envelope was derived.

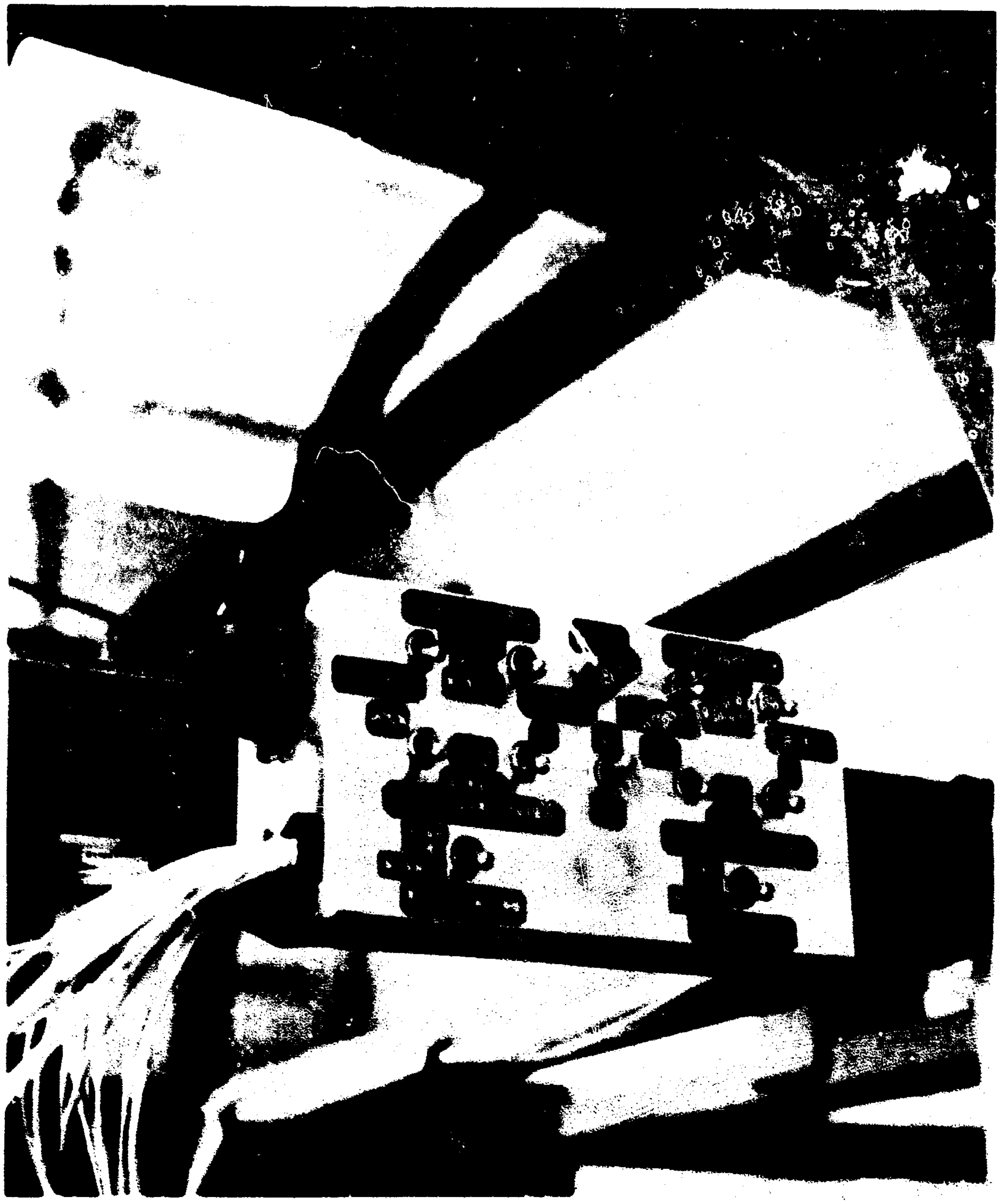

Figure 48 Stabilizer Control Panel

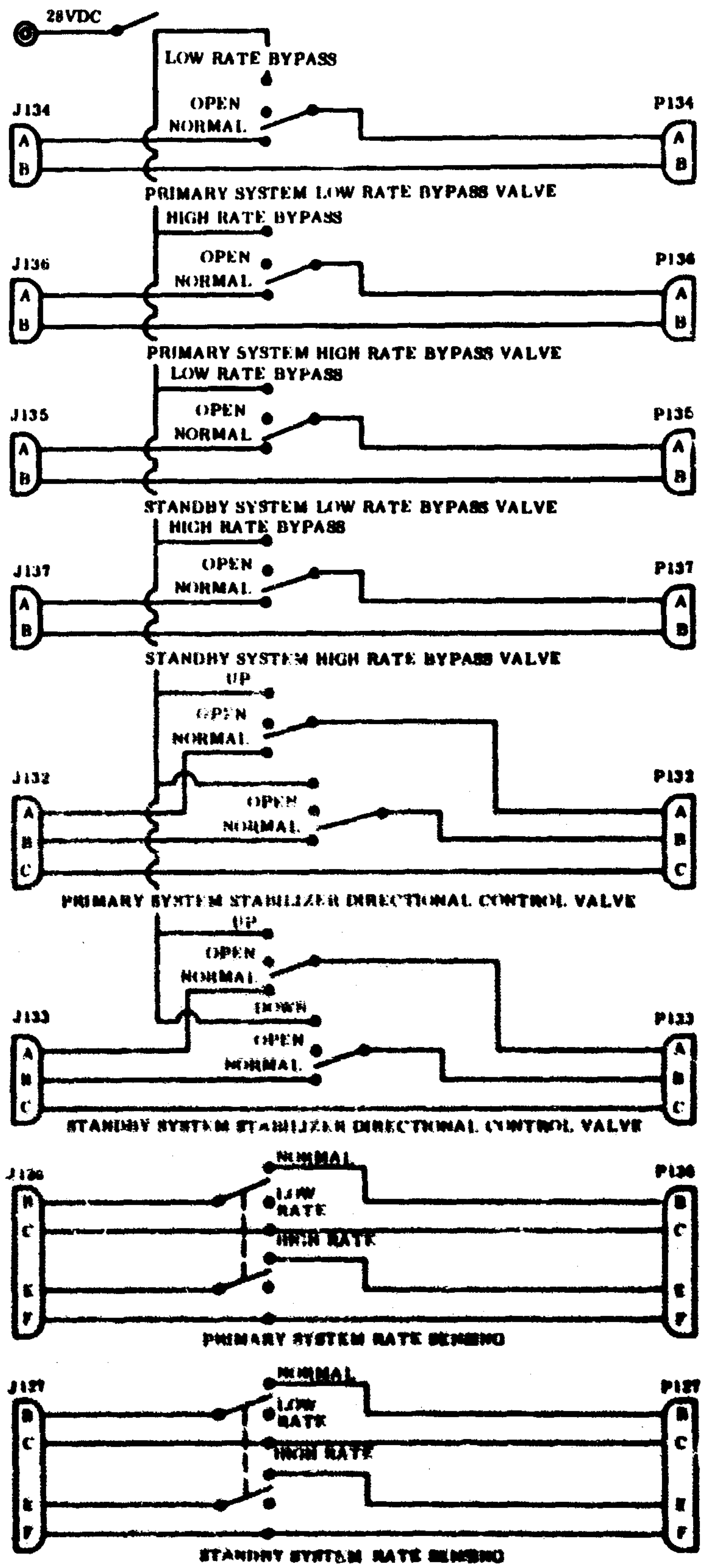

Figure 49 Standby System Rate Sensing

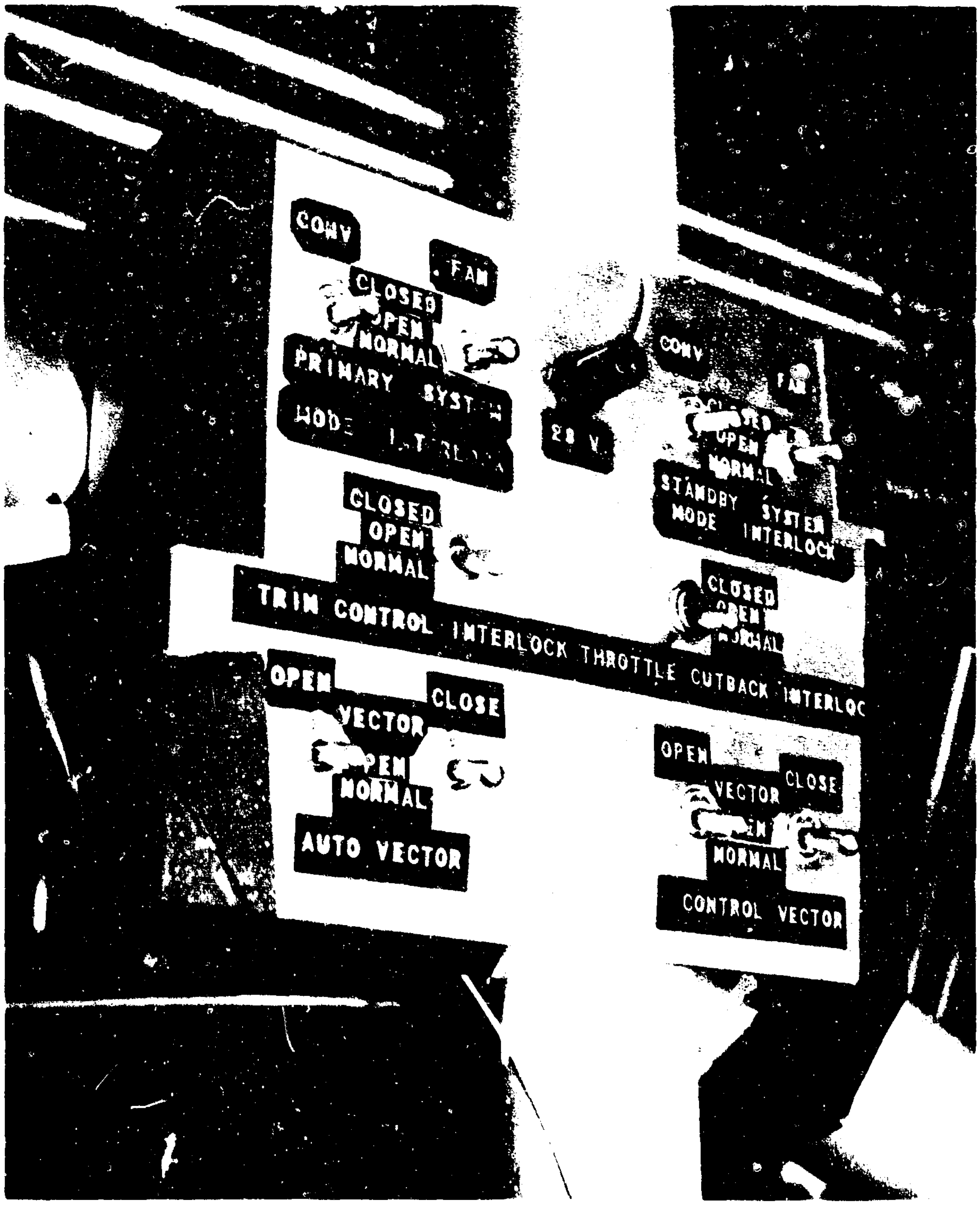

Figure 50 Thrust Vector Actuator Control Panel

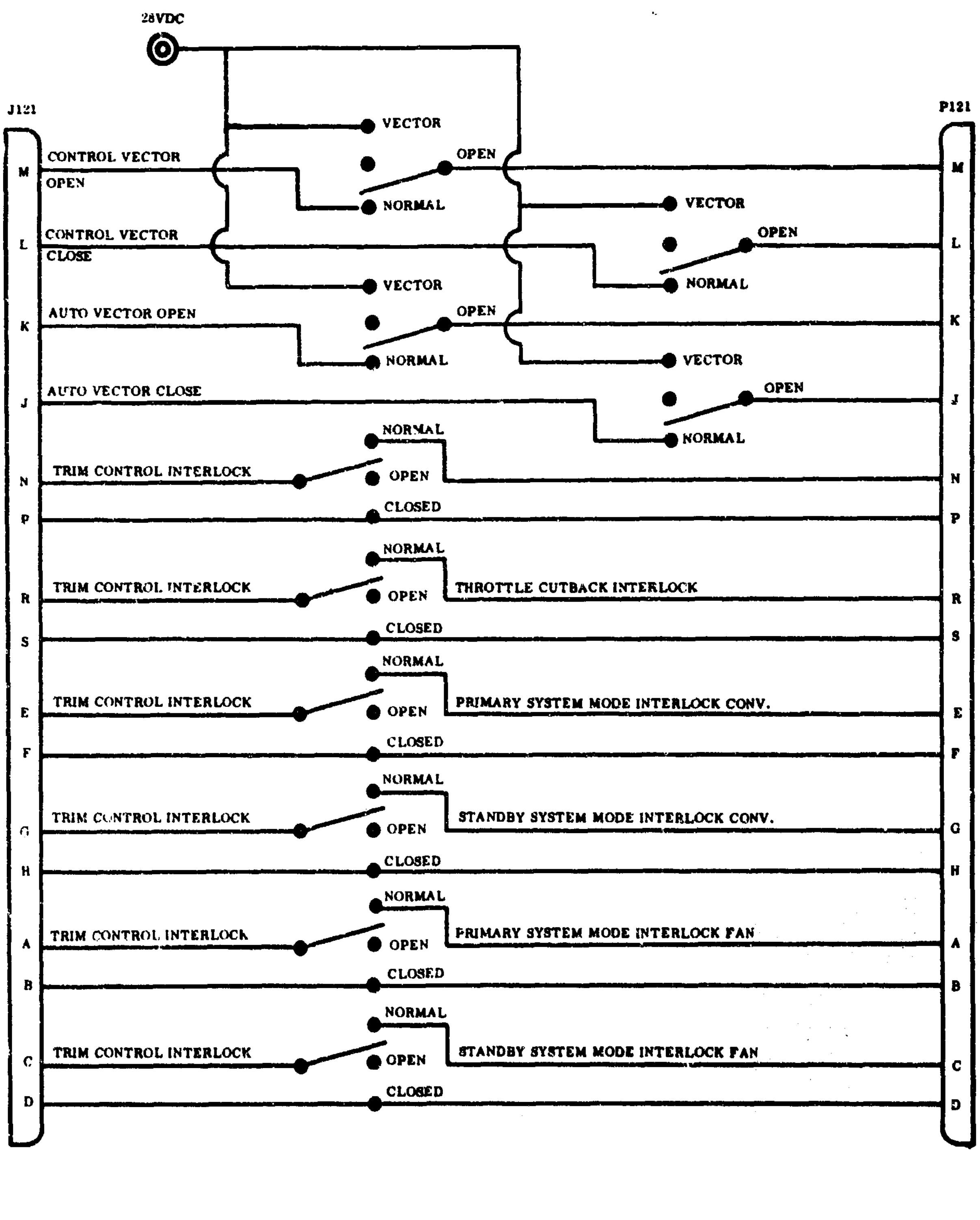

Figure 51 Thrust Vector Actuator Control Panel Schematic

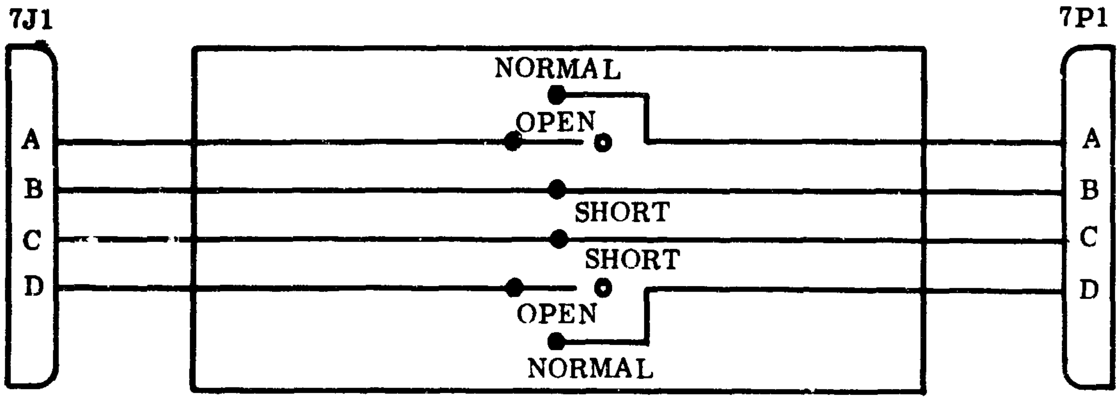

Figure 52 Louver Servo Valve Coil Failure Control Panel Schematic

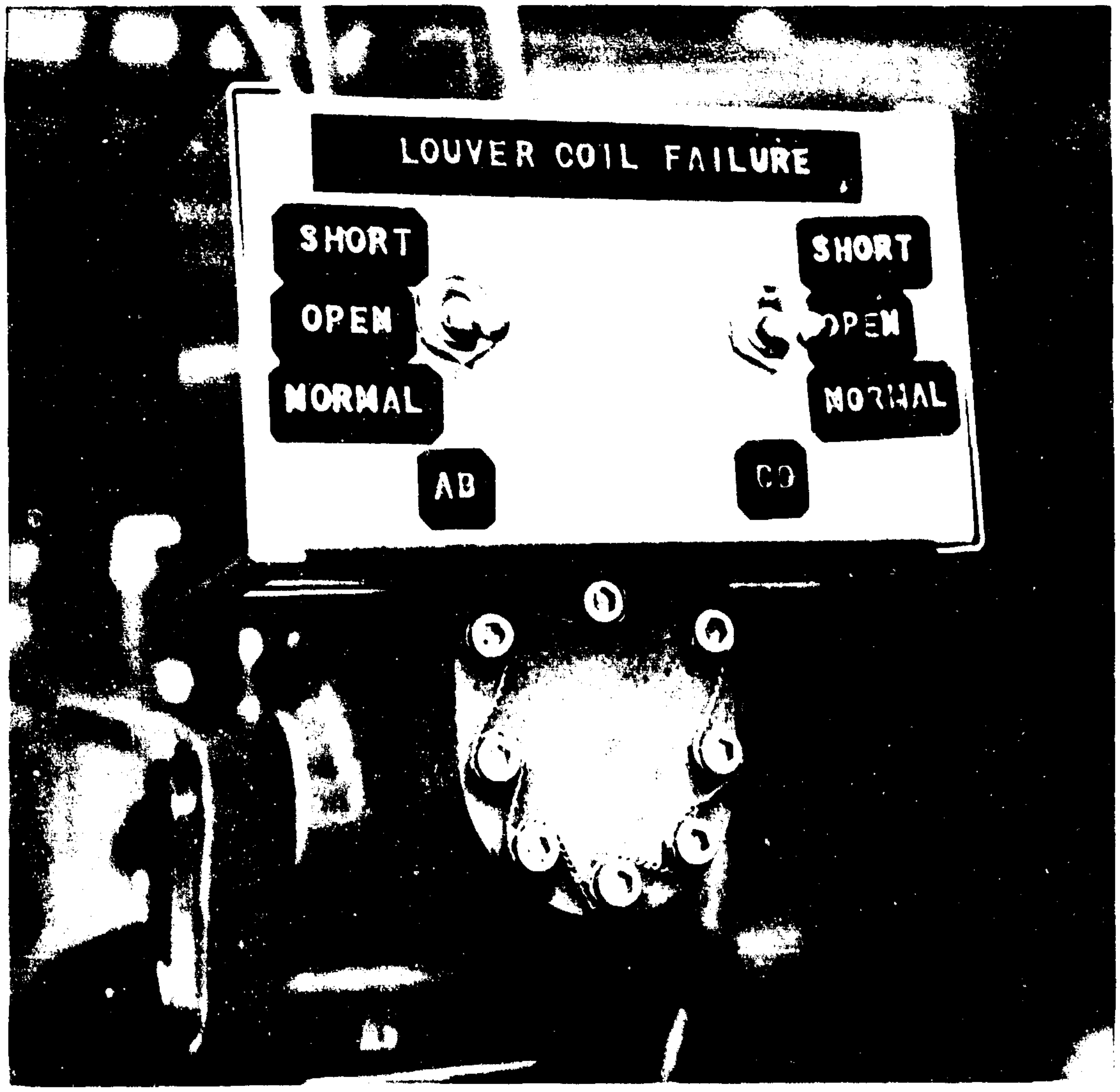

Figure 53 Louver Servo-Valve Coil Failure Control Panel

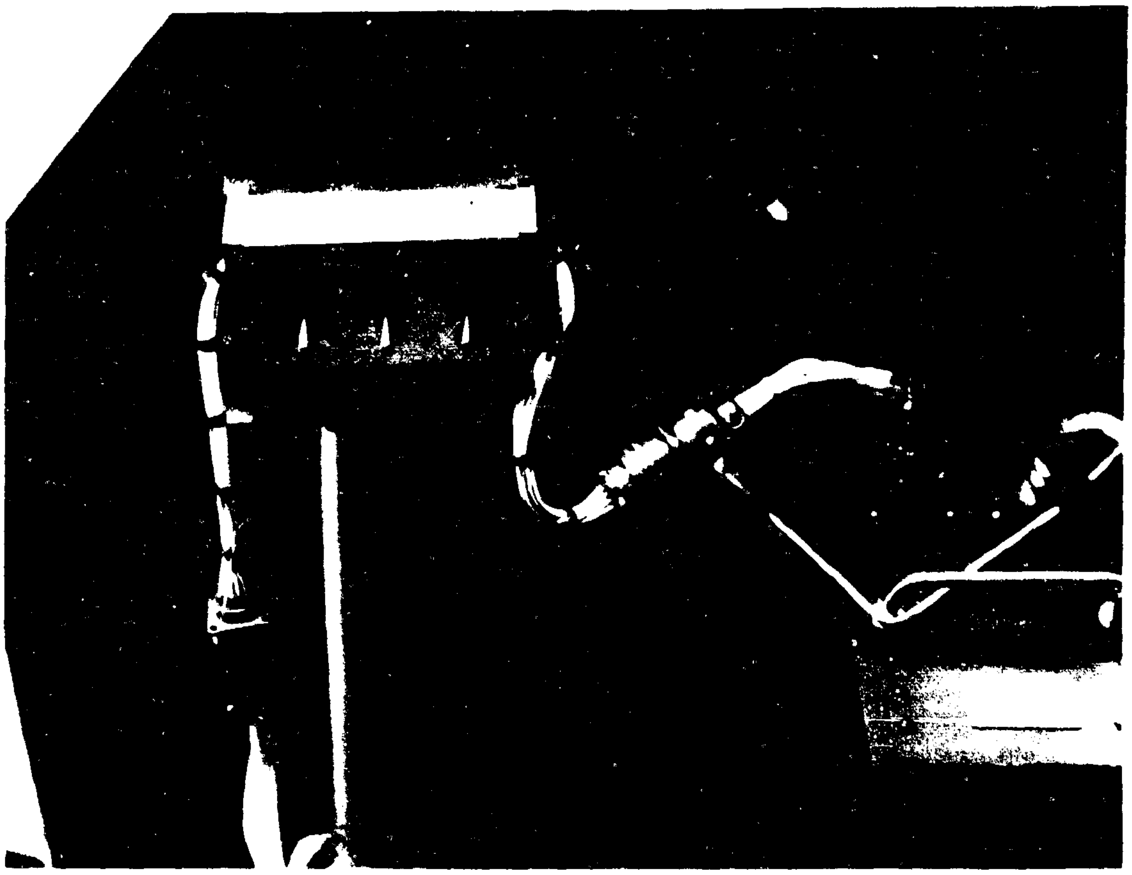

Figure 54 Integrator Cutout Switch Control Panel

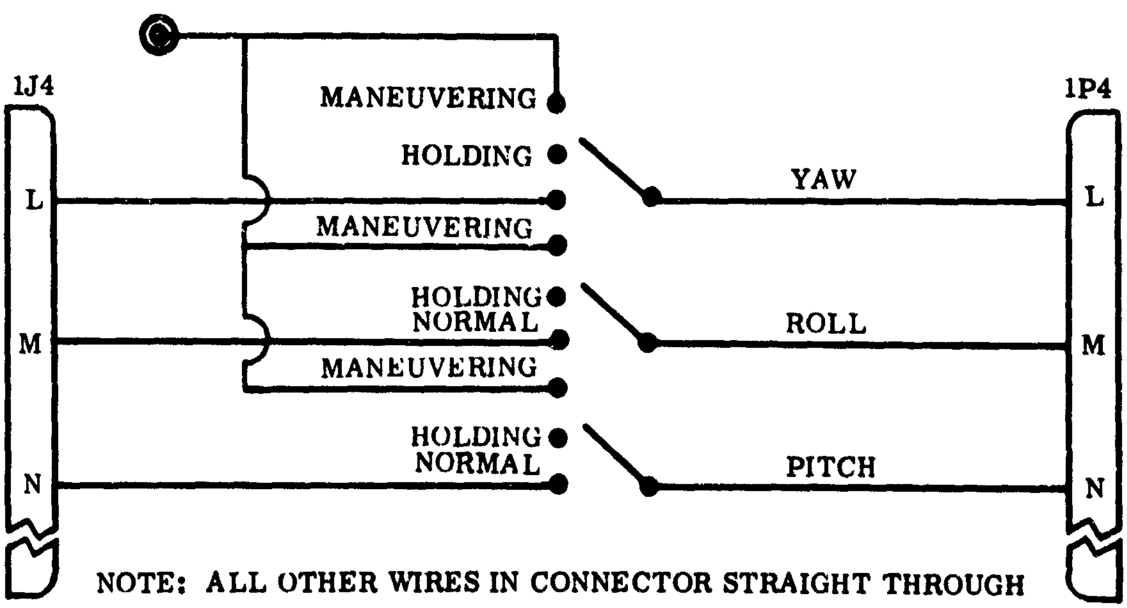

Figure 55 Integrator Cutout Switch Control Panel Schematic

9.1.6.1 Fan Mode Single Engine Recovery Procedure

A. Between 60 knots and conversion speed, use excess speed for altitude and sink rate control to achieve 60 knots minimum STOL landing.

   1. If conditions permit, hold 75 knot glide. This will permit holding zero fpm sink rate for four to five seconds following landing flare.

B. At 60 knots and less the standard single engine recovery is:

   1. Immediate pushover to accelerate aircraft (normal nose down tendency due to reduced power will help).

   2. Maintain approximately zero degrees α and accelerate to 60 knots.

   3. Roundout minimum altitude is 80 feet.

   4. Flare, to hold sink rate at 600 fpm or less at touchdown.

The recovery envelope is shown in Figure 56. Shown also is a discontinuous line marked "Hold 600 Per Minute." Several flights were completed in the flared condition starting from this initial velocity at failure. The aircraft was flown, in all cases, for altitude losses of 200 feet or more before stall occurred.

9.1.7 Simulator Test Cases and Other Data

Table 2 is a summary of the complete simulated failures program. Each failure, if successfully recovered or not, and the flight phase is indicated on this table. The eight categories for the 45 failure modes are also shown. Each category is briefly discussed. The significant failure modes, flight conditions, failure symptoms and recommended corrective action are indicated. Updating is also included and is based on material that has been derived from systems failure analysis conducted after completion of the simulated failures tests, and from the actual Phase I Flight Test Program.

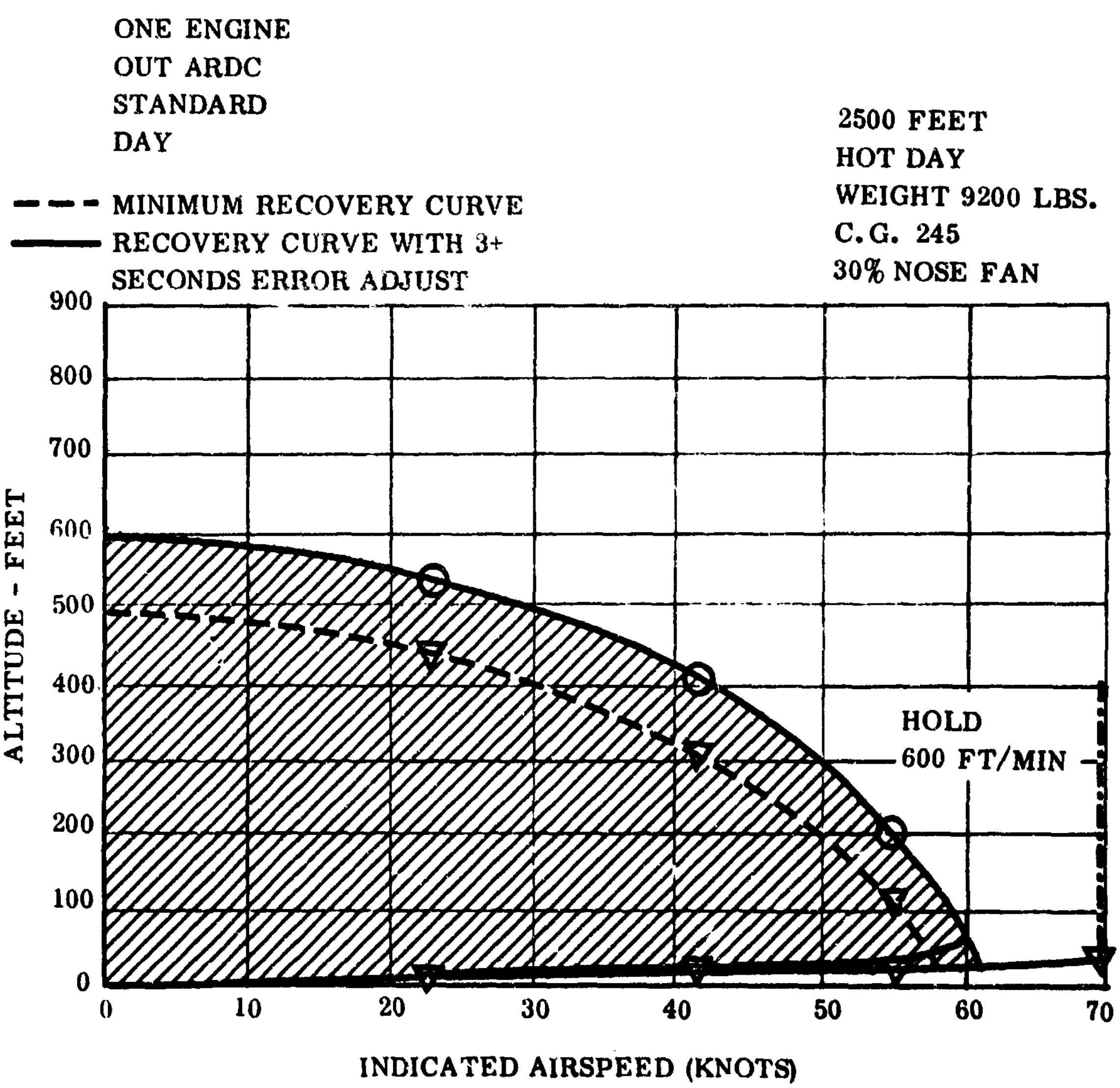

Figure 56 Single Engine Out Recovery (Simulator Results)

9.1.7.1 Stability Augmentation System

In general, no major problems could be detected with respect to stability augmentation system failures throughout the entire fan mode flight regime. This insensitivity to failures included inoperative single axis, inoperative multiple axes, and single axis hard-over signals. Hard-over signals result only in a biased displacement of the affected control, as if a trim change had taken place. This is accompanied by an apparent looseness in the affected control axis. Single axis inoperative failures appear in the same manner, but without the trim change effect. Most single component failures were practically undetectable

Airplane flying qualities and pilot workload during SAS failures was not well reported. Subsequent information indicates that aerodynamic stability looses its effectiveness below 40 knots.

9.1.7.1.1 Significant Failure Modes and Recommended Corrective Action

A. Primary Channel Dead (all 3 axes)

1. Flight Conditions

   a. VTOL, less than 40 knots. Note: Above 40 knots aerodynamic stability effective.

2. Failure Symptoms

   a. Stability looseness as velocity decreases below 40 knots, approaching uncontrollability as hovering is approached.

      (1) Roll problem below 40 knots

      (2) Pitch problem below 30 knots

      (3) Yaw relatively no problem, even at hover

3. Corrective Action

   a. Select STANDBY SAS (switch on control stick)

4. Flight termination for VTOL flight below 40 knots. CTOL flight plan may be resumed. Hover, STOL or conversion to CTOL for landing optional.

B. Single Axis Hard-over

1. Flight Conditions

a. VTOL, less than 40 knots (roll) or 30 knots (pitch) depending on axis affected.

2. Failure Symptoms

a. Trim change in axis affected accompanied by stability looseness in that axis as described above.

3. Corrective Action

a. Select STANDBY SAS

4. Flight termination for VTOL flight below 40 knots. CTOL flight plan may be resumed. However, STOL or conversion to CTOL for landing optional.

9.1.7.2 Horizontal Stabilizer Control

Stabilizer runaway trim as simulated by directional control valve hard up and hard down signals was the only significant "single failure" mode in this category. By the time the pilot could diagnose the nature of the problem, effective corrective action was quite unlikely. A combination audible warning (in the pilot's headset) and a visual warning (on the instrument panel) proved effective in reducing pilot recognition time, and improving corrective action. This warning system was subsequently incorporated in the airplanes. (See Paragraph 9.1.7.8 for other effects of stabilizer problems.)

9.1.7.2.1 Horizontal Stabilizer Failure Modes and Corrective Action Recommendations

A. Runaway (nose down trim)

1. Flight Conditions

a. CTOL mode

b. Preconversion mode

2. Failure Symptoms

   a. Stabilizer motion warning: light, sound and trim indicator moving in nose down direction.

   b. Increasing nose down trim (1/4° - 1/2° per second stabilizer travel).

3. Corrective Action

   a. Immediate corrective action required.

   b. Simultaneously select emergency stabilizer trim and nose up trim. Retrim on emergency trim as required.

4. Flight termination recommended.

5. Conventional landing may be made in either CTOL or pre-conversion configuration.

6. Conversion not possible with emergency trim selected. (Wing fan doors open and cycle stops.)

7. If conditions permit, STANDBY Conversion Control Interlock System may be selected to restore control stick pitch trim switch authority; proceed as follows:

   a. Select STANDBY (lift stick switch).

   b. Test STANDBY by selecting emergency trim off. If stabilizer still moves, select emergency trim and stay in the STANDBY configuration.

8. Conversion not recommended on STANDBY under conditions of single system capability.

9. Never return to PRIMARY after selecting STANDBY.

B. Runaway nose up trim

1. Flight Conditions

   a. VTOL mode

2. Failure Symptoms

   a. Stabilizer motion warning: light and sound and trim indicator moving in nose up direction.

   b. Increasing nose up trim (2.8° per second stabilizer travel).

3. Corrective Action

   a. Simultaneously select emergency stabilizer trim. Retrim on emergency as required.

   b. Immediate corrective action required above 30 knots. Note: As speed decreases, stabilizer becomes less effective so failure becomes less critical.

4. Flight termination recommended.

5. STOL or hover landing may be made.

6. Conversion not possible with emergency trim selected.

7. If conditions permit, STANDBY may be selected to restore control stick pitch trim switch authority; proceed as follows:

   a. Select STANDBY (lift stick switch)

   b. Test STANDBY by selecting emergency trim off. If stabilizer still moves, select emergency trim and stay in STANDBY configuration

8. Conversion not recommended on STANDBY under conditions of single system capability.

9. Never return to PRIMARY after selecting STANDBY.

9.1.7.3. Fan Overspeed Cutback

No problems were encountered as a result of overspeed cutbacks, including both false cutbacks and normal cutbacks. Cutbacks are regularly encountered during actual VTOL flight and have not displayed any dangerous characteristics.

9.1.7.3.1 Fan Overspeed Cutback Flight Conditions and Symptoms

A. CTOL to VTOL Conversion

1. Flight Conditions

   a. Preconversion configuration at 87 - 97% J-85 rpm and 105 knots.

   b. Increase J-85 rpm to 98 - 100% J-85 rpm just prior to VTOL mode select to achieve 100% fan rpm after conversion.

   c. If J-85 rpm power setting too high (102 - 103% rpm normal max. power), fan over-speed throttle cutback will occur after conversion.

2. Symptoms

   a. Power lever stiffens.

   b. Engine and fan rpm low.

   c. Throttle cutback to 70% power setting (99% rpm)

3. Corrective Action

   a. Retard throttle levers and Reset Power (pushbutton switch on lift stick).

4. Resume flight plan.

B. VTOL Mode

1. Flight Conditions

   a. Vectoring toward 45°.

2. Failure Symptoms

   a. Power lever stiffens.

   b. Engine and fan rpm low.

3. Corrective Action

   a. Retard throttle levers and Reset Power (pushbutton switch on lift stick)

4. Resume flight plan.

9.1.7.4 Gas Generators

Loss of power from one engine when the airplane is in the fan mode and inside the predicted recovery envelope is a critical failure. However, most failures outside the recovery envelope were recovered. Flaring too high and/or stalling was a common cause of nonrecovery.

The initial conditions for several engine failure simulations with the results are shown in Figure 57. Crashes are indicated by X. Recoveries (safe landings) are indicated by ▭

An oscillating fuel control failure resulted in significant pilot workload increase for both trials. No conclusions for appropriate corrective actions were reached.

9.1.7.4.1 Single Engine Failure Recovery Procedures

A. Success criteria sink rate < 600 fpm and $< 15° \alpha$.

B. In conventional or preconversion. Conventional landing as is.

C. During conversion either way. Fan or conventional recovery optional.

D. Between 60 knots and conversion speed. Use excess speed for altitude and sink rate control to achieve 60 knots minimum STOL landing.

E. If conditions permit, hold 75 knots glide. This will permit holding 0 fpm sink rate for 4 - 5 seconds following landing flare.

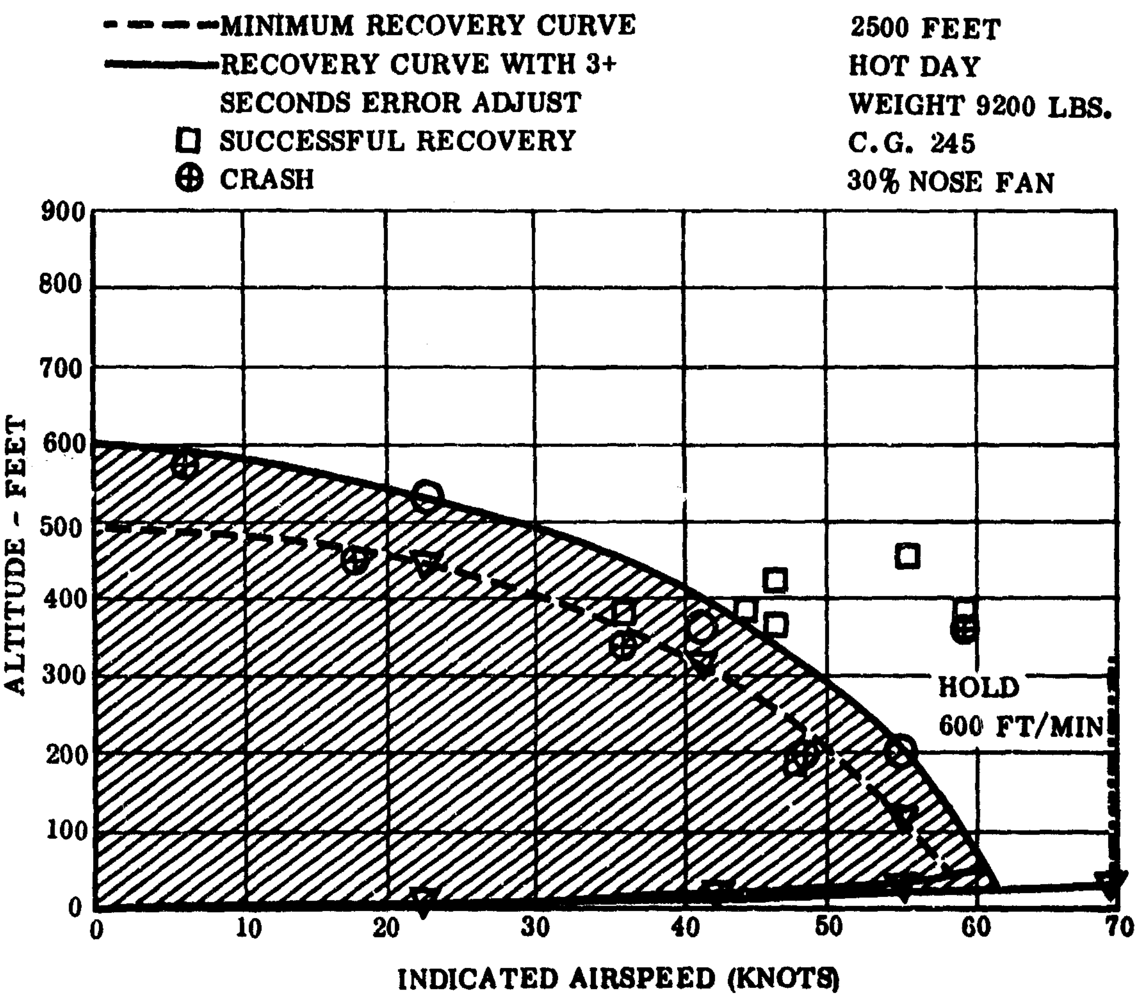

Figure 57 Single Engine Out Recovery (Simulator Results)

F. Below 60 knots Standard single engine fan recovery.

1. Immediate pushover to accelerate aircraft. (Normal nose down tendency due to reduced power will help.)

2. Maintain approximately 0° α and accelerate to 60 knots.

3. Roundout minimum altitude = 80 ft.

4. Flare to keep sink rate 600 fpm or less at touchdown.

9.1.7.5 Thrust Vector Actuator

Only one failure in the thrust vector actuator program resulted in a crash during the simulated failures study. This was a trim interlock switch short-circuit failure. The pilots had become accustomed to depending upon this switch to stop the vector actuator at 45° β_v. The failure was introduced without the pilot's knowledge (as were all failures during the study). As a result, he vectored beyond 45° inducing fan stall and overspeed cutback. Airplane stall and a crash followed. Subsequently, the pilots were careful not to exceed 45° vector angle and this failure caused no further problems.

This failure is felt to be representative of many of the failures that did not result in crashes because they all contributed to forcing the pilots to become totally aware of all the cockpit instruments and functions. All previous work in the simulator had been devoted to the development of particular handling qualities, or specific single items of interest. During this phase of the simulation, they could not allow their attention to become so focused, and therefore this work was good preparation for actual flight.

Vector actuator runaway did not cause any significant problems during this simulation, because it was not permitted to proceed beyond 45° vector angle on the presumption that this would require a double failure. When the pilot recognized the failure, which was artificially induced, it would be removed. This was necessary because the method chosen to simulate the failure prevented damage to the thrust vector actuator.

After a more complete failure analysis of the system, this assumption was proved false. A vector stop switch was added to the airplanes to allow the pilot to prevent a runaway vector from becoming catastrophic.

9.1.7.5.1 Thrust Vector Actuator Programmer Failure Modes and Corrective Actions

A. Trim Interlock Failed Short

1. Flight Conditions

a. VTOL flight approaching conversion to CTOL.

2. Failure Symptoms

a. Loss of lift

b. Fan overspeed

c. Vector angle greater than 45°.

3. Corrective Action

a. De-vector to 45°

b. Maintain 100% fan rpms

c. Convert to CTOL

4. Resume flight plan.

B. Vector Runaway

1. Flight Conditions

a. VTOL from hover to conversion

2. Failure Symptoms

a. Aircraft starts to accelerate

b. Vector angle increasing.

c. If vector angle greater than 50° fan overspeed and throttle cutback occur.

3. Corrective Action

   a. Immediately deactivate vector actuator with vector stop switch.

   b. If occurs during takeoff and altitude permits, land as is.

   c. For runaway in 0° to 45° direction, if vector runaway is stopped at low vector angle, at altitude, and fuel supply and VTOL landing conditions make CTOL landing imperative, use vector stop switch to advance vector angle to accomplish transition to conversion speed permissible.

   d. If vector angle is greater than 45°, convert as soon as possible.

4. CTOL or VTOL landing optional.

5. Flight termination recommended.

9.1.7.6 Diverter Valves

Diverter valve time delay relay failures did not cause any unrecoverable flight conditions. However, another possibility of a split mode configuration was discovered. The pilot made a conversion from fan to conventional, and then deliberately and quickly re-selected the fan mode. The stabilizer actuator position switch interlock circuitry was re-configured to prevent this problem from re-occurring, both on the airplanes and on the simulator. (See Paragraph 9.1.7.8 for the effects of diverter valve "NO GO" problems.)

9.1.7.7 Mechanical Mixer

Most of the failures in this category resulted in crashes. Those that did not probably would have resulted in crashes had they not been removed as soon as the pilot recognized the failure. These failures were introduced by slipping bolts out of place in the mechanical control system, and were replaced as soon as necessary to prevent damage to the simulator hardware. The usual reaction was loss of attitude control followed by stall and high sink rates. Attempted landings were unsuccessful. These failures should be classed with comparable failures in conventional mechanical control systems.

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d. Moderate to fast nose down pitch trim change requiring moderate to heavy aft stick force to maintain level flight.

e. Sink rate may develop.

f. Stall may occur.

3. Corrective Action

a. Immediately reselect CTOL, add full power, and be prepared to prevent stall.

4. Flight termination recommended.

5. Stay in PRIMARY recommended.

6. Standard CTOL landing recommended (select "CONV" louver switch position to lock wing fan door latches, close wing fan exit louvers, pitch fan thrust reverser doors and pitch fan inlet louvers). Preconversion configuration CTOL landing can be made.

7. Conversion not recommended on STANDBY under conditions of single system capability.

8. Never return to PRIMARY after selecting STANDBY.

C. CTOL to VTOL Split Mode (C Stab, V Diverter)

1. Flight Conditions

a. Conversion maneuver just after VTOL mode selection and wing fan doors open.

2. Failure Symptoms

a. Stabilizer motion (and motion warning) for 0.15 second <u>only</u> (normal is approximately 2 seconds).

b. Diverted light <u>ON</u> 0.4 second later.

c. Severe nose-up pitching moment (full fwd stick to control, pitching moment marginal).

d. Note: Conversion half completed - i.e., CTOL stabilizer and VTOL diverter.

3. Corrective Action

a. Simultaneously

(1) Immediately reselect CTOL mode and then STANDBY.

(2) Reduce power

(3) Apply full forward stick

b. After return to CTOL apply full power, and be prepared to prevent stall.

4. Flight termination recommended.

5. Standard CTOL landing recommended (select CONV louver switch position to lock wing fan door latches, close wing fan exit louvers, pitch fan thrust reverser doors and pitch fan inlet louvers). Preconversion configuration CTOL landing can be made.

6. Conversion not recommended on STANDBY under conditions of single system capability.

7. Never return to PRIMARY after selecting STANDBY.

D. VTOL to CTOL Wing Fan Door Diverter CTOL Interlock Failed Open.

1. Flight Conditions

a. VTOL to CTOL conversion completed (except wing fan doors do not close).

2. Failure Symptoms

a. Sink rate continues and stall buffet.

b. Nose down tendency (approx. 5° elevator required to trim out).

3. Corrective Action

   a. Return to VTOL or continue CTOL optional.

   b. If remain in CTOL

      (1) Add full power

      (2) Be prepared to prevent stall

      (3) When aircraft fully controlled select STANDBY to close wing fan doors.

4. Flight termination recommended.

5. Standby CTOL landing recommended (select CONV louver switch position to lock wing fan door latches, close wing fan exit louvers, pitch fan thrust reverser doors and pitch fan inlet louvers). Preconversion configuration CTOL landing can be made.

6. Conversion not recommended on STANDBY under conditions of single system capability unless fuel supply and VTOL landing conditions make CTOL landing imperative.

7. Never return to PRIMARY after selecting STANDBY.

8. If wing fan doors do not close after selecting STANDBY, aircraft may still be cleaned up to CTOL configuration by selecting LOUVERS CONVENTIONAL. Calculated stall speed with aircraft in normal landing configuration except wing fan doors open, 90 knots.

   If fault should clear (fault is open interlock circuit that spontaneously closes) doors will automatically close and lock if LOUVER switch is in CONV position and aircraft is in CTOL mode.

E. VTOL to CTOL Stabilizer No Go

1. Flight Conditions

   a. Conversion maneuver just after CTOL mode selection.

2. Failure Symptoms

a. No stabilizer motion warning and diverted light stays on.

b. No conversion - no change in aircraft.

3. Corrective Action

a. Check vector angle - if vector clears problem, resume mission.

b. If not, reselect VTOL mode.

4. Flight termination recommended.

5. Stay in PRIMARY recommended.

6. STOL or hover landing optional.

7. Conversion not recommended on STANDBY under conditions of single system capability unless fuel supply and VTOL landing conditions make CTOL landing imperative.

8. Never return to PRIMARY after selecting STANDBY.

F. VTOL to CTOL Diverter No Go

1. Flight Conditions

a. Conversion maneuver just after CTOL mode selection

2. Failure Symptoms

a. Stabilizer motion warning stops 0.2 second after mode select and diverted light stays on.

b. No conversion.

c. Nose up pitching moment (requires approx. 3° elevator to trim out).

3. Corrective Action

   a. Immediately reselect VTOL mode.

4. Flight termination recommended.

5. Stay in PRIMARY.

6. STOL or hover landing optional.

7. Conversion not recommended on STANDBY under conditions of single system capability unless fuel supply and VTOL landing conditions make CTOL landing imperative.

8. Never return to PRIMARY after selecting STANDBY.

G. VTOL to CTOL Split Mode (VTOL Stabilizer, CTOL Diverter)

1. Flight Conditions

   a. Conversion maneuver just after CTOL mode selection.

2. Failure Symptoms

   a. Stabilizer motion warning (and stabilizer motion) stop <u>DURING DIVERTER VALVE MOTION</u>.

   b. Severe nose down pitching moment (requires 30-40° elevator to trim out (25° elevator available) at 9° incidence).

3. Corrective Action

   a. Simultaneously

      (1) Full back stick

      (2) <u>Immediately reselect VTOL mode followed by STANDBY Conversion</u>

      (3) If no response: <u>EJECT</u>

4. Flight termination recommended.

5. STOL or hover landing optional.

6. Conversion not recommended on STANDBY under conditions of single system capability, unless fuel supply and VTOL landing conditions make VTOL landing imperative.

7. Never return to PRIMARY after selecting STANDBY.

9.1.8 Other Recommendations

As a result of the experience gained during the simulated failures program, additional control systems changes were recommended and incorporated. They include:

1. Eliminate the fan mode stabilizer trim function from longitudinal stick position. At vector angles less than 40° position, the stabilizer in the full nose down (airplane) trim position. At vector angles greater than 40°, the stabilizer trim function to be controllable from the longitudinal stick grip pitch trim switch.

2. Extend the range of the longitudinal fan-powered trim so that the aircraft may be trimmed hands-off at any region in the fan-powered regime.

3. Change the lateral stick displacement to ±4 inches from ±5 inches maximum travel.

4. Change fan mode roll and pitch stick force gradient to approximately 1-1/3 pounds per inch.

5. Change SAS hold-maneuver switch band so that ±3/4 inch of stick displacement about center will actuate switches.

6. Modify yaw and pitch SAS channels in the primary mode so that position feedback in the holding configuration is eliminated. (To obtain a system with two rate gains, depending on control input, instead of a system with a position gain in the holding configuration and a rate gain in the maneuvering configuration.)

TABLE 2
FAILURE/PILOT RESPONSE SUMMARY

	TEST CODE	SIMULATED FAILURE / FLIGHT PHASE	$i_v = 0°$	360° CLIMB	360° DESCEND	ACCEL. TO $i_v = 15°$	360° CLIMB	360° DESCEND	DECEL. TO $i_v = 0°$	ACCEL. TO $i_v = 30°$	360° CLIMB	360° DESCEND	DECEL. TO $i_v = 0°$	ACCEL. TO $i_v = 45°$	360° CLIMB	360° DESCEND	CONVERT F → C	CTOL	CONVERT C → F	DECEL. TO $i_v = 0°$
STAB. AUG. SYS.	A1	SAS Roll Hard Over	O, O																	
	A2	SAS Yaw Hard Over		O, O																
	A3	SAS Pitch Hard Over		O, O																
	B1	SAS Roll Axis Dead		O, O																
	B2	SAS Yaw Axis Dead	O, O																	
	B3	SAS Pitch Axis Dead	O, O, O																	
	C1	One R/T Coil Open	O, O																	
	C2	One R/T Coil Short				OA														
	C3	Two R/T Coils Open				OA														
	C4	Two R/T Coils Short	O, O																	
	C5	One R/T Coil Open One Short	O, O																	
	D1	All SAS SW to Maneuver	OB	OB	OB															
	D2	All SAS SW to Hold	OB	OB	OB															
	B4	All SAS Axes Dead	OA			O			OE	O, O	DA	DA								
MIN. STAB. CONT.	E1	B/S #1 Contr. Vlv. Hard Down	O, O			O			OE	O, O, XO, DE, C	O, O	O, X		O, O, O, O, O						
	E2	B/S #2 Contr. Vlv. Hard Up												OC, O				O, X, X, O, O		
	E4	B/S #2 LS Bypass Open									O									
	E13	B/S #1 Contr. Vlv. Up Coil Open																O	(X) OB	
	E14	B/S #1 Contr. Vlv. Down Coil Open															O, OB	O	O	
	E15	B/S #1 Pt. Short + MRBP Closed															OB		OB	
	E16	B/S #1 Flow Trans. Open															O, (X) O		O, O	
	E17	B/S #1 Flow Trans. Short															O		OB, O	
	E19	B/S #1 & 2 Pt. Short - LRBP Open	All B/S Rate 3-1.2° Per Second														X, X, OA, O			
	E20	B/S #1 & 2 Pt. Short + Both BP Closed	All B/S Rate 1.6° Per Second														X, O		X, O	
	E22	B/S #1 Both Bypass Valves Closed	All B/S Rate 2.4° Per Second															O		
FAN O/S	E10	False Fan O/S Signal							O											
	E11	False Fan O/S Cutback	O, OE		O			O		O, O		O								
	F3	Normal Fan O/S Cutback	(Special Maneuver-1 Pass at 0 i_v =45° O O				-2 Max Speed @ i_v=45°		O, O, O											
GAS GEN.	E12	Single Engine Failure		XA		X, O			X	O, O, O, X, O, O		OE, (X), O		X, X, O, X, O, O		O	O, O, O, O, O, O	(C) X, O, O, O, (X) (X) (X) (X) O	O, O, O	
	F1	Fuel Control Oscillation												OC, OD						
VECTOR ACTUATOR	F10	Fan Mode Interlk. Failed Open																	O	
	F11	Conv. Mode Interlk. Failed Open															O, O			
	F12	Vctr. Lvr. Cmy. Close F Short	(Bumpy Acceleration from i_v = 0 to i_v = 45°, see Data Sheets											X, O, O						
	E7	Vctr. Actu. Close Relay Short				O				O, O										
	L5	Vector Runaway → 45°	O			O, DA				O, O, O										
	L6	Vector Runaway → 0°	O							O										
	L1	Vector L/C Open SW Fail Open				O														
DIVERTER VALVE STR.	F5	Dvtr. Vlv. TD Relay Short															O		O, O, O	
	F6	Dvtr. Vlv. TD Relay Open																	O, O	
MECHAN. MIXER	K1	Ped. Lvr. Torque Tube Out												O, X						
	K2	Aft Lvr. Torque Tube Out								X				O						
	K3	Open Lift System				X				X, X										
	K4	Main to Pitch Mxr. Intrcon. Open				O				X, O										
INSTR.	G1	Air Speed Indicator Out											DE							
	G2	Vctr. Angle Indicator Out												O						

A = Faulty Pilot Diagnosis
B = Pilot Did Not Detect Failure
C = Small Pilot Workload Increase
D = Large Pilot Workload Increase

E = Pilot Response Slow
O = Recovered Aircraft
X = Lost Aircraft
(X) = Circle Indicates Questionable Data

9.2 FIELD FAILURE SUMMARY

The basic information regarding failure was obtained from the Ryan Form R-2073 Ryan Equipment Failure Report completed and submitted by personnel with the airplane at EAFB. A sample form is presented in Figure 58. Instructions for completing this form are contained in Ryan Aeronautical Company Quality Bulletin XV10.

For the purpose of this summary, the airplane was divided into the following subsystems:

1. Airframe - Fuselage
2. Airframe - Wing
3. Airframe - Empennage
4. Controls
5. Electrical
6. Hydraulic
7. Cockpit
8. Landing Gear
9. Propulsion - Power Plant
10. Propulsion - Fuel
11. Propulsion - Miscellaneous
12. Parachute

The data recorded on the Failure Report for time in service showed no correlation to operating times as recorded in the Airframe and Power Plant Running Time Log and Electroplating Clocks installed on the electrical system inverters. The estimates of hangar operating times were made as follows:

1. Control and Hydraulic Systems and the Cockpit - Two (2) hours per shift, six (6) days per week.

2. Electrical System - Four (4) hours per shift, six (6) days per week.

For the airframe, landing gear, parachute and propulsion systems, it was assumed that design loads appeared on these systems only during flight or ground engine running. The operating time for these systems is only that which appears in the Airframe and Power Plant Running Time Log. The hangar operating time was calculated for each aircraft with the start date taken as the date at which the airplane was in flight condition after arrival at EAFB. For Aircraft No. 2 this date was 5 March 1964, and for Aircraft No. 1, 12 October 1964. To these figures was added the time recorded in the Power Plant Log. The data are tabulated with respect to the mode of aircraft operation during which the particular failure occurred (i.e. conventional flight, taxi tests, etc.)

Individual system average failure rate tabulations are presented in Table 3 through 14. Rate data by monthly period plus cumulative totals are shown from March 1964 through January 1965. For these system tabulations, the total number of failures accumulated by both aircraft for the period for a particular operating mode was divided by the total time accumulated by both aircraft for the same period and mode. Both the system average failure rate for each period, and the cumulative average failure rate, are obtained by adding the individual mode failure rates for that period.

Cumulative failure rates for each system and for the total aircraft by period are shown in Table 15. The total aircraft failure rate is obtained by adding the individual system failure rates. Individual system cumulative failure rates are shown graphically in Figures 59 through 65. Total aircraft rate is shown in Figure 66. For these latter tables and corresponding plots, the three propulsion systems have been grouped together as have the three airframe subsystems.

The plot of the complete airplane data indicate a stabilization of the failure rate after the original infant mortality had been overcome. The plot of the complete airplane follows closely the plot of the propulsion system which comprised approximately 40% of the total failures. The other plots also indicate a stabilization of their individual failure rates, with the exception of those which exhibit extremely low rates.

Insufficient data are available to permit drawing statistically valid inferences or conclusions regarding failure rates for some systems. The systems with low rates mentioned above exemplify this, since single failures cause wide variations in the failure rate.

Time between failure data was tabulated and plotted for those systems which had the largest number of failures. These are Propulsion-Miscellaneous, Propulsion-Total and Electrical, Figures 67, 68, and 69, respectively. For this data, the time recorded was to the end of the day on which the failure occurred. In the case of multiple failures, reported on the same day, each was treated as if it occurred alone. This was done to be more representative of time between failure. Noted on these figures are the Median and Arithmetic Mean values of time between failure. Also noted is the Average value which is obtained by taking the reciprocal of the failure rate obtained from the previous tables. The difference between these values is felt to be due to the relative small data sample from which the information is derived. This difference would indicate that these figures are not the final absolute value, but that they are converging upon the final value. This difference does not alter the fact that the overall aircraft failure rate has stabilized after the fourth reporting period.

RYAN EQUIPMENT FAILURE REPORT

* Answer Req'd: Yes ___ No ___

1. REPORTED BY ____________
2. REPORT NO. ___ * Date ___
3. REFR. NO. ____________ 1-6

LEAVE BLANK

* Est. Time to Repair ____________ Man Hours

____________ Clock Hours

4. ACTIVITY 6
5. FAILURE DATE 7-12
6. SYS. NAME 13-17
9. SYS. SER. NO. 38-41
12. SUBSYS. NAME 48

FAILED ASSEMBLY/INSTALLATION

7. NAME 18-25
10. SER. NO. 44-54
13. MFG. NAME 69-74
15. PART NO. 6-22

FAILED PART

8. NAME 26-38
11. SER. NO. 55-65
14. MFG. NAME 75-80
16. PART NO. 23-39
17. REF. DESIG. 40-49

NOTE

X ONLY THE ONE MOST PERTINENT IN EACH SECTION.

18. WHERE WAS FIRST SYMPTOM NOTICED 51

1 [ ] SYSTEM
2 [ ] SUB-SYSTEM
3 [ ] ASSEMBLY
4 [ ] PART

19. WHAT WAS FIRST SYMPTOM 52

1 [ ] INOPERATIVE
2 [ ] INTERMITTENT
3 [ ] IMPROPER REACTION TO CMND.
4 [ ] INTERFERED WITH
5 [ ] LEAKING
6 [ ] LOST CARRIER
7 [ ] OUT OF ADJUSTMENT
8 [ ] OUT OF TOLERANCE
9 [ ] OVERHEATING
A [ ] ROTATING OUT OF LIMITS
B [ ] SUB NORMAL PERFORMANCE
C [ ] UNSTABLE
D [ ] UNCOMMANDED ACTION
E [ ] VIBRATING EXCESSIVELY
F [ ] OTHER (SEE REMARKS)

20. WHEN SYMPTOM NOTICED 53-54

1 [ ] STORAGE INSPECTION
2 [ ] UNCRATING
3 [ ] BEFORE INSTALLING PART
4 [ ] GROUND OPERATION OR TEST
5 [ ] COMBINED SYSTEMS CHECK
6 [ ] PRE-FLITE CHECKS
7 [ ] PRE-LAUNCH CHECKS
8 [ ] IN FLITE - FLT NO ___ 54
9 [ ] POST FLITE INSPECTION
A [ ] SCHEDULED MAINTENANCE
B [ ] TROUBLE SHOOTING
C [ ] FINAL ACCEPTANCE CHECKOUT
D [ ] RECEIVING INSPECTION

E ___ Conv. G ___ Taxi

F ___ Fan H ___ Ground c/o

21. CAUSE OF FAILURE 59 60

[ ] CERTAIN [ ] SUSPECT 59

1 [ ] DESIGN DEFICIENCY 60
2 [ ] FAULTY MFG./INSP.
3 [ ] FAULTY PRESER./PKG.
4 [ ] DAMAGE IN SHIPMENT
5 [ ] IMPROPER STORAGE
6 [ ] IMPROPER HANDLING
7 [ ] OPERATING TECH. ADJUST.
8 [ ] NORMAL USE
9 [ ] IMPROPER MAINTENANCE
A [ ] CONTAMINATION
B [ ] FOREIGN OBJECT/HIT
C [ ] OVERHEATING
D [ ] WEATHER
E [ ] SUBCOMPONENT FAILURE (SEE REMARKS)
F [ ] OTHER PART FAILURE
G [ ] NOT DETERMND. (SEE REMARKS)

22. FAILED ITEM CONDITION 61

1 [ ] ASSEMBLED IMPROPERLY
2 [ ] ARCING
3 [ ] BROKEN
4 [ ] CORRODED
5 [ ] DISTORTED
6 [ ] ERECTED IMPROPERLY
7 [ ] GASSY
8 [ ] LOW GM/EMISSION
9 [ ] LEAKS
A [ ] NOISY
B [ ] OPEN
C [ ] OUT OF ADJUSTMENT
D [ ] RESTRICTED
E [ ] SCORED
F [ ] SHORTED
G [ ] TESTS O.K. DIDN'T WORK
H [ ] TORN
I [ ] VALUE CHANGED
J [ ] WORN EXCESSIVELY
K [ ] CANNOT BE DETERMND. (SEE REMARKS)

23. RESULT OF FAILURE 63

1 [ ] FLITE DELAY
2 [ ] FLITE ABORT
3 [ ] ABNORMAL FLITE CHRCTRSTCS.
4 [ ] INACCURATE DATA DISPLAY
5 [ ] DID NOT AFFECT OTHER SYS. OR COMPONENTS
6 [ ] AFFECTED OTHER SYS. OR COMPONENTS
7 [ ] FLITE TERMINATION
8 [ ] VEHICLE EXPENDED

24. TROUBLE CAUSED ACCIDENT 64

[ ] YES [ ] NO

25. HISTORY OF ITEM 64

1 [ ] ORIGINAL INSTALLATION
2 [ ] INSTALLED AS REPLACEMENT

* Est. Ground Time on Part ___ Hrs.

26. 66-70

1 [ ] NEW
2 [ ] OVERHAULED/REPAIRED

USED ON ___ PREVIOUS FLITES 66

___ HOURS IN SERVICE 68

27. DISPOSITION OF ITEM 71

1 [ ] SCRAPPED
2 [ ] REPAIRED REINSTALLED
3 [ ] REPAIRED RETURNED TO STOCK
4 [ ] TAGGED FOR REPAIR
5 [ ] HELD FOR EXHIBIT
6 [ ] REFERRED TO SALVAGE
7 [ ] REFERRED TO M.R.B.
8 [ ] OTHER (SEE REMARKS)

28. INSPECTION ONLY 72-77

CAUSED PRODUCTION DELAY

[ ] YES [ ] NO 72

QUANTITY ACCEPTED ___ 73

QUANTITY REJECTED ___ 76

* Remarks and Description of Deficiency:

Figure 58 Ryan Equipment Failure Report

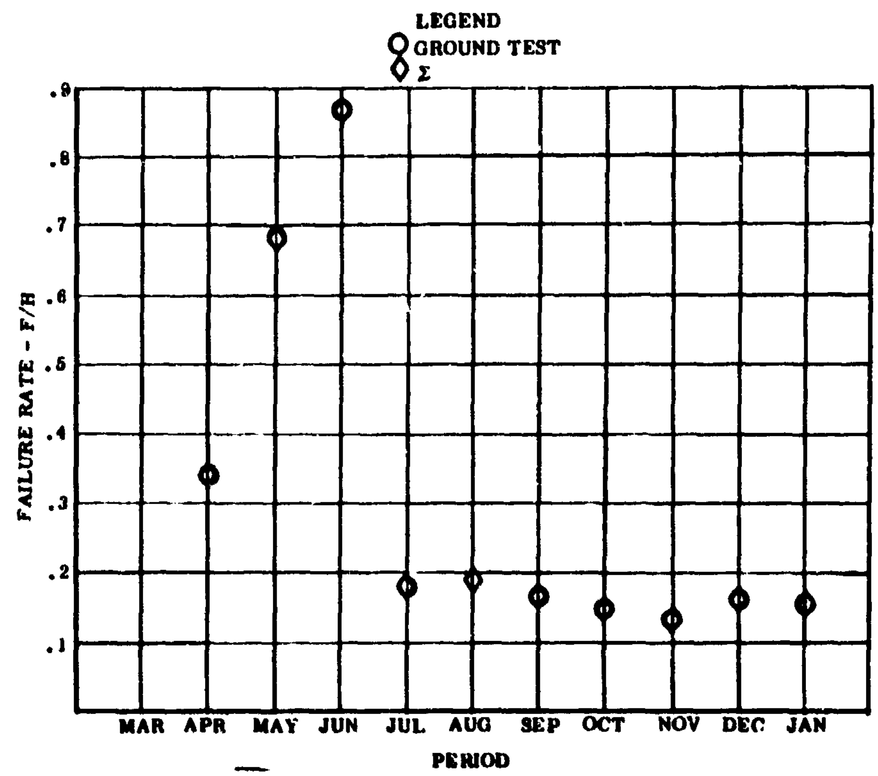

Figure 59 Airframe System

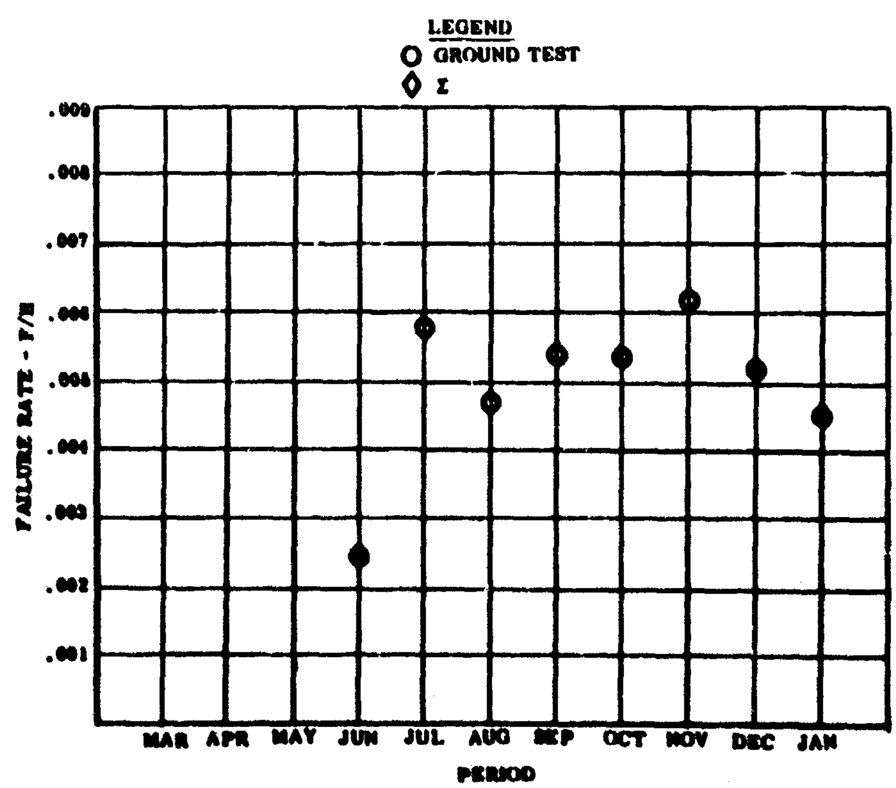

Figure 60 Controls System

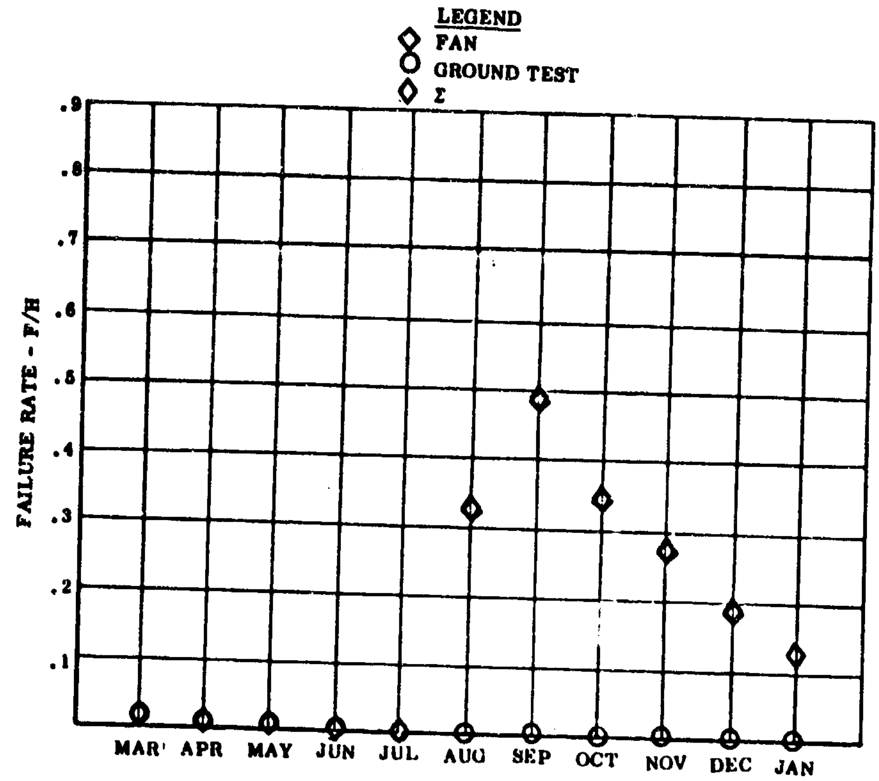

Figure 61 Electrical System

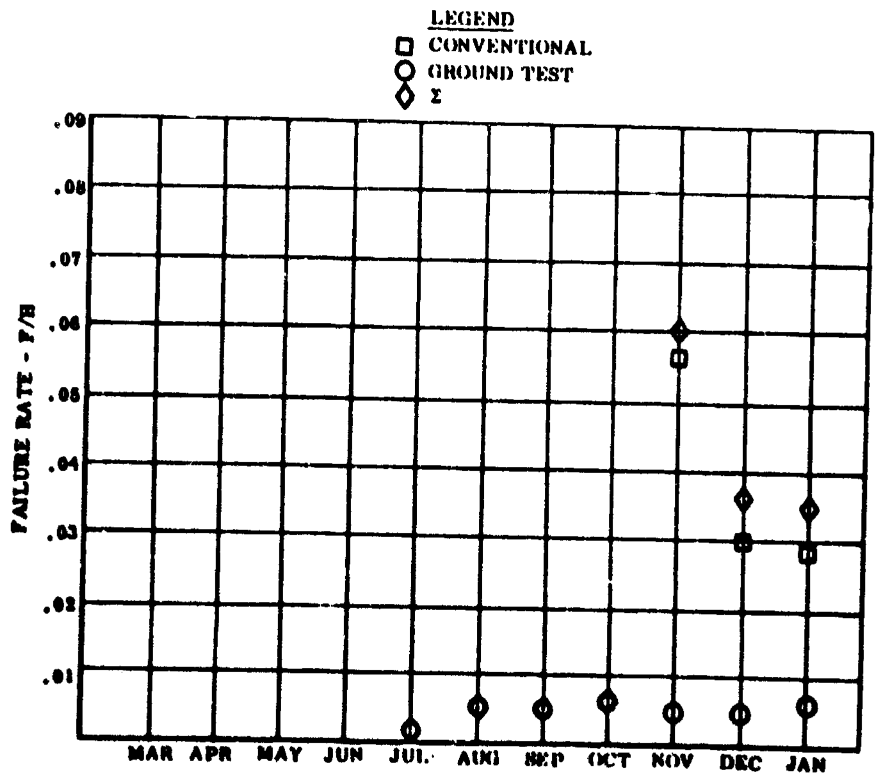

Figure 62 Hydraulic System

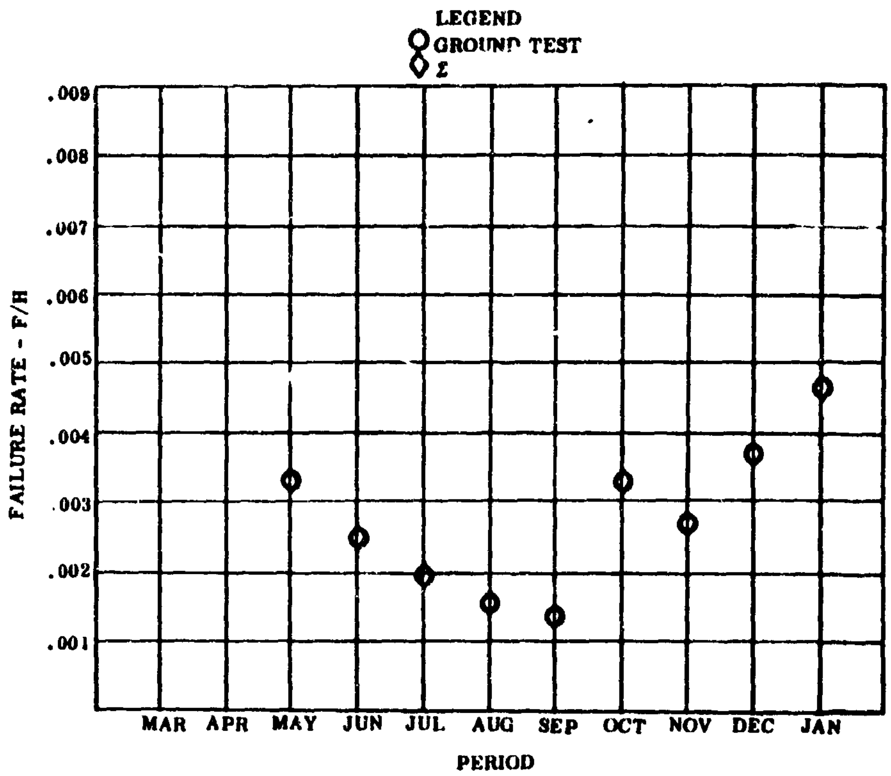

Figure 63 Cockpit System

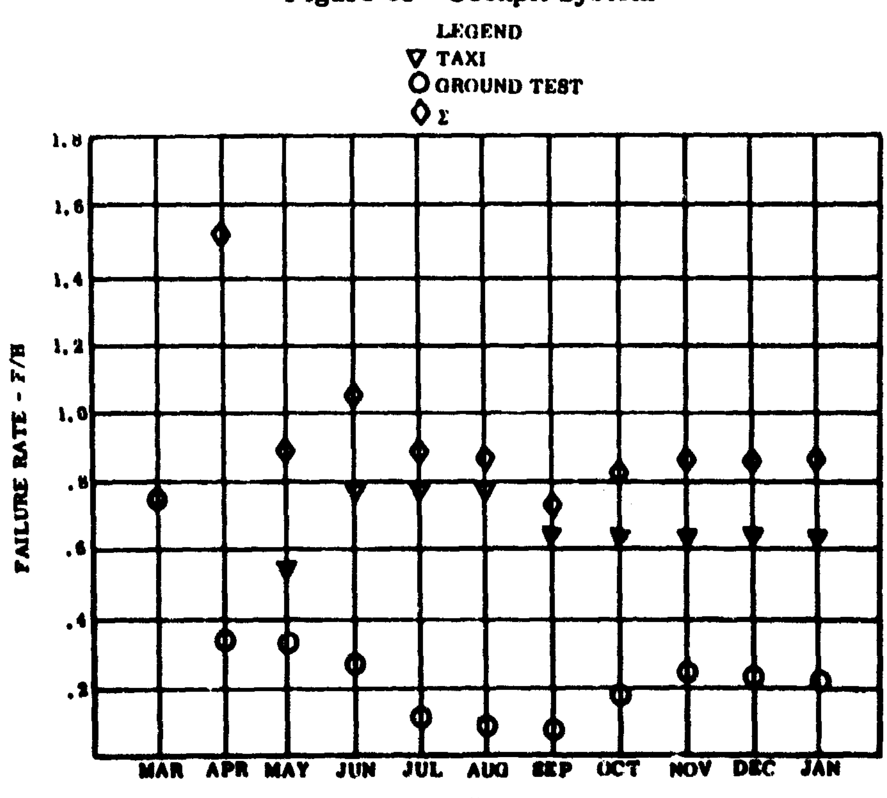

Figure 64 Landing Gear System

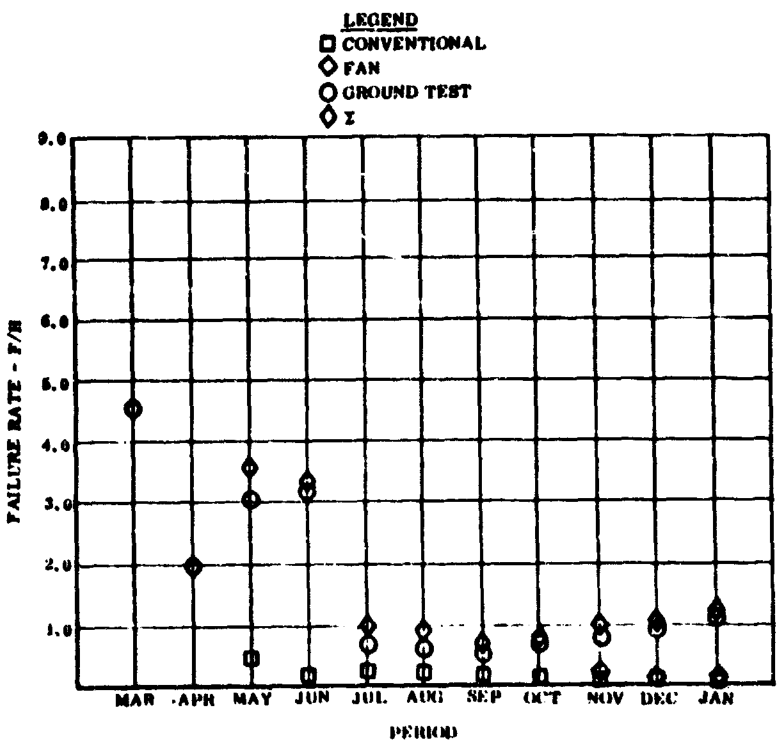

Figure 65 Propulsion System

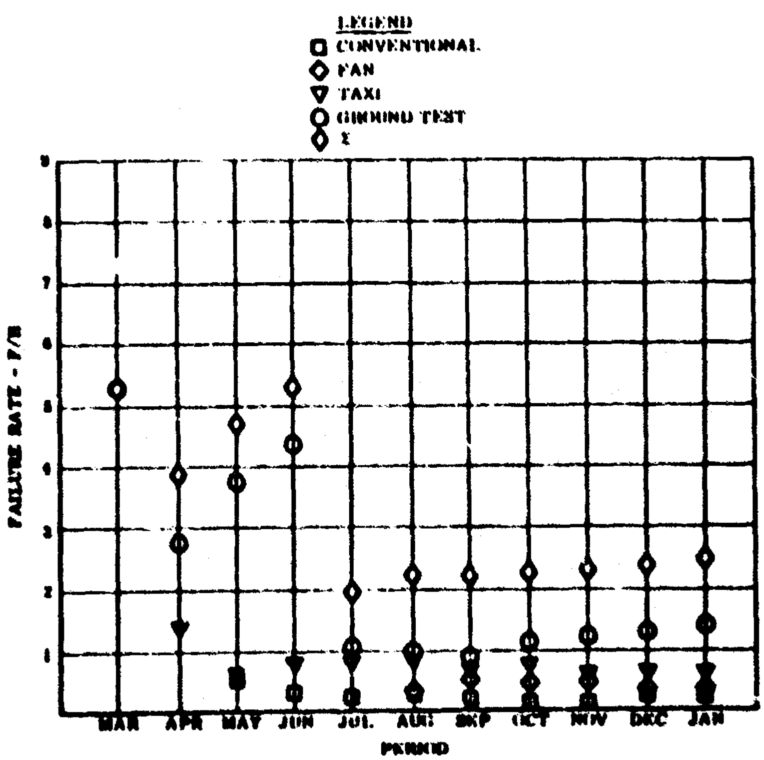

Figure 66 Airplane Failure

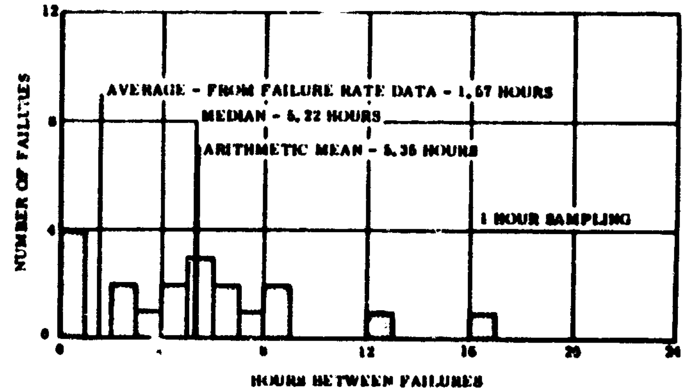

Figure 67 Time Between Failure Propulsion System, Miscellaneous

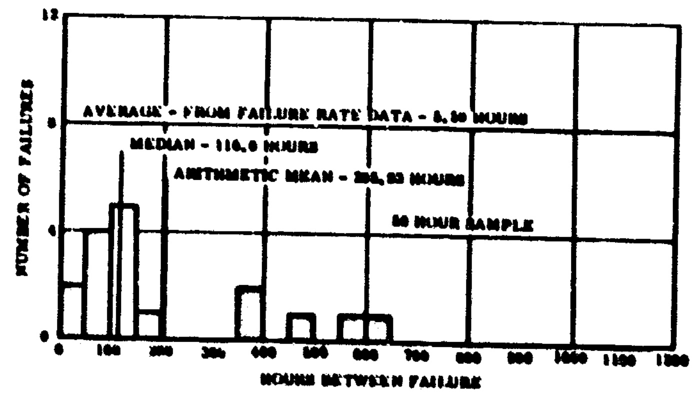

Figure 68 Time Between Failure Electrical System - (E, IP, αIS)

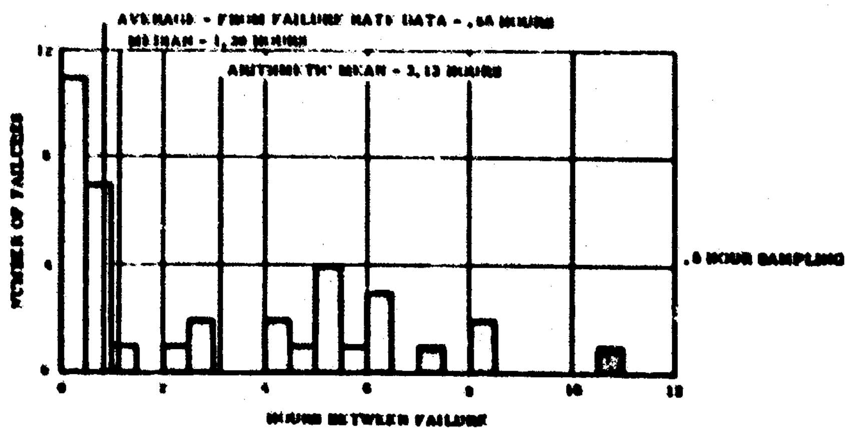

Figure 69 Time Between Failure Propulsion System Total (PL, PR, PN, PF, P)

TABLE 3
XV-5A AVERAGE FAILURE RATES

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAILURES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAILURES	CUMULATIVE RATE
AIRFRAME-FUSELAGE SYSTEM							
5 Mar.- 31 Mar. 1964	Conventional	0			0		
	Fan	.58			.58		
	Taxi	.98			.98		
	Ground	1.32			1.32		
	Average						
1 Apr. - 30 Apr. 1964	Conventional	0			0		
	Fan	.58			1.16		
	Taxi	.70			1.68		
	Ground	1.63	1	.613	2.95	1	.339
	Average			.613			.339
1 May - 31 May 1964	Conventional	2.08			2.08		
	Fan	0			1.16		
	Taxi	2.0			3.68		
	Ground	0			2.95	1	.339
	Average						.339
1 June - 30 June 1964	Conventional	2.50			4.58		
	Fan	0			1.16		
	Taxi	.22			3.90		
	Ground	.50			3.45	1	.290
	Average						.290
1 July - 31 July 1964	Conventional	0			4.58		
	Fan	.75			1.91		
	Taxi	0			3.90		
	Ground	13.30			16.75	1	.0597
	Average						.0597
1 Aug. - 31 Aug. 1964	Conventional	0			4.58		
	Fan	1.17			3.08		
	Taxi	0			3.90		
	Ground	4.30			21.05	1	.0475
	Average						.0475
1 Sept. - 30 Sept. 1964	Conventional	1.67			6.25		
	Fan	1.00			4.08		
	Taxi	2.43			6.33		
	Ground	3.12			24.17	1	.0414
	Average						.0414

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAIL-URES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAIL-URES	CUMULATIVE RATE
AIRFRAME-FUSELAGE SYSTEM (Continued Table 3)							
1 Oct. - 31 Oct. 1964	Conventional	3.0			9.25		
	Fan	1.67			5.75		
	Taxi	0			6.33		
	Ground	2.75			26.92	1	.0371
	Average						.0371
1 Nov. - 30 Nov. 1964	Conventional	8.67			17.92		
	Fan	1.92			7.67		
	Taxi	.00			6.33		
	Ground	3.23			30.15	1	.0332
	Average						.0332
1 Dec. - 31 Dec. 1964	Conventional	15.0			32.92		
	Fan	2.92			10.59		
	Taxi	0			6.33		
	Ground	1.30	1	.769	31.45	2	.0636
	Average			.769			.0636
1 Jan. - 26 Jan. 1965	Conventional	1.92			34.84		
	Fan	.83			11.42		
	Taxi	0			6.33		
	Ground	1.32			32.77	2	.0610
	Average						.0610

TABLE 4
AIRFRAME-WING SYSTEM

5 Mar. - 31 Mar. 1964	Conventional	0			0		
	Fan	.58			.58		
	Taxi	.98			.98		
	Ground	1.32			1.32		
	Average						
1 Apr. - 30 Apr. 1964	Conventional	0			0		
	Fan	.58			1.16		
	Taxi	.70			1.68		
	Ground	1.63			2.95		
	Average						
1 May - 31 May 1964	Conventional	2.00			2.00		
	Fan	0			1.16		
	Taxi	2.0			3.68		
	Ground	0	1		2.95	1	.339
	Average						.339

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAIL-URES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAIL-URES	CUMULATIVE RATE
AIRFRAME-WING SYSTEM (Continued Table 4)							
1 June - 30 June 1964	Conventional	2.50			4.58		
	Fan	0			1.16		
	Taxi	.22			3.90		
	Ground	.50	1	2.0	3.45	2	.580
	Average			2.0			.580
1 July - 31 July 1964	Conventional	0			4.58		
	Fan	.75			1.91		
	Taxi	0			3.90		
	Ground	13.30			16.75	2	.119
	Average						.119
1 Aug. - 31 Aug. 1964	Conventional	0			4.58		
	Fan	1.17			3.08		
	Taxi	0			3.90		
	Ground	4.30			21.05	2	.0950
	Average						.0950
1 Sept. - 30 Sept. 1964	Conventional	1.67			6.25		
	Fan	1.00			4.08		
	Taxi	2.43			6.33		
	Ground	3.12			24.17	2	.0827
	Average						.0827
1 Oct. - 31 Oct. 1964	Conventional	3.0			9.25		
	Fan	1.67			5.75		
	Taxi	0			6.33		
	Ground	2.75			26.92	2	.0743
	Average						.0743
1 Nov. - 30 Nov. 1964	Conventional	8.67			17.92		
	Fan	1.92			7.67		
	Taxi	.00			6.33		
	Ground	3.23			30.15	2	.0663
	Average						.0663
1 Dec. - 31 Dec. 1964	Conventional	15.00			32.92		
	Fan	2.92			10.59		
	Taxi	0			6.33		
	Ground	1.30			31.45	2	.0636
	Average						.0636

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAIL-URES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAIL-URES	CUMULATIVE RATE
AIRFRAME-WING SYSTEM (Continued Table 4)							
1 Jan. - 26 Jan. 1965	Conventional	1.92			34.34		
	Fan	.83			11.42		
	Taxi	0			6.33		
	Ground	1.32			32.77	2	.0610
	Average						.0610

TABLE 5
AIRFRAME-EMPENNAGE SYSTEM

5 Mar. - 31 Mar. 1964	Conventional	0			0		
	Fan	.58			.58		
	Taxi	.98			.98		
	Ground	1.32			1.32		
	Average						
1 Apr. - 30 Apr. 1964	Conventional	0			0		
	Fan	.58			1.16		
	Taxi	.70			1.68		
	Ground	1.63			2.95		
	Average						
1 May - 31 May 1964	Conventional	2.08			2.08		
	Fan	0			1.16		
	Taxi	2.0			3.68		
	Ground	0			3.95		
	Average						
1 June - 30 June 1964	Conventional	2.50			4.58		
	Fan	0			1.16		
	Taxi	.22			3.90		
	Ground	.50			3.45		
	Average						
1 July - 31 July 1964	Conventional	0			4.58		
	Fan	.75			1.91		
	Taxi	0			3.90		
	Ground	13.30			16.75		
	Average						
1 Aug. - 31 Aug. 1964	Conventional	0			4.58		
	Fan	1.17			3.08		
	Taxi	0			3.90		
	Ground	4.30	1	.233	21.05	1	.0475
	Average			.233			.0475

PERIOD	MODE	THIS PERIOD TIME (HOURS)	FAIL-URES	RATE	CUMULATIVE TIME (HOURS)	FAIL-URES	RATE
AIRFRAME-EMPENNAGE SYSTEM (Continued Table 5)							
1 Sept. - 30 Sept. 1964	Conventional	1.67			6.25		
	Fan	1.00			4.08		
	Taxi	2.43			6.33		
	Ground	3.12			24.17	1	.0414
	Average						.0414
1 Oct. - 31 Oct. 1964	Conventional	3.0			9.25		
	Fan	1.67			5.75		
	Taxi	0			6.33		
	Ground	2.75			26.92	1	.0371
	Average						.0371
1 Nov. - 30 Nov. 1964	Conventional	8.67			17.92		
	Fan	1.92			7.67		
	Taxi	.00			6.33		
	Ground	3.23			30.15	1	.0332
	Average						.0332
1 Dec. - 31 Dec. 1964	Conventional	15.00			32.92		
	Fan	2.92			10.59		
	Taxi	0			6.33		
	Ground	1.30			31.45	1	.0318
	Average						.0318
1 Jan. - 26 Jan. 1965	Conventional	1.92			34.84		
	Fan	.83			11.42		
	Taxi	0			6.33		
	Ground	1.32			32.77	1	.0305
	Average						.0305

TABLE 6
CONTROLS SYSTEM

5 Mar. - 31 Mar. 1964	Conventional	0			0		
	Fan	.58			.58		
	Taxi	.98			.98		
	Ground	89.32			89.32		
	Average						
1 Apr. - 30 Apr. 1964	Conventional	0			0		
	Fan	.58			1.16		
	Taxi	.70			1.68		
	Ground	105.63			194.95		
	Average						

PERIOD	MODE	THIS PERIOD			CUMULATIVE		
		TIME (HOURS)	FAIL-URES	RATE	TIME (HOURS)	FAIL-URES	RATE
CONTROLS SYSTEM (Continued Table 6)							
1 May - 31 May 1964	Conventional	2.08			2.08		
	Fan	.00			1.16		
	Taxi	2.00			3.68		
	Ground	104.00			298.95		
	Average						
1 June - 30 June 1964	Conventional	2.50			4.58		
	Fan	0			1.16		
	Taxi	.22			3.90		
	Ground	104.50	1	.00957	403.45	1	.00248
	Average			.00957			.00248
1 July - 31 July 1964	Conventional	0			4.58		
	Fan	.75			1.91		
	Taxi	0			3.90		
	Ground	121.30	2	.0165	524.75	3	.00572
	Average			.0165			.00572
1 Aug. - 31 Aug. 1964	Conventional	0			4.58		
	Fan	1.17			3.08		
	Taxi	0			3.90		
	Ground	108.30			633.05	3	.00474
	Average						.00474
1 Sept. - 30 Sept. 1964	Conventional	1.67			6.25		
	Fan	1.00			4.08		
	Taxi	2.43			6.33		
	Ground	107.12	1	.00934	740.17	4	.00540
	Average			.00934			.00540
1 Oct. - 31 Oct. 1964	Conventional	3 00			9.25		
	Fan	1.67			5.75		
	Taxi	.00			6.33		
	Ground	182.75		.00547	922.92	5	.00542
	Average			.00547			.00542
1 Nov. - 30 Nov. 1964	Conventional	8.67			17.92		
	Fan	1.92			7.67		
	Taxi	.00			6.33		
	Ground	203.23	2	.00984	1126.15	7	.00622
	Average			.00984			.00622

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAIL-URES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAIL-URES	CUMULATIVE RATE
CONTROLS SYSTEM (Continued Table 6)							
1 Dec. -	Conventional	14.17			32.92		
31 Dec.	Fan	2.92			10.59		
1964	Taxi	0			6.33		
	Ground	217.38			1343.53	7	.00521
	Average						.00521
1 Jan. -	Conventional	1.92			34.83		
26 Jan.	Fan	.83			11.42		
1965	Taxi	0			6.33		
	Ground	177.32			1520.85	7	.00460
	Average						.00460

TABLE 7
ELECTRICAL SYSTEM

5 Mar. -	Conventional	.00			.00		
31 Mar.	Fan	.58			.58		
1964	Taxi	.98			.98		
	Ground	177.32	3	.0169	177.32	3	.0169
	Average			.0169			.0169
1 Apr. -	Conventional	.00			.00		
30 Apr.	Fan	.58			1.16		
1964	Taxi	.70			1.68		
	Ground	209.63			386.95	3	.00775
	Average						.00775
1 May -	Conventional	2.06			2.06		
31 May	Fan	.00			1.16		
1964	Taxi	2.00			3.68		
	Ground	208.00			594.95	3	.00504
	Average						.00504
1 June -	Conventional	2.50			4.56		
30 June	Fan	.00			1.16		
1964	Taxi	.22			3.90		
	Ground	208.50	1	.00480	803.45	4	.00498
	Average			.00480			.00498
1 July -	Conventional	.00			4.56		
31 July	Fan	.75			1.91		
1964	Taxi	.00			3.90		
	Ground	229.30			1032.75	4	.00387
	Average						.00387

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAIL-URES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAIL-URES	CUMULATIVE RATE
ELECTRICAL SYSTEM (Continued Table 7)							
1 Aug. - 31 Aug. 1964	Conventional	.00			4.58		
	Fan	1.17	1	.855	3.08	1	.325
	Taxi	.00			3.90		
	Ground	212.30			1245.05	4	.00321
	Average			.855			.32821
1 Sept. - 30 Sept. 1964	Conventional	1.67			6.25		
	Fan	1.00	1	1.00	4.08	2	.490
	Taxi	2.43			6.33		
	Ground	211.12			1456.17	4	.00275
	Average			1.00			.49275
1 Oct. - 31 Oct. 1964	Conventional	3.00			9.25		
	Fan	1.67			5.75	2	.348
	Taxi	0			6.33		
	Ground	354.75	8	.0226	1810.92	12	.00663
	Average			.0226			.35463
1 Nov. - 30 Nov. 1964	Conventional	8.67			17.92		
	Fan	1.92			7.67	2	.260
	Taxi	.00			6.33		
	Ground	403.23			2214.15	12	.00542
	Average						.26542
1 Dec. - 31 Dec. 1964	Conventional	15.00			32.92		
	Fan	2.92			10.59	2	.189
	Taxi	.00			6.33		
	Ground	433.30	1	.00231	2647.45	13	.00491
	Average			.00231			.19391
1 Jan. - 26 Jan. 1965	Conventional	1.92			34.84		
	Fan	.83			11.42	2	.175
	Taxi	.00			6.33		
	Ground	353.32	2	.00566	3000.77	15	.00500
	Average			.00566			.18000

TABLE 8
HYDRAULIC SYSTEM

5 Mar. - 31 Mar. 1964	Conventional	.00			.00		
	Fan	.58			.58		
	Taxi	.98			.98		
	Ground	60.32			60.32		
	Average						

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAIL-URES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAIL-URES	CUMULATIVE RATE
HYDRAULIC SYSTEM (Continued Table 8)							
1 Apr. - 30 Apr. 1964	Conventional	.00			.00		
	Fan	.58			1.16		
	Taxi	.70			1.68		
	Ground	105.63			194.95		
	Average						
1 May - 31 May 1964	Conventional	2.08			2.08		
	Fan	.00			1.16		
	Taxi	2.00			3.68		
	Ground	104.00			298.95		
	Average						
1 June - 30 June 1964	Conventional	2.50			4.58		
	Fan	.00			1.16		
	Taxi	.22			3.90		
	Ground	104.50			403.45		
	Average						
1 July - 31 July 1964	Conventional	.00			4.58		
	Fan	.75			1.91		
	Taxi	.00			3.90		
	Ground	121.30	1	.00824	524.75	1	.00191
	Average			.00824			.00191
1 Aug. - 31 Aug. 1964	Conventional	.00			4.58		
	Fan	1.17			3.08		
	Taxi	.00			3.90		
	Ground	108.30	2	.0185	633.05	3	.00474
	Average			.0185			.00474
1 Sept. - 30 Sept. 1964	Conventional	1.67			6.25		
	Fan	1.00			4.08		
	Taxi	2.43			6.33		
	Ground	107.12			740.17	3	.00405
	Average						.00405
1 Oct. - 31 Oct. 1964	Conventional	3.00			9.25		
	Fan	1.67			5.75		
	Taxi	.00			6.33		
	Ground	182.75	2	.0109	922.92	5	.00542
	Average			.0109			.00542

PERIOD	MODE	THIS PERIOD TIME (HOURS)	FAIL-URES	RATE	CUMULATIVE TIME (HOURS)	FAIL-URES	RATE
HYDRAULIC SYSTEM (Continued Table 8)							
1 Nov. - 30 Nov. 1964	Conventional	8.67	1	.115	17.92	1	.0558
	Fan	1.92			7.67		
	Taxi	.00			6.33		
	Ground	203.23			1125.62	5	.00444
	Average			.115			.06024
1 Dec. - 31 Dec. 1964	Conventional	14.17			32.92	1	.0304
	Fan	2.92			10.59		
	Taxi	.00			6.33		
	Ground	217.38	2	.00920	1343.53	7	.00521
	Average			.00920			.03561
1 Jan. - 26 Jan. 1965	Conventional	1.92			34.84	1	.0287
	Fan	.83			11.42		
	Taxi	.00			6.33		
	Ground	177.32	3	.0169	1520.85	10	.00658
	Average			.0169			.03528

TABLE 9
COCKPIT SYSTEM

5 Mar. - 31 Mar. 1964	Conventional	.00			.00		
	Fan	.58			.58		
	Taxi	.96			.94		
	Ground	89.32			89.32		
	Average						
1 Apr. - 30 Apr. 1964	Conventional	.00			.00		
	Fan	.58			1.16		
	Taxi	.70			1.68		
	Ground	105.63			194.96		
	Average						
1 May - 31 May 1964	Conventional	2.06			2.06		
	Fan	.00			1.16		
	Taxi	2.00			3.68		
	Ground	104.00	1	.00962	298.96	1	.00335
	Average			.00962			.00335
1 June - 30 June 1964	Conventional	2.50			4.56		
	Fan	.00			1.16		
	Taxi	.22			3.90		
	Ground	104.50			403.46	1	.00248
	Average						.00248

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAIL-URES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAIL-URES	CUMULATIVE RATE
COCKPIT SYSTEM (Continued Table 9)							
1 July - 31 July 1964	Conventional	.00			4.58		
	Fan	.75			1.91		
	Taxi	.00			3.90		
	Ground	121.30			524.75	1	.00191
	Average						.00191
1 Aug. - 31 Aug. 1964	Conventional	.00			4.58		
	Fan	1.17			3.08		
	Taxi	.00			3.90		
	Ground	108.30			633.05	1	.00158
	Average						.00158
1 Sept. - 30 Sept. 1964	Conventional	1.67			6.25		
	Fan	1.00			4.08		
	Taxi	2.43			6.33		
	Ground	107.12			740.17	1	.00135
	Average						.00135
1 Oct. - 31 Oct. 1964	Conventional	3.00			9.25		
	Fan	1.67			5.75		
	Taxi	.00			6.33		
	Ground	182.75	2	.0109	922.92	3	.00325
	Average			.0109			.00325
1 Nov. - 30 Nov. 1964	Conventional	8.67			17.92		
	Fan	1.92			7.67		
	Taxi	.00			6.33		
	Ground	203.23			1126.15	3	.00266
	Average						.00266
1 Dec. - 31 Dec. 1964	Conventional	14.17			32.92		
	Fan	2.92			10.59		
	Taxi	.00			6.33		
	Ground	217.38	2	.00920	1343.53	5	.00372
	Average			.00920			.00372
1 Jan. - 26 Jan. 1965	Conventional	1.92			34.84		
	Fan	.83			11.42		
	Taxi	.00			6.33		
	Ground	177.32	2	.0113	1520.85	7	.00460
	Average			.0113			.00460

TABLE 10
LANDING GEAR

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAIL-URES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAIL-URES	CUMULATIVE RATE
LANDING GEAR							
5 Mar. - 31 Mar. 1964	Conventional	.00			.00		
	Fan	.58			.58		
	Taxi	.98			.98		
	Ground	1.32	1	.758	1.32	1	.758
	Average			.758			.758
1 Apr. - 30 Apr. 1964	Conventional	.00			.00		
	Fan	.58			1.16		
	Taxi	.70	2	2.86	1.68	2	1.19
	Ground	1.63			2.95	1	.339
	Average			2.86			1.529
1 May - 31 May 1964	Conventional	2.08			2.08		
	Fan	.00			1.16		
	Taxi	2.00			3.68	2	.543
	Ground	.00			2.95	1	.339
	Average						.882
1 June - 30 June 1964	Conventional	2.50			4.58		
	Fan	.00			1.16		
	Taxi	.22	1	4.55	3.90	3	.769
	Ground	.50			3.45	1	.290
	Average			4.55			1.059
1 July - 31 July 1964	Conventional	.00			4.58		
	Fan	.75			1.91		
	Taxi	.00			3.90	3	.769
	Ground	13.30	1	.0752	16.75	2	.119
	Average			.0752			.888
1 Aug. - 31 Aug. 1964	Conventional	.00			4.58		
	Fan	1.17			3.00		
	Taxi	.00			3.90	3	.769
	Ground	4.30			21.05	2	.0950
	Average						.8640
1 Sept. - 30 Sept. 1964	Conventional	1.67			6.25		
	Fan	1.00			4.00		
	Taxi	5.43	1	.306	6.33	4	.632
	Ground	3.12			24.17	2	.0824
	Average			.306			.7144

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAIL-URES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAIL-URES	CUMULATIVE RATE
LANDING GEAR (Continued Table 10)							
1 Oct. - 31 Oct. 1964	Conventional	3.00			9.25		
	Fan	1.67			5.75		
	Taxi	.00			6.33	4	.632
	Ground	2.75	3	1.09	26.92	5	.186
	Average			1.09			.818
1 Nov. - 30 Nov. 1964	Conventional	8.67			17.92		
	Fan	1.92			7.67		
	Taxi	.00			6.33	4	.632
	Ground	3.23	2	.619	30.15	7	.232
	Average			.619			.864
1 Dec - 31 Dec. 1964	Conventional	15.00			32.92		
	Fan	2.92			10.59		
	Taxi	.00			6.33	4	.632
	Ground	1.30			31.45	7	.223
	Average						.855
1 Jan. - 26 Jan. 1965	Conventional	1.92			34.84		
	Fan	.83			11.42		
	Taxi	.00			6.33	4	.632
	Ground	1.32			32.77	7	.214
	Average						.846

TABLE 11
PROPULSION-POWER PLANT SYSTEM

5 Mar. - 31 Mar. 1964	Conventional	.00			.00		
	Fan	.58			.58		
	Taxi	.98			.98		
	Ground	1.32	2	1.52	1.32	2	1.52
	Average			1.52			1.52
1 Apr. - 30 Apr. 1964	Conventional	.00			.00		
	Fan	.58			1.16		
	Taxi	.70			1.68		
	Ground	1.63			2.95	2	.678
	Average						.678
1 May - 31 May 1964	Conventional	2.08			2.08		
	Fan	.00			1.16		
	Taxi	2.00			3.68		
	Ground	.00	1		2.95	3	1.02
	Average						1.02

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAILURES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAILURES	CUMULATIVE RATE
PROPULSION-POWER PLANT SYSTEM (Continued Table 11)							
1 June - 30 June 1964	Conventional	2.50			4.58		
	Fan	.00			1.16		
	Taxi	.22			3.90		
	Ground	.50			3.45	3	.870
	Average						.870
1 July - 31 July 1964	Conventional	.00			4.58		
	Fan	.75			1.91		
	Taxi	.00			3.90		
	Ground	13.30			16.75	3	.179
	Average						.179
1 Aug. - 31 Aug. 1964	Conventional	.00			4.58		
	Fan	1.17			3.08		
	Taxi	.00			3.90		
	Ground	4.30			21.05	3	.143
	Average						.143
1 Sept. - 30 Sept. 1964	Conventional	1.67			6.25		
	Fan	1.00			4.08		
	Taxi	2.43			6.33		
	Ground	3.12			24.17	3	.124
	Average						.124
1 Oct. - 31 Oct. 1964	Conventional	3.00			9.25		
	Fan	1.67			5.75		
	Taxi	.00			6.33		
	Ground	2.75			26.92	3	.111
	Average						.111
1 Nov. - 30 Nov. 1964	Conventional	8.67			17.92		
	Fan	1.92			7.67		
	Taxi	.00			6.33		
	Ground	3.23			30.15	3	.0995
	Average						.0996
1 Dec. - 31 Dec. 1964	Conventional	15.00			32.92		
	Fan	2.92			10.59		
	Taxi	.00			6.33		
	Ground	1.30			31.45	3	.0954
	Average						.0954

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PERIOD	MODE	THIS PERIOD TIME (HOURS)	FAIL-URES	RATE	CUMULATIVE TIME (HOURS)	FAIL-URES	RATE
PROPULSION-POWER PLANT SYSTEM (Continued Table 11)							
1 Jan. -	Conventional	1.92			34.84		
26 Jan.	Fan	.83			11.42		
1965	Taxi	.00			6.33		
	Ground	1.32			32.77	3	.0915
	Average						.0915

TABLE 12
PROPULSION-FUEL SYSTEM

5 Mar. -	Conventional	.00			.00		
31 Mar.	Fan	.58			.58		
1964	Taxi	.98			.98		
	Ground	1.32	2	1.52	1.32	2	1.52
	Average			1.52			1.52
1 Apr. -	Conventional	.00			.00		
30 Apr.	Fan	.58			1.16		
1964	Taxi	.70			1.68		
	Ground	1.63			2.95	2	.678
	Average						.678
1 May -	Conventional	2.08			2.08		
31 May	Fan	.00			1.16		
1964	Taxi	2.00			3.68		
	Ground	.00			2.95	2	.678
	Average						.678
1 June -	Conventional	2.50			4.58		
30 June	Fan	.00			1.16		
1964	Taxi	.22			3.90		
	Ground	.50			3.45	2	.580
	Average						.580
1 July -	Conventional	.00			4.58		
31 July	Fan	.75			1.91		
1964	Taxi	.00			3.90		
	Ground	13.30	1	.0752	16.75	3	.179
	Average			.0752			.179
1 Aug. -	Conventional	.00			4.58		
31 Aug.	Fan	1.17			3.08		
1964	Taxi	.00			3.90		
	Ground	4.30	1	.233	21.05	4	.190
	Average			.233			.190

PERIOD	MODE	THIS PERIOD TIME (HOURS)	FAIL-URES	RATE	CUMULATIVE TIME (HOURS)	FAIL-URES	RATE
PROPULSION-FUEL SYSTEM (Continued Table 12)							
1 Sept. - 30 Sept. 1964	Conventional	1.67			6.25		
	Fan	1.00			4.08		
	Taxi	2.43			6.33		
	Ground	3.12			24.17	4	.165
	Average						.165
1 Oct. - 31 Oct. 1964	Conventional	3.00			9.25		
	Fan	1.67			5.75		
	Taxi	.00			6.33		
	Ground	2.75	2	.727	26.92	6	.223
	Average			.727			.223
1 Nov. - 30 Nov. 1964	Conventional	8.67			17.92		
	Fan	1.92			7.67		
	Taxi	.00			6.33		
	Ground	3.23	2	.619	30.15	8	.265
	Average			.619			.265
1 Dec. - 31 Dec. 1964	Conventional	15.00	2	.133	32.97	2	.0607
	Fan	2.92			10.59		
	Taxi	.00			6.33		
	Ground	1.30	1	.769	31.45	9	.286
	Average			.902			.3467
1 Jan. - 26 Jan. 1965	Conventional	1.92			34.84	2	.0574
	Fan	.83			11.42		
	Taxi	.00			6.33		
	Ground	1.32	4	3.03	32.77	13	.397
	Average			3.03			.4544

TABLE 13
PROPULSION-MISCELLANEOUS SYSTEM

5 Mar. - 31 Mar. 1964	Conventional	.00			.00		
	Fan	.58			.58		
	Taxi	.98			.98		
	Ground	1.32	2	1.52	1.32		1.52
	Average			1.52			1.52
1 Apr. - 30 Apr. 1964	Conventional	.00			.00		
	Fan	.58			1.16		
	Taxi	.70			1.68		
	Ground	1.63			2.96	2	.678
	Average						.678

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PERIOD	MODE	THIS PERIOD TIME (HOURS)	FAIL-URES	RATE	CUMULATIVE TIME (HOURS)	FAIL-URES	RATE
PROPULSION-MISCELLANEOUS SYSTEM (Continued Table 13)							
1 May 31 May 1964	Conventional	2.08	1	.481	2.08	1	.481
	Fan	.00			1.16		
	Taxi	2.00			3.68		
	Ground	.00	2	-	2.95	4	1.36
	Average			.481			1.841
1 June - 30 June 1964	Conventional	2.50			4.58	1	.218
	Fan	.00			1.16		
	Taxi	.22			3.90		
	Ground	.50	2	4.0	3.46	6	1.74
	Average			4.0			1.958
1 July - 31 July 1964	Conventional	.00			4.58	1	.218
	Fan	.75			1.91		
	Taxi	.00			3.90		
	Ground	13.30			16.75	6	.358
	Average						.576
1 Aug. - 31 Aug. 1964	Conventional	.00			4.58	1	.218
	Fan	1.17			3.08		
	Taxi	.00			3.90		
	Ground	4.30	1	.233	21.05	7	.333
	Average			.233			.551
1 Sept. - 30 Sept. 1964	Conventional	1.67			6.25	1	.160
	Fan	1.00			4.08		
	Taxi	2.43			6.33		
	Ground	3.12	1	.321	24.17	8	.331
	Average			.321			.491
1 Oct. - 31 Oct. 1964	Conventional	3.00			9.25	1	.108
	Fan	1.67			5.75		
	Taxi	.00			6.33		
	Ground	2.75	3	1.09	26.92	11	.409
	Average			1.09			.517
1 Nov. - 30 Nov. 1964	Conventional	8.67			17.92	1	.0558
	Fan	1.92	1	.521	7.67	1	.130
	Taxi	.00			6.33		
	Ground	3.23	2	.619	30.15	13	.431
	Average			1.140			.6168

PERIOD	MODE	THIS PERIOD TIME (HOURS)	THIS PERIOD FAIL-URES	THIS PERIOD RATE	CUMULATIVE TIME (HOURS)	CUMULATIVE FAIL-URES	CUMULATIVE RATE
PROPULSION-MISCELLANEOUS SYSTEM (Continued Table 13)							
1 Dec. - 31 Dec. 1964	Conventional	15.00			32.92	1	.0303
	Fan	2.92			10.59	1	.0944
	Taxi	.00			6.33		
	Ground	1.30	2	1.54	31.45	15	.477
	Average			1.54			.6017
1 Jan. 0 26 Jan. 1965	Conventional	1.92			34.84	1	.0287
	Fan	.83			11.42	1	.0876
	Taxi	.00			6.33		
	Ground	1.32	2	1.52	32.77	17	.519
	Average			1.52			.6353

TABLE 14
PARACHUTE SYSTEM

1 Nov. - 30 Nov. 1964	Conventional				17.92		
	Fan				7.67		
	Taxi				6.33		
	Ground				30.15		
	Average						
1 Dec. - 31 Dec. 1964	Conventional	15.00	3	.200	32.92	3	.0911
	Fan	2.92			10.59		
	Taxi	.00			6.33		
	Ground	1.30			31.45		
	Average						.0911
1 Jan. 0 26 Jan. 1965	Conventional	1.92			34.84	3	.0861
	Fan	.83			11.42		
	Taxi	.00			6.33		
	Ground	1.32			32.77		
	Average						.0861

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TABLE 15
XV-5A CUMULATIVE FAILURE RATE SUMMARY

PERIOD	MODE	SYSTEM: AIRFRAME	CONTROLS	ELECTRICAL	HYDRAULIC	COCKPIT	LANDING GEAR	PROPULSION	PARACHUTE	TOTAL AIRCRAFT (System Summation)
5 Mar. - 31 Mar. 1964	Conventional									
	Fan									
	Taxi									
	Ground			.0169			.758	4.56		5.3349
	Total			.0169			.758	4.56		5.3349
1 Apr. - 30 Apr. 1964	Conventional									
	Fan									
	Taxi						1.19			1.19
	Ground	.339		.00775			.339	2.034		2.71975
	Total	.339		.00775			1.529	2.034		3.90975
1 May - 31 May 1964	Conventional							.481		.481
	Fan									
	Taxi						.543			.543
	Ground	.679		.00504		.00335	.339	3.059		3.74776
	Total	.678		.00504		.00335	.882	3.539		4.76178
1 June - 30 June 1964	Conventional							.218		.218
	Fan									
	Taxi						.769			.769
	Ground	.870	.00248	.00498		.00248	.290	3.190		4.35994
	Total	.870	.00248	.00498		.00248	1.059	3.408		5.34894
1 July - 31 July 1964	Conventional							.218		.218
	Fan									
	Taxi						.769			.769
	Ground	.1787	.00572	.00387	.00191	.00191	.119	.716		1.02711
	Total	.1787	.00572	.00387	.00191	.00191	.888	.934		2.01411
1 Aug. - 31 Aug. 1964	Conventional							.218		.218
	Fan			.325						.325
	Taxi						.769			.769
	Ground	.1900	.00474	.00321	.00474	.00158	.0950	.666		.96527
	Total	.1900	.00474	.32821	.00474	.00158	.8640	.884		2.22727
1 Sept. - 30 Sept. 1964	Conventional							.160		.160
	Fan			.490						.490
	Taxi						.632			.632
	Ground	.1655	.00540	.00275	.00405	.00135	.0824	.620		.88145
	Total	.1655	.00540	.49275	.00405	.00135	.7144	.780		2.16345
1 Oct. - 31 Oct. 1964	Conventional							.106		.106
	Fan			.348						.348
	Taxi						.632			.632
	Ground	.1465	.00542	.00463	.00542	.00325	.186	.743		1.09822
	Total	.1465	.00542	.35463	.00542	.00325	.818	.851		2.19822
1 Nov. - 30 Nov 1964	Conventional				.0558			.0558		.1116
	Fan			.260				.130		.390
	Taxi						.632			.632
	Ground	.1327	.00822	.00542	.00444	.00266	.232	.7955		1.17894
	Total	.1327	.00822	.26542	.06024	.00266	.864	.9913		2.31254
1 Dec. - 31 Dec. 1964	Conventional				.0304			.0910	.0911	.2125
	Fan			.189				.0944		.2834
	Taxi						.632			.632
	Ground	.1590	.00521	.00491	.00521	.00372	.223	.8584		1.26946
	Total	.1590	.00521	.19391	.03561	.00372	.855	1.0434	.0911	2.30736
1 Jan. - 23 Jan. 1965	Conventional				.0287			.0861	.0861	.2009
	Fan			.175				.0876		.2626
	Taxi						.632			.632
	Ground	.1825	.00460	.00500	.00658	.00460	.214	1.0075		1.39478
	Total	.1825	.00460	.18000	.03528	.00460	.846	1.1812	.0861	2.48028

10.0 COMPONENTS

All components used have been qualified for use in this aircraft according to provisions of the contract. The parts have, by one procedure or another, been found satisfactory for flight according to requirements set up by the Design Engineering Group. These procedures were as follows: use of MIL-STD-QPL parts, use of aircraft industry STD parts, use of parts as designed with test procedures required by Design Engineering Group, and by similarity to parts already qualified for use on other aircraft.

Proof of compliance for parts was accomplished by several methods; certification, designers witnessing required tests, formal reporting of proof tests, and common agreement of the manufacturers capability plus functional tests in the case of some industry standard parts.

Listing of these parts and methods follows:

TABLE 16
COMPONENT QUALIFICATION DATA

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI- CATION	COMPLIANCE DATA	QUALIFICATION STATUS
MAIN LANDING GEAR SYSTEM - MAJOR COMPONENTS				
1510L000 Loud Co. Code 75662	MLG Assy.	SCD L0001	Ryan Static Test Reports 64B026, 63B048, 64B024. Ryan Installed Systems Functional Test Report 64B089. Ryan ITO 1112. Loud Acceptance Test Procedure 1510LTP3.	Qualified
1510L100 Loud Co. Code 75662	MLG Shock Strut Assy.	SCD L0001	Loud Drop Test Procedure 1510LTP4, Rev. "A" Loud Drop Test Report 1510LTR-1, Rev. "A" (Also Published as Ryan Report 64B044)	Qualified
942106 (PD 2212) Goodyear Code 73842	Brake Assy.	SCD L0003	Goodyear Test Plan GA 1094R Goodyear Qual. Test Report GA 118R	Qualified
9533223 Goodyear 73842	Main Wheel Assy.	SCD L0003	Same as Above	Qualified
20 x 4.4 Type VII 12PR Goodyear Code 73842	Tire	56D1171	Standard Equipment	Qualified
A62280 Vinson Code 91130	MLG 2-Position Actuator	SCD L0008	Vinson Test Procedure HQTP 62280 Ryan ITO 1130	Qualified
A62276 Vinson Code 91130	MLG Door Actuator	SCD L0006	Vinson Test Procedure PTP-62276 Ryan ITO 1129	Qualified
A62378 Vinson Code 91130	MLG Unlatch Actuator	SCD L0009	Vinson Test Procedure PTP-62378 Ryan ITO 1159	Qualified
24320 Sterer Code 99643	Brake Master Cyl.	SCD K0013	Sterer Acceptance Test Procedure 24320	Qualified
NOSE LANDING GEAR - MAJOR COMPONENTS				
1511L100 Loud Co. Code 75662	NLG Shock Strut Assy.	SCD L0002	Ryan Static Test Reports 64B026, 64B024, 63B048, Ryan Installed Systems Functional Test Report 64B089, Ryan ITO 1113, Loud Acceptance Test 1511 LTP-3, Loud Drop Test Procedure 1511 LTP4 Rev. "A", Loud Drop Test Report 1511 LTR-1, Shimmy Test Summary (See Text)	Qualified
1511L200 Loud Co. Code 75662	NLG Drag Brace Assy.	SCD L0002	Same as 1511L100	Qualified
1511L400-501 Loud Co. Code 75662	Shimmy Damper Assy.	SCD L0002	Shimmy Test Summary. Loud Qual. Test Report 1511-LTR-2 Rev. "A" Loud Acceptance Test Procedure 1511 LTP-5	Qualified

TABLE 16 (Continued)

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI-CATION	COMPLIANCE DATA	QUALIFICATION STATUS
NOSE LANDING GEAR SYSTEM - MAJOR COMPONENTS (Continued)				
3-1124 B. F. Goodrich Code 97153	Nose Wheel Assy.	SCD L0004	Ryan ITO 1115 Similar to Goodrich 3-933 (Used on T28 Aircraft)	Qualified
16 x 4.4 Type VII-10 PR B. F. Goodrich Code 97153	Tire		FAA 16 x 4.4 - 10TL-2001 Standard Equipment	Qualified
LANDING GEAR SYSTEM - MISCELLANEOUS COMPONENTS				
404EN1-6 Microswitch Code 91929	Limit Switch	MS21321-2		Qualified
402EN1-6 Microswitch Code 91929	Limit Switch	MS21321-1		Qualified
DREM 5-050 Southwest Products Code 81376	Rod End	Mfg'r.	DOD Approved Source Southwest Products Report D-125A	Qualified
DREM 4-060 Southwest Products Code 81376	Rod End	Mfg'r.	DOD Approved Source Southwest Products Report D-125A	Qualified
NOTE: Hydraulic and Pneumatics System Components Associated with Landing Gear Operation are Listed in "Hydraulics and Controls" Section.				
COCKPIT SYSTEM				
A50M3 Avionics Products Corp. Code 99146	Landing Gear Control		Pending	
R1240-3 Radar Relay, Inc. Code 00712	Master Caution	ITO-1140	This Unit is Qualified for use on F104G and as such is Considered Qualified for use on XV-5A when tested to ITO 1140.	Qualified
R4077 Radar Relay, Inc. Code 00712	Fire Warn.	MIL-E-5272	Similarity to R1005 qualified for use on Norair F-5.	Qualified
R4073C Radar Relay, Inc. Code 00712	Annunciator Panel	MIL-E-5272 ITO-1140	This Unit is Qualified for use on F104G and as such is Considered Qualified for use on XV-5A when Tested to ITO 1140.	Qualified
R1000 Hetherington Code 97707	Switch	To Meet MIL-S-6743	Qualified for use on Convair 880 and Thereby Considered Qualified for use on XV-5A.	Qualified
LW-3 NAA, Columbus Code 80972	Ejection Seat	NA63H-62	See North American Aviation-Columbus Report No. NA63H-617	Qualified

TABLE 16 (Continued)

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI-CATION	COMPLIANCE DATA	QUALIFICATION STATUS
		COCKPIT SYSTEM (Continued)		
8566 Rosemont Eng'r'g. Co. Code 94274	Pilot Static Tube	SCDT0001-3 IT-0168	Qualified for use on other contracts (QZC).	Qualified
7U7315 Ronson Hyd. Units Code 98215	Valve 3-Way Solenoid	7315T	Test Procedures According to 7315T Were Conducted and Found Satisfactory	Qualified
D3WB Specialities, Inc. Code 10630	Instantaneous Vertical Speed Indicator	FAA TSO C8a SAE AS394	This Instrument is Qualified for use on Commercial Aircraft and Under SL29004 for Army Helicopters.	Qualified
1201126-0 Pacific Scientific Code 45402	Speed Sensor	NA62H-62 IT-302	Tested and Found Acceptable See IT 302 dated 4-2-63	Qualified
561248 Adams Rite Mfg. Co. Code 80477	Uni-Direction Brake	SCD K0011-1	Similar to Boeing 707 Part D10-60721 See American Laboratories Rep #93583	Qualified
4TL55-3 Microswitch Code 91929	Switch	MIL-S-3950A	Similar to MS24525 But With a Modified Handle	Qualified
W302 Controls Co. of America Code 05402	Switch	MIL-S-6743	Designed and Made to the Essentials of this MIL Spec and Tested Functionally. In Use on Similar Installations	Qualified
6IMJ21GAR General Electric Code 97424	Fuel Flow Indicator		Pending	
1045-1-4 U.S. Gauge Code 09005	Oil Pressure Indicator		This Instrument is Same as SHD 7K Except for Dial. Considered Qualified by Similarity	Qualified
SHD 7K U.S. Gauge Code 09005	Hydraulic Pressure		This Instrument is Same as P/N 826170-1 Qualified for Use on the Lockheed Electra	Qualified
4TL1-7D Microswitch Code 91929	Switch	MIL-S-3950A	Similar to MS24525 Which is Qualified. The Only Difference is the Outside Shape	Qualified
2TL 61-7 Microswitch Code 91929	Switch	MIL-S-3950A	Similar to MS24524 Which is a Qualified Switch. The Difference is the Outside Shape.	Qualified
N2100 Controls Co. of America	Switch	Made MIL-S-6743	Designed and Made to the Requirements of MIL-S-6743 and in Use on Many Installations. This Switch was Believed Adequate	Qualified
W301 Controls Co. of America Code 05402	Switch	MIL-S-6743	Same as Above	

TABLE 16 (Continued)

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI-CATION	COMPLIANCE DATA	QUALIFICATION STATUS
HYDRAULICS AND CONTROLS SYSTEM				
061-13098 Moog Servo-mechanisms, Inc. Code 94697	Servo Actuator-Exit Louver (Fwd.)	SCD H0002	Qualified in Accordance with Moog Report MR482	Qualified
061-13099 Moog Servo-mechanisms, Inc. Code 94697	Servo Actuator-Exit Louver (Aft)	SCD H0002	Qualified in Accordance with Moog Report MR482	Qualified
010-30728 Moog Servo-mechanisms, Inc. Code 94697	Servo Actuator-Pitch Fan Control	SCD H0003	Qualified in Accordance with Moog Report MR481	Qualified
23270 Sterer Eng. & Mfg. Co. Code 09643	Reservoir - Bootstrap	SCD H0006	Design approved by Stress-Qualification Consisted of Proof Pressure Test to 4500 PSI, Performance Tests per Dwg. & Approx. 500 Hours of Operation Under Simulated Flight Conditions on the Controls Simulator.	Qualified as Noted
57003 Kellogg Div. American Brake Shoe Co. Code 78350	Pump-Variable Displacement	SCD H0007	Qualification Consisted of Performance Testing to Ryan Report IT 1150 and Approx. 500 Hours of Operation Under Simulated Flight Conditions on the Controls Simulator.	Qualified as Noted
84200 Bertea Products Code 92105	Servo Actuator Aileron Boost	SCD H0010	Bertea Report No. 1133	Qualified
2053 Aqualite Corp Code [illegible]	Restrictors - Fixed Orifice and One-Way	SCD H0012	Qualified by Approx. 500 Hours of Operation Under Simulated Flight Conditions on the Controls Simulator	Qualified as Noted
6-12000 Prosser Industries, Inc. Code 19430	Actuator, Hyd. - Thrust Spoiler	SCD H0013	Design Approved by Stress-Qualification Consisted of Proof Pressure Test to 4500 PSI & Performance Tests per Prosser Report ITP6-12000	Qualified as Noted
1300-363322 Parker-Hannifin Corp Code 40681	Accumulator	MIL-A-8897	Qualified in Accordance with MIL-A-8897 per Parker-Hannifin Dwg. 1300-363322	Qualified
1T30068 Bendix Corp Code [illegible]	Filter Assy.	MS-28730-6	Qualified by Similarity to Previously Qualified Units plus Approx. 500 Hours of Operation under Simulated Flight Conditions on the Controls Simulator	Qualified as Noted
1T30069 Bendix Corp Code [illegible]	Filter Assy.	MS-28730-4	Qualified by Similarity to Previously Qualified Units plus Approx. 500 Hours of Operation under Simulated Flight Conditions on the Controls Simulator	Qualified as Noted

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TABLE 16 (Continued)

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI-CATION	COMPLIANCE DATA	QUALIFICATION STATUS
HYDRAULICS AND CONTROLS SYSTEM (Continued)				
1732882 Bendix Corp. Code [illegible]	Filter Assy. -In Line	Dwg. 1732882	Same as Above	Qualified as Noted
A-62184 Vinson Mfg. Co. Code 91136	Relief Valve	Dwg. A-62184	Qualification Consisted of Proof Pressure Test and Performance Tests per Ryan Report [illegible] plus Approx. 500 Hours of Operation Under Simulated Flight Conditions on the Controls Simulator.	Qualified as Noted
A-61562-[illegible] Vinson Mfg. Co. Code 91136	Relief Valve	MS-28893	Functionally Equivalent to MS 28893 per Vendor's Dwg.	Qualified
A-63007 Vinson Mfg. Co. Code [illegible]	Priority Valve	Dwg. A-63007	Qualified by Similarity-Identical to A-40217 Except Pressure Settings which was Qualified to Convair Spec. CVAC [illegible] "B"	Qualified
362-[illegible] Parker Aircraft Co. Code [illegible]	Selector Valve	Dwg. 362-0371	Parker Report No. [illegible]	Qualified
263-0075 Parker Aircraft Co. Code [illegible]	Selector Valve	Dwg. 263-0075	Qualified by Similarity to Above plus Approx. 500 Hours of Operation Under Simulated Flight Conditions on the Controls Simulator	Qualified as Noted
141915 Dumont Eng. Co. Code 97920	Swivel Fitting	Dwg. 141915	Qualified by Similarity to Previously Qualified Units plus Approx. 500 Hours of Operation Under Simulated Flight Conditions on the Controls Simulator	Qualified as Noted
72317 Custom Components Code [illegible]	Pressure Differential Switch	Dwg. 72317	Same as Above	Qualified as Noted
[illegible] Custom Components Code [illegible]	Pressure Switch	Dwg. [illegible]	Same as Above	Qualified as Noted
[illegible] Custom Components Code [illegible]	Pressure Switch	Dwg. [illegible]	Qualified by Similarity to Previously Qualified Units plus Performance Tests per Ryan Report [illegible]	Qualified as Noted
[illegible] Custom Components Code [illegible]	Pressure Switch	Dwg. [illegible]	Same as Above	Qualified as Noted
13530 Consolidated Controls Corp. Code [illegible]	Shutoff Valve	Dwg. 13530	Qualified per Vendor's Test Report [illegible] T-13530	Qualified
[illegible] M C Mfg. Co. Code [illegible]	Relief Valve	Dwg. 1613	Qualified per Vendor's Report No. 43	Qualified

TABLE 16 (Continued)

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI-CATION	COMPLIANCE DATA	QUALIFICATION STATUS
HYDRAULICS AND CONTROLS SYSTEM (Continued)				
R23000 Series Resistoflex Corp. Code 08051	Hose Assy.	ARP 604A	Resistoflex Report TRA-1534	Qualified
1819DA Sargent Eng. Corp. Code 78082	Shuttle Valve Landing Gear Emergency Pneumatic System	Convair Spec. 8-08364	1819DA is Functionally identical to Sargent No. 385DA-3 Built to Convair F106 Spec. See Sargent Report No. 385A983	Qualified
2725-718 Rochester Mfg. Co. Code 51240	Pressure Gauge Landing Gear Emer. Pneumatic	MIL-E-5272A	Mfg. Brochure States Movement Meets Requirements of MIL-E-5272A, Procedure I	Qualified
ELECTRICAL SYSTEM				
L10-7 Airborne Code 81030	Actuator-Wing Fan Door Latch	SCD E0028	Ven. Q.C.T. Report No. 287 Ven. Test Report No. QCSL10-7 Ryan Test Report No. IT 1160	Qualified
L12-52 Airborne Code 81030	Actuator - Aileron Trim	SCD E0034	Ven. Q.C.T. Report No. 292 Ven. Test Report No. QCSL12-52 Ryan Test Report No. IT 1165	Qualified
D1518 EEMCO Code 72121	Actuator - Wing Flaps	SCD E0039	Ven. Q.C.T. Report No. 818 Appendix No. 1 Report No. 2706 Ryan Test Report No. IT 1167	Qualified
SYLC 8840 Barber Colman Code 06624	Actuator - VTOL Roll Trim	SCD E0044-1	Ven. Qual. by Similarity to -3 Actuator Ven. Test Report No. SYL1543 Ryan Test Report No. IT 1131	*
SYLC 8849 Barber Colman Code 06624	Actuator - VTOL Yaw Trim	SCD E0044 -3* (Qual. Unit)	Ven. Q.C.T. Report No. SYL1540 Ven. Test Report No. SYL1544 Ryan Test Report No. IT 1131	Qualification Pending
SYLC 8850 Barber Colman Code 06624	Actuator - VTOL Pitch Trim	SCD E0044 -5	Ven. Qual. by Similarity to -3 Actuator Ven. Test Report No. SYL1545 Ryan Test Report No. IT 1131	*
SYLC 8860 Barber Colman Code 06624	Actuator - Thrust Vector	SCD E0045	Ven. Q.C.T. Report No. SYL1541 Ven. Test Report No. SYL1547 Ryan Test Report No. IT 1153	Qualification Pending
L12-50 Airborne Code 81030	Actuator - Rudder Trim	SCD E0040	Ven. Q.C.T. Reports No. 290 & 188 Ven. Test Report No. QCSL12-50 Ryan Test Report No. IT 1164	Qualified
G08810 Guardian Elec Code 82403	Voltage Sensor Time Delay, SPDT	SCD E0053	Ven. Q.C.T. Report No. 14789 Ryan Test Report No. IT 1161	Qualification Pending
G08811 Guardian Elec Code 82403	Voltage Sensor Time Delay, DPDT	SCD E0054	Ven. Q.C.T. Report No. 14788 Ryan Test Report No. IT 1162	Qualification Pending

TABLE 16 (Continued)

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI-CATION	COMPLIANCE DATA	QUALIFICATION STATUS
		ELECTRICAL SYSTEM (Continued)		
L12-51 Airborne Code 81039	Actuator- Aileron Droop	SCD E0059	Ven. Q. C. T. Report No. 291 Ven. Test Report No. QCSL12-51 Ryan Test Report No. IT 1166	Qualified
R-5284 Airborne Code 81039	Actuator - Pitch Fan Inlet Louver	SCD E0066	Ven. Q. C. T. Report No. 304 Ven. Test Report No. QCS R-5284 Ryan Test Report No. IT 1173	Qualification Pending
2CMD99D1 Gen. Elec. Co. Code 01526	Generator - Brushless	BDC-165-1	Ven. has Tested & Qual. by Similarity Prototype Tests on Aerocommander Aircraft. Ryan Test Report No. IT 1169	Qualified
3S2060DC125A1 Gen. Elec. Co. Code 01526	Cont. Panel - Generator, Brushless	BDC-165-1	Ven. has Tested & Qual. by Similarity Prototype Testing on Aerocommander Aircraft. Ryan Test Report No. IT 1170	Qualified
32B50-4 Bendix Corp. Code 83298	Inverter (MS21983-1)	MIL-I-7032	Ven. Q. P. L. 7032 & in Addition Meets Ryan Low Voltage per Exhibit: - Ven. Test Report No. ESO 1474	Qualified
17-S-25 Elec. Storage Batt. Code 11511	Battery - Silver-Zinc	ALM143E-14 (MIL-B-1181)	Ven. Quals. by Similarity to Existing Navy & Air Force Specs. Exhibit Typ Cell Data Bul. 7-11000 Type S-25	Qualified
14602 Bendix Corp. Code 19315	Phase Adapter	MIL-I-5133B *	*Ven. Certifies Part Will Properly Supply Power to MIL-I-5133B Attitude Indicator for Which he is Q. P. L. Exhibit Dwg. X1804489 & Specs. Ryan Test Report No. IT 1138	Qualified
77-775 Arnold Corp Code [illegible]879	Transformer	MIL-T-27A MIL-E-5272C	Vendor Certifies to Specs-Exhibit Vendor Dwg. 03. 11. 008	Qualified
BR9AX-G7-V3 Babcock Corp. Code 82050	Relay - Magnetic Latching	MIL-R-5757 MIL-R-25018	Vendor Certifies to Specs Except for Form Factor Ryan Test Report No. IT 1124	Qualified
BR7X-300D7-26V Babcock Corp. Code 82050	Relay - DPDT	MIL-R-5757 MIL-R-25018	Vendor Certifies to Specs Except for Form Factor Ryan Test Report No. IT 1125	Qualified
BR14X-150B4-26V Babcock Corp Code 82050	Relay - 4PDT	MIL-R-5757	Vendor Certifies to Spec Except for Form Factor Ryan Test Report No. IT 1126	Qualified
2112-D-H3 Agastat (ESNA) Code 98403	Relay - Time Delay	MIL-E-5272	Ven. Certifies to MIL-E-5272 Has Qual. for "Minute Man" & "Titan" Projects. Ryan Test Reports No. IT 1127 & IT 1130	Qualified
DH-71. Hartman Elec. Code 74063	Relay - Contactor	MIL-R-6106	Ven. Meets MIL-R-6106 Except for Form Factor. Ryan Test Report No. IT 1132	Qualified

TABLE 16 (Continued)

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI-CATION	COMPLIANCE DATA	QUALIFICATION STATUS
			ELECTRICAL SYSTEM (Continued)	
4-14137 Daystrom Corp. Code 98350	Relay - Sensitive Set Point	MIL-E-5272	Vendor Certifies to MIL-E-5272 Ryan Test Reports IT 1141 & IT 1160	Qualified
AU-0643 Jordan Elec. Code 01878	Signal - Audible Warning	MIL-S-9320	Vendor Certifies to Dwg. Exhibit Dwg. No. DI-0393 & Funct. Simil. to MIL-S-9320 Except for Frequency	Qualified
324-28-2 Edison Co. Code 80099	Control Assy. (Fire and Overheat Wing)	MIL-D-7006	Vendor & System Qual. & In Use on the Following Aircraft: F-102, F-106, DC-8, C-133, T2J, T-37, T-38, T-39, F8U, F9F, B-70	Qualified
90131 Ryan Aero. Co. Code 78022	Dual Timer Assy. .5 Sec	12459-239	Ryan Q.C.T. Report 12459-230 Ryan Funct. Test Report IT 0647 Qual. for Air Force on Q2C Target	Qualified
T106-10-68-C Packard Bell Code 45413	Connector - Elec., Plug	MIL-C-5015	Vendor Certifies to Exceed 5015 Specs. has Qual. for A.E.C. Use (Special - High Temp. Connector)	Qualified
CARX-TYPE Cannon Elec. Code 71468	Connectors - Elec. Plug/Receptacle	MIL-C-5015 MS 3190	Common Usage Where MS 5015 Type Req'd. Vendor Certifies to 5015 Similarity Except for MS 3190 Type Pins	Qualified
PT8E-TYPE Bendix-Scintilla Code 77820	Connectors - Elec. Plug/Receptacle	MIL-C-26482 MS 3190	Common Usage Where Pygmy Type Req'd. Ryan Evaluates "Best Pygmy Crimp". Vendor Certifies to 26482 Similarity Except for MS 3190 Type Pins.	Qualified
PTE-TYPE Bendix-Scintilla Code 77820	Connectors - Elec. Plug	MIL-C-26482	Limited Usage Where Installed Equip. Has Parent Connector Requirement. Vendor Certifies by Similarity to 26482	Qualified
DS-TYPE Deutsch Co. Code 17419	Connectors - Elec. Plug	MIL-C-26482	Limited Usage Where Installed Equip. Has Parent Connector Requirement. Vendor Certifies to 26482	Qualified
A/B43-TYPE Microdot, Inc. Code 98278	Connectors - Elec.- Multi-pin Type	MIL-C-26482	Limited Usage Where Installed Equip. Has Parent Connector Requirement Vendor Certifies to 26482	Qualified
------- Amp, Inc. Code 00779	Terminals - Wire - Elec.	MIL-T-7928	Vendor Meets MIL-T-7928 All Terminals Used Have MS25036 Equivalents.	Qualified
------- Thos. & Betts Code 59730	Sleeves - Grounding Sheath	MIL-F-21608	Vendor QPL 21608	Qualified
S20 GN 1 J20 GN 1 (Ryan Mat Code)	Cable-Elec. Special Purpose	MIL-W-5086	Wire Meets 5086 Except Has Braided Shield	Qualified
SH 20 JH 20 T (Ryan Mat Code)	Cable - Elec. Special Purpose	MIL-W-7139	Wire Meets 7139 Except Has Braided Shield	Qualified

TABLE 16 (Continued)

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI-CATION	COMPLIANCE DATA	QUALIFICATION STATUS
ELECTRICAL SYSTEM (Continued)				
B20(19)N XXXX (Ryan Mat Code)	Wire - Elec. Color Coded	MIL-W-16878	Procurement per QPL 16878	Qualified
WI 6528 Revere Corp. Code 50625	Thermocouple Lead Iron/Const.	MIL-W-5845	Vendor QPL 5845	Qualified
WI 6529 Revere Corp. Code 50625	Thermocouple Lead Iron/Const.	MIL-W-5845	Vendor QPL 5845	Qualified
WC 4142 Revere Corp. Code 50625	Thermocouple Lead Chro/Alum	MIL-W-5846	Vendor QPL 5846	Qualified
WC 6583 Revere Corp. Code 50625	Thermocouple Lead Chro/Alum	MIL-W-5846	Vendor QPL 5846	Qualified
MP 700 Series Mech. Prod., Inc. Code 76374	Circuit Breaker	MIL-C-5809	Vendor QPL 5809	Qualified
U.S. Time P/N 300850 - United States Time Corp. Code 61515	Three Axis Rate Gyro Assembly	Ryan Spec. SCD-X-0014	U.S. Time Corp. Test Procedure (U.S.T. 1082) Ryan Report IT-1136 U.S. Time Corp. Certification of Compliance	Qualified
Ryan Electronics P/N 500913-01 Code 07765	Stabilization Control Assembly	Ryan Spec SCD-X-0015	Ryan Aeronautical Co. Certification of Compliance	Qualified
PROPULSION SYSTEM				
1-000526-1 Aeroflex Code 10212	Bellows 11.5 in. Dia.	SCD P0011-1	Tested in conjunction with G.E. Lift Fan Qualification on G.E. Evendale Test Facility Approx. 130 hours	Qualified
1-000526-3 Code 10212	Bellows 11.5 in. Dia.	SCD P0011-3	Same as -1	Qualified
1-000526-5 Code 10212	Bellows 16.5 in. Dia.	SCD P0011-5	Same as -1	Qualified
1-000530-7 Code 10212	Bellows 16.5 in. Dia.	SCD P0011-7	Tested in conjunction with Tailpipe Tests on G.E. Evendale Test Facility. Approx. 30 hours	Qualified
0280-1-1 Rohrick Code 07811	Valve - Air Check J-85 Mart	SCD P0012	Pending	
1-000522 Aeroflex Code 10212	Pin Joint Assy.	SCD P0014	Tested in conjunction with Pitch Fan Qualification on G.E. Evendale Test Facility. Approx. 130 hours	Qualified

TABLE 16 (Continued)

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI-CATION	COMPLIANCE DATA	QUALIFICATION STATUS
		PROPULSION SYSTEM (Continued)		
42805E1 Western Gear Code 07196	Shaft - Flex - Acc. Drive	SCD P0021	50 Hr., Continuous 100% Rated Load, Qualification Run on Simulator Prior to Flight.	Qualified
42810E1 Western Gear Code 07196	Gear Box - Fan Assy.	SCD P0026	Same as Above.	Qualified
2F1-6-31857 Goodyear Code 89611	Tank Assy. - Fwd. Fuel	SCD P0027	Refer to Goodyear Qualification Test Report No. 300	Qualified
------- Liquidometer Code 36536	Gauging Syst. Fuel	SCD P0028	Refer to Liquidometer Qualification Test Report No. ER2092 IT-1172	Qualified
60-425 Hydroaire Code 81982	Pump - Fuel Booster	SCD P0029	Similar to 60-351 and 60-401 for Performance Refer to Test Report No. TP60-425	Qualified
V-14500-29 Valcor Code 98487	Valve - Air Shutoff	SCD P0030	Similar to V-14500 Valve Qualified per Aerotest Lab. Report No. 6Q426-7	Qualified
F-4512 Microporous Code 14034	Strainer Fuel	SCD P0035	Manufactured to Meet F4512 and SCD P0035-1	Qualified
A-62220 Vinson Code 91130	Valve - Check Hot Air	SCD P0036	Similar to Vinson P/N A40033 Ref Vinson Ltr 5-25-65	Qualified
------- Kirkhill Rubber Code 75345	Coupling - Fireproof	SCD P0039	Pending	
Space Flex Corp. Code 16367	Duct - Flex 4-1/2" Dia.	SCD P0040-1	Material per MIL-D-6441 Flame Resistant	
------- Space Flex Corp. Code 16367	Duct - Flex 4-1/2" Dia.	SCD P0040-3	Material per MIL-D-6441 Flame Resistant	
7G40 Custom Comp Code [illegible]	Switch - Pressure	SCD P0041-1	Refer to Statement of Similarity to Qualified Switch P/N 1G62 per Report No. 1G62-63	Qualified
6A[illegible]-3 Custom Comp Code [illegible]	Switch - Pressure	SCD P0041-2	Refer to Statement of Similarity to Qualified Switch P/N 8G21-01 per Report No. 8G21-01	Qualified
------- Arrowhead Prod Code [illegible]	Duct - Cooling Air	SCD P0042	Material per MIL-D-6441 Flame Resistant	Qualified
281314 Tavco Code [illegible]	Pressure Vessel Fire Ext.	SCD P0043 MIL-C-22284	Similar to Part Qualified to MIL-C-22284 Ref Tavco Test Report 63-121 BUWEPS Aug. 1964	Qualified

TABLE 16 (Continued)

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI-CATION	COMPLIANCE DATA	QUALIFICATION STATUS
PROPULSION SYSTEM (Continued)				
------- L. A. Std. Rubber Code 84914	Seal - Fire Resistant	SCD P0044	Pending	
5959 Conn. Hard Rubber Code 71643	Seal - Eng. Inlet Duct	SCD P0045	Similar to Convair 880 Seal	Qualified
------- H. I. Thompson Code 78741	Blanket - Insulating Div. Duct	SCD P0046	JM A-100 Insulation Covered with Cres Foil - Ref.	Qualified
------- H. I. Thompson Code 78741	Insulation Instl. - Pitch Fan Duct	SCD P0047	J-M A-100 Insulation Covered with Cres Foil	Qualified
2636095 Parker Aircraft Code 92003	Valve - Vent Float	SCD P0032	Similar to Parker No. 1119-577179 Ref. Qual. Test Report No. 1119-Q2404	Qualified
ST-504N U.S. Gauge Code 61349	Transducer - Oil Press.	MIL-T-25624	Ref: QPL 25624-5	Qualified
8TJ61GBA2 G. E. Code 97424	Transmitter - Fuel Flow	MIL-T-26298	Ref: QPL 26298 TSO-C44	Qualified
AV24B1108 Gen. Controls Code 73760	Valve - Fuel Shutoff		Pending	
416-50 Shaw Aero Dev. Code 99321	Cap - Fuel 3"	MIL-C-7244	Ref: QPL 7244-6	Qualified
429-2 Shaw Aero Dev. Code 99321	Cap - Fuel 2"	MIL-C-7244 Modified	Similar to 416-50 Except Size	Qualified
601700 Accessory Prod. Code 96124	Valve-Drain Fuel		Pending	
460-015-16 F. C. Wolfe Co. Code 83259	Gasket Gask-O-Seal	MS 27	Pending	
3605-16D Wiggins Code 79326	Coupling - Tube	MIL-C-25014	Used and Approved for Military and Commercial Aircraft by USAF and FAA	Qualified
BAR 6445 Southwest Prod. Code 81376	Bearing - Mono. Ball	Mfgr. Spec.	DOD Approved Source	Qualified
2 BREM-6A Southwest Prod. Code 81376	Rod End - Mono. Ball	Mfgr. Spec.	DOD Approved Source	Qualified

TABLE 16 (Continued)

MFG. PART NO. MFR. & CODE NO.	PART NAME	SPECIFI-CATION	COMPLIANCE DATA	QUALIFICATION STATUS
PROPULSION SYSTEM (Continued)				
2 BREM-5A Southwest Prod. Code 81376	Rod End	Mfgr. Spec.	DOD Approved Source	Qualified
2 BREM-4A Southwest Prod. Code 81376	Rod End	Mfgr. Spec.	DOD Approved Source	Qualified
215314 Tavco Code 99221	Valve - Shuttle Check	None	Tested and Used on a Commercial Aircraft	None
F-8300-102 Revere Corp. Code 50625	Switch - Float	USAF or AND Spec. No. WCLP1-3/ GRG/SC	F-8300 Type Switch Qualification Test Revere Report No. 113 dated 8 January 1954	Qualified
600-015-10 F.C. Wolfe Co. Code 83259	Stat-O-Seal	NAS 1598	Pending	Qualified
143P025-1	Tank - Fuel Aft	HAD	For Slosh and Vibration Test Ref. Goodyear Test Report No. 304	Qualified

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