REPORT NUMBER 125
AUGUST, 1963

PRELIMINARY RELIABILITY REPORT

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LIFT FAN FLIGHT RESEARCH AIRCRAFT PROGRAM

CONTRACT DA44-177-TC-715

PRELIMINARY RELIABILITY REPORT

REPORT NO. 125

AUGUST 1, 1963

GENERAL ELECTRIC COMPANY
ADVANCED ENGINE & TECHNOLOGY DEPARTMENT
Cincinnati 15, Ohio

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LIST OF SYMBOLS

Symbol	Definition
MTBF	Mean time between failures
Q	Unreliability - Probability of failure
R	Reliability - Probability of success
τ	Time interval
m	MTBF
$\lambda = \frac{1}{m}$	Failure rate - failures / unit time
T	Time of test experience
N	Number of failures experienced
%N	% RPM
W	Precession velocity - radians/sec.
Vp	Aircraft flight speed
Ksi	1000 pounds per square inch
σ	Standard deviation
rms	Root mean square
μ	Mean value of a parameter
psi	Pounds per square inch
t	Number of σ
σ_d	Standard deviation of the population of differences
Bs	Exit louver stagger angle
Bavg	Exit louver vector angle

PURPOSE

Clause No. 2 of Contract No. DA44-177-TC-715 requires the preparation and submission of a reliability and failure analyses report for the XV-5A Flight Research Vehicle.

This report presents in three parts the results of reliability programs of both the propulsion and aircraft systems. Part I describes the objectives, plans, and results of the overall XV-5A Reliability Program. Part II covers the propulsion system reliability program for Task I, Design & Engineering and Task II, Manufacture & Flight Worthiness Test. Part III covered the aircraft program for the early part of Task I, Design & Engineering.

Prior to beginning Task IV, Flight Test Program, a second report covering predicted quantitative XV-5A aircraft reliability will be submitted as part of the XV-5A Flight Worthiness Report.

REFERENCES

1. X353-5B Flight Worthiness Test Report, Volume 1, 2, & 3, January, 1963; General Electric Company.

2. X353-5 Fan Design Report, May, 1960; General Electric Company.

3. R59FPD793, R. V. Garvin, December, 1959; General Electric Company.

4. Airplane Detail Specification, April 1963; General Electric Company.

XV-5A RELIABILITY PROGRAM

A. SUMMARY

1. A mean time between catastrophic failures of 6700 hours has been established as an objective for the XV-5A system to provide a chance catastrophic failure probability of 1 per 10^5 fan powered takeoff and landing sequences.

2. No prediction of XV-5A reliability has been made; airframe designs and reliability analyses are incomplete.

3. Propulsion system reliability has been predicted from component design analyses. Results for the critical failure modes are tabulated in Tables 1 & 3.

4. A mathematical model of XV-5A reliability has been programmed for digital computation. This will permit rapid failure analyses and reliability predictions as component inputs become available.

5. Among design changes made to improve reliability are the adoption of a dual tandem hydraulic system, mechanically coupled diverter valves, positive overspeed protection for all fans, and use of warning instrumentation including fan unbalance, fan and aircraft structural heating, and excessive fan rpm.

B. INTRODUCTION

1. Background

The scope and objectives for the XV-5A reliability program were mutually established by the General Electric and Ryan Companies and are as follows:

2. Scope

The XV-5A research program is limited to two aircraft and a fifty hour flight test program. It is therefore impractical to conduct a complete reliability program wherein thousands of hours of component sub-system, and system tests and design improvements are accumulated. Scope of the XV-5A reliability effort covers Design, Reliability Analysis, System Testing, Reliability Prediction and Measurement.

The ultimate responsibility for achieving reliable components rests with the individual designers who must obtain adequate safety margins. Reliability analysis includes determining the required reliability of a system, identifying the critical components of a system and their significant methods of failure, determining the probability of failure and the need for redesign, re-rating or redundancy. Testing includes a very limited amount of component tests; major emphasis is on system testing to substantiate design analyses and to determine incompatibilities of components with each other or with operating techniques. Reliability prediction is the mathematical summation of the various component and system analyses in terms that permit prediction of success or failure for discrete periods of operating time. Measurement is simply the bookkeeping of simulator and flight test experience and is the proof check of all other reliability work.

B. 3. Objectives

A quantitative reliability objective was established early in the XV-5A program. Although research or prototype aircraft experience indicates a high probability of catastrophic failure, the XV-5A objective was set two to three orders in magnitude better. The objective probability of catastrophic failure is 1 per 10^5 take-off and landing sequences.

In establishing objectives, MIL-R-27542 (Reliability Program Requirements For Aerospace Systems, Subsystems and Equipment) was reviewed. Certain applicable portions of the specification's requirements were adopted as guiding principles of the XV-5A reliability program. These are repeated below:

* The reliability program shall recognize the concept of inherent reliability of design, i.e., the inherent reliability is established by the basic design and can be improved only by design changes.

* Improvement of reliability is best achieved in the early phases of development and the testing program.

* A factor in system reliability is human reliability which includes the extent to which the equipment has been engineered to minimize human error in the manufacture, test, operation, and maintenance of the system.

* Reliability must be a major factor in planning, management and engineering.

C. DISCUSSION

1. Reliability Defined

As used in the XV-5A program, reliability means the probability that the aircraft performs its required function under the anticipated flight conditions without failure for a specified period of time.

2. Emphasis on Take-Off & Landing Sequence

As stated in the objectives, the purpose of this program is to achieve a specific level of basic flight safety during the critical portion of XV-5A flight, i.e., fan-powered take-off and landing. If that objective can be achieved, then the reliability of the research "missions" and conventional flight in general will be more than adequate. The reasoning is as follows: The propulsion system and aircraft systems are the most complicated in fan-powered mode. Many potential failures are critical only in the fan mode. In take-off, for example, once the fan mode operation ends with conversion to conventional flight, the propulsion system and aircraft systems are significantly reduced in complexity to the point where reliability is determined primarily by the aircraft structure, conventional controls, and turbojet propulsion.

3. Catastrophic Failure

The type of failure described in the objectives refers to catastrophic events such as:

* Pilot fatality or serious injury due to design deficiency or malfunction.

* Total loss of an aircraft due to fire or other malfunction resulting in pilot ejection.

* Irreparable damage due to malfunction.

C. 3. Basic causes of catastrophic failure are as follows:

a. Pilot error

b. Aircraft structural failure

c. Aircraft control system failure including hydraulic and electrical

d. Propulsion system destructive mechanical failure

e. Uncontrollable fire

Pilot error is controllable in design by the extent of authority available to make mistakes. The quantity of simultaneous or sequential pilot functions determines error probability more than the choice of pilots. The four remaining major causes of catastrophic events are essentially independent of each other. Therefore reliability, R_T, of the XV-5A is Rb x Rc x Rd x Re. Conversely, the unreliability is approximately the sum of failure probabilities - Qb + Qc + Qd + Qe. Therefore, to achieve a catastrophic failure probability no worse than 1 per 10^5 take-off and landing sequences, each of the four major system including the poorest should have catastrophic failure probabilities better than 1 per 10^5 take-off and landing sequence.

4. Mean Time Between Failure Required

Assuming that the minimum practical duration of a fan-powered take-off and landing sequence is four minutes, the minimum mean time between catastrophic failures can be approximated:

Q_T = .00001 or R_T = .99999 for 4 minute period

$R = e^{\frac{\tau}{m}}$ where m = MTBF and τ = the time interval involved.

$$R_T \cong \left(1 - \frac{\tau}{m}\right) \text{ where } \frac{\tau}{m} << 1$$

$$M = \frac{\tau}{1 - R_T} = \frac{\tau}{Q_T} = \frac{.067 \text{ hrs.}}{.00001} = 6700 \text{ hours}$$

C. 4. Thus aggregate mean time between catastrophic failures of the XV-5A should be not less than 6700 hours. Since the failure rate, $\lambda = \frac{1}{m}$

$$\lambda_T = \frac{1}{6700} = 1.5 \times 10^{-4} \text{ failures per hour.}$$

As previously described, the XV-5A contains four major systems - each with its attendant λ

Therefore $R_T = e^{\lambda_b T} \times e^{\lambda_c T} \times e^{\lambda_d T} \times e^{\lambda_e T}$

$$= e^{T(\lambda_b + \lambda_c + \lambda_d + \lambda_e)}$$

If arbitrarily

$$\lambda_b = \lambda_c = \lambda_d = \lambda_e$$

then, for example, the propulsion system failure rate,

$$\lambda_d = \frac{1.5 \times 10^{-4}}{4} = 3.75 \times 10^{-5} \text{ failures per hour.}$$

It should be remembered that these failure rates are the chance unreliabilities which would produce catastrophy - not just a mission abort, reduced performance, or nuisance.

5. Further Breakdown of Aircraft Systems

In Part III of this report, the XV-5A is shown as further divided into ten systems. These are the significant separate systems which are then sub-divided further, down to the component level. A digital computer program has been developed as a mathematical model of XV-5A reliability to aid in evaluating effects of component and system reliability changes and to determine component reliabilities required to provide a given system or aircraft reliability.

At this writing, there is insufficient component data to predict an overall XV-5A reliability or catastrophic failure rate; this will come as designs are completed.

C. 6. Propulsion System Reliability

The propulsion system including engines, fans, and diverter valves has been evaluated from two approaches: test experience and reliability analyses of component designs. Except for the J-85 turbojet engine, test experience is too limited to use as a measure of catastrophic chance failure rate. Therefore the predicted rates were obtained from analyses of the critical potential failures using a common reliability evaluation technique of determining the safety margins between strength and stress. The results of these analyses are discussed in Part II.

Propulsion system reliability efforts have been after-the-fact for the most part. This situation is due to the fact that the principal XV-5A research program objective is to demonstrate in-flight an existing X353-5 propulsion system. Design changes have been held to a minimum; most of these were undertaken to correct known deficiencies and therefore improve reliability, or to facilitate installation in the airframe.

In contrast, the airframe is an entirely new design; reliability effort has therefore been concentrated before-the-fact.

PROPULSION SYSTEM RELIABILITY X353-5B & X376

A. SUMMARY

1. Fans & Diverter Valve

A technique of failure rate prediction described in Ref. 2, has been applied to critical components of the fans and diverter valves. While the method does not provide exact probabilities of component failure, it is the best available approach to estimating such information from the combination of design analyses and limited test experience. Results are summarized in Table 3, page II 18-A.

2. J85 Turbojet

Extensive factory and flight test experience has provided failure rate data. Allowing for the simpler non-reheat XV-5A J85 configuration, predicted performance is summarized in Table 1, page II-8.

3. Complete Propulsion System Reliability

Analysis of complete installed propulsion system reliability is incomplete at this writing and requires further work. Early results are discussed in Part III.

4. Chance Failure - vs - Wearout Failures

The predicted probabilities of chance or random failures should not be confused with wearout failures. Component deterioration due to erosion, for example, must be continuously monitored by a vigorous routine of post-flight and periodic inspections. These routines will be defined in the maintenance and operating instructions.

B. INTRODUCTION

1. Description of Propulsion System

The propulsion system described in this part of the report includes these components:

* 2 J-85-GE5 turbojet engines with controls and accessories but modified for dry operation only.

* 2 diverter valves including mechanical coupling linkage mechanisms, and hydraulic actuator but not the hydraulic controls.

* 2 X353-5B lift fans including mounts and exit louvers.

* 1 X376 pitch fan

2. Operating Requirements

In operational use, the propulsion system will be converted between the fan-power mode and turbojet power mode during each vertical take-off and landing. Certain sequences and precautions must be maintained during conversions to assure safe flight. These are described below.

Fan power-to-turbojet power conversion - The first action responding to pilot command must be the switching of diverter valves to remove gas power from the fans. This produces an immediate buildup of turbojet thrust and allows the three fans to coast down toward windmilling rpm. After gas power is removed from the fans, a time interval of approximately two seconds must be allowed before fan inlets may be closed-off. The interval provides a rundown to approximately 40% rpm at which time the inlets may be closed-off by the wing fan cover doors and the nose fan inlet louvers.

As soon as inlets are closed-off, the exit variable geometry may be closed to complete the propulsion system conversion sequence.

B. 2. Turbojet power to fan power conversion - The action of variable geometry in this sequence is essentially the reverse of the procedure described previously. Initial action responding to pilot's conversion command must be the opening of all fan inlet and exit geometry. Only after the inlets and exits attain open positions can diverter valves be switched to deliver gas power to the fans.

Part III of this report describes a conversion interlock system which, among other functions, positively assures the proper sequence of conversion variable geometry in response to a single pilot operated mode selector.

Fan overspeed tendency - Four external factors affect X353-5B fan rpm at constant throttle setting.

* Turbojet ram recovery - increased gas power with forward flight speed.
* Fan crossflow effect - unloading at increased flight speed.
* Exit louver throttling - unloading with reduced discharge area from vectoring or staggering louvers.
* Fan inlet stall - unloading at full wing stall.

Tests have shown the first three causes of increased rpm to be continuous functions and therefore predictable and controllable by the pilot. For this reason, the fan tachometers include a sensor and indicator to visually warn the pilot of high rpm operation. The wing stall causes a 10% rpm discontinuity. Because the resultant rpm can quickly exceed continuous operating limits, an automatic partial power cut-back has been incorporated into the J85 control mechanism. Authority to restore full power or override the power cut is provided to the pilot.

B. 3. Fans As Control Devices

At flight speeds where conventional surface controls are ineffective the three fans provide maneuver and trim moments for control of aircraft attitude. These functions along with longitudinal and vertical acceleration of the aircraft are obtained from combinations of fan thrust spoiling and vectoring. The controlling devices are located in exhaust streams of the three fans. Reliability aspects of fan exit louvers mechanisms are covered in this section of the report. Part III includes X376 fan exit geometry and control of all fan variable geometry.

C. DISCUSSION

1. Fan Test Experience

At this writing, X353-5 fans have accumulated over 400 hours of operating experience in both static and wind tunnel tests. X376 fans have accumulated over 75 hours of static operation. Although X353-5 fans have sustained several occurrences of minor damage and one instance of significant damage,[1] no failures have occurred which would have caused catastrophic consequences in XV-5A flight.

2. Fan Reliability From Test Experience

An attempt to apply a statistical evaluation technique to the available test experience proved to be meaningless due to the relatively small accumulated time as compared with objectives for mean time between catastrophic failure discussed in Part I. The results are included only to show that probably never during the XV-5A research program will experience be sufficient to permit meaningful statistical evaluation of reliability.

The procedure used was obtained from Reference 3. It provides probable range of MTBF for a chosen confidence level based on total test time and the number of failures occurring within that time.

First use only that test time at greater than 75% rpm as being significant in XV-5A operation.

$$T_T = 400+ \text{ hours}$$

$$T_{N > 75\%} = 70 \text{ hours} = T$$

---

[1] Refer to Reference 2 for discussion of Flight Worthiness Test results.

C. 2. Let total number of failures, N = actual number + 1

$$\therefore N = 0 + 1 = 1$$

then $\frac{T}{N} = \frac{70}{1} = 70$

From Reference 2 table of probability for $\frac{T}{N} = 70$, for 80% confidence,

find $\frac{T}{N}_{upper} = 2.33$ & $\frac{T}{N}_{lower} = 0.1$

Then (MTBF) upper $= \frac{70}{0.1} = 700$ hours

and (MTBF) lower $= \frac{70}{2.33} = 30$ hours

These results say with 80% confidence that for 70 hours experience without failure, the probable MTBF of similar components is between 30 and 700 hours. Thus the futility of statistics is seen when the experience is limited.

The foregoing discussion is not intended to belittle the practice of development and qualification testing. Later in this section it will be shown that the measurement of performance and design parameters for substantiation or modification is a requirement in the analytical determination of reliability.

3. J85 Test Experience

Extensive J85 test experience is available from thousands of hours of factory tests, flight tests, and operational use in T 38 aircraft. Although this experience is based on a reheat-configured J85-5, simple bookkeeping provided the dry engine reliability data needed.

C. 3. J85 failure significance - The type of failures investigated included total and partial power loss. Consequence of these failures in conventional turbojet mode XV-5A flight is not catastrophic since single engine performance is adequate. Consequence of power failure in fan-powered mode depends upon the flight condition. The engine-out recovery curve in Reference 4 illustrates this point. At XV-5A design gross weight (9200#) and hot day conditions, the maximum hover lift / weight ratio of 1.15 decreases to 0.69 in event of a single J85 total power loss. Above a few feet altitude, there is an envelope of altitude - vs - flight speed in which recovery cannot be effected by either of the following engine-out procedures:

a) Pitch over and accelerate during descent until wing lift plus .69 (G.W.) fan lift is sufficient to effect a flare of 6 ft./sec. or less sink rate at touchdown followed by a short roll-out.

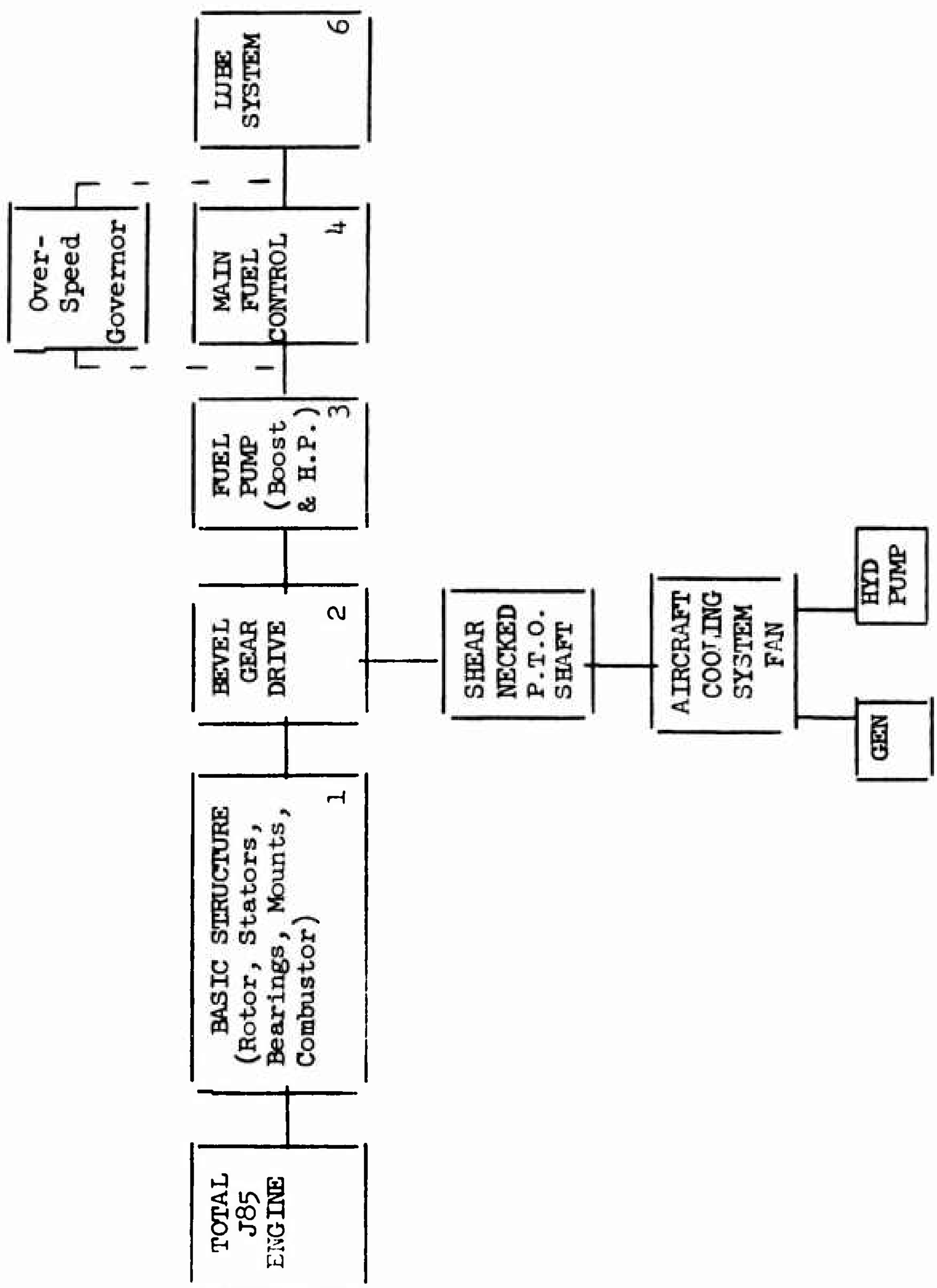
b) Pitch over and vector fans for accelerating thrust until sufficient altitude is traded for acceleration to a flight speed at which conversion to single engine wing-supported flight can be made.

Further studies of J85 failure probabilities have been performed with results shown in Table 1.

J85 reliability diagram - Figure 1 shows the serial arrangement of basic J85 components. Also shown is the power take off for aircraft accessories.

FIGURE 1

J85 RELIABILITY DIAGRAM



C. 3. Predicted J85 MTBF - Table 1 below summarizes the random or chance failures expected.

TABLE 1

J85 Reliability

Type of Failure	MTBF
a. Unscheduled maintenance or premature removal	100 hours
b. 90% < Power available < 100%	129
c. 75% < Power available < 90%	165
d. Zero Power available	1000
e. Internal destruction	2235

4. Fan Overspeed Control Reliability

In Paragraph II B 2 of this report, the causes and prevention of fan overspeed operation were described. Use of protective devices generally introduces additional failure mechanisms to a system. The power cutback device is no exception. However, careful design, restricted power authority, and limited range of operation have been exploited to minimize consequences of overspeed protection failures.

Authority of the overspeed limiter has been restricted to 30% of J85 power; i.e., it cannot reduce power below 70% of maximum which corresponds to 95% J85 rpm. The power authority is equivalent to roughly 10% fan rpm or 20% lift. Therefore an unnecessary power cutback would reduce lift to 80% of maximum. The pilot has authority to restore full power by use of a thumb switch in the collective lift control stick. Full power can be restored in just under one second. Since fan overspeed is improbable at hover or low speed and since the consequence of unjustified power cut at these conditions is

C. 4. critical, the power cutback function is dis-armed at exit louver angles less than 20 degrees. It is armed between 20 and 50 degrees which corresponds to the high transition speed range where the overspeed is likely to occur. Beyond 50 degrees, it is again disarmed so that inadvertant power cuts will not occur in conventional turbojet-powered flight.

In addition, the overspeed system is designed for random MTBF as shown in Table 2.

TABLE 2

Fan Overspeed Control Reliability

Failure Mode	MTBF
a. Power cut at less than cut-in rpm	10^5 hours
b. No power cut at cut-in rpm or greater	10^3 hours
c. Loss of rpm indication in cockpit	10^3 hours
d. Loss of high rpm warning function	10^3 hours

5. Design Analysis of Fan and Diverter Valve Reliability

Summary of analytical technique - Reference 3 defines a design procedure which provides estimates of chance failure probabilities for mechanical components. The technique can be summarized as follows:

a) Determine the important loading conditions in fan-powered XV-5A flight in terms defining mechanical design parameters.

b) Delineate those critical components of each assembly whose failure can cause significant consequences in XV-5A flight.

C. 5.

c) Define the component failure mechanisms; e.g., stress rupture, fatigue, deflection, yield, etc.

d) Using the designers' calculated or measured stresses, define normal distributions of stress variation.

e) Using average material property data define probable normal distributions of strength variation.

f) From statistical mathematics, calculate the probabilities of events in which stress exceeds strength.

g) A summation of these event frequencies then yields the chance failure rate of the assembly.

Loading conditions - Three major factors determine stresses in propulsion system components. These are: (1) power level setting, (2) aircraft forward flight speed, and (3) precession velocity. Transient power changes, aircraft angle of attack, landings, and other factors also affect loading but to a lesser extent than those above. Drawing from the X353-5B Propulsion System Specification, four XV-5A flight conditions were used to define loads:

a) Hover at maximum power and essentially zero precession velocity

b) Hover at maximum power and 0.8 radian / second precession velocity.

c) Maximum fan powered flight speed and essentially zero precession velocity.

d) Maximum fan powered flight speed and 0.8 radian / second precession velocity.

These four conditions were established as an estimate of maximum power operation at hover and conversion speed, with and without stability augmentation.

C. 5. Critical components - reliability diagram - In each propulsion system component, there are many critical components. In the example which follows, i.e. an X353-5 lift fan, the critical components defined are those which can produce catastrophic consequences in event of failure. These components are shown in their proper relation by Figure 2. Obviously test experience influences the definition of critical components. Some of the more significant test experience includes:

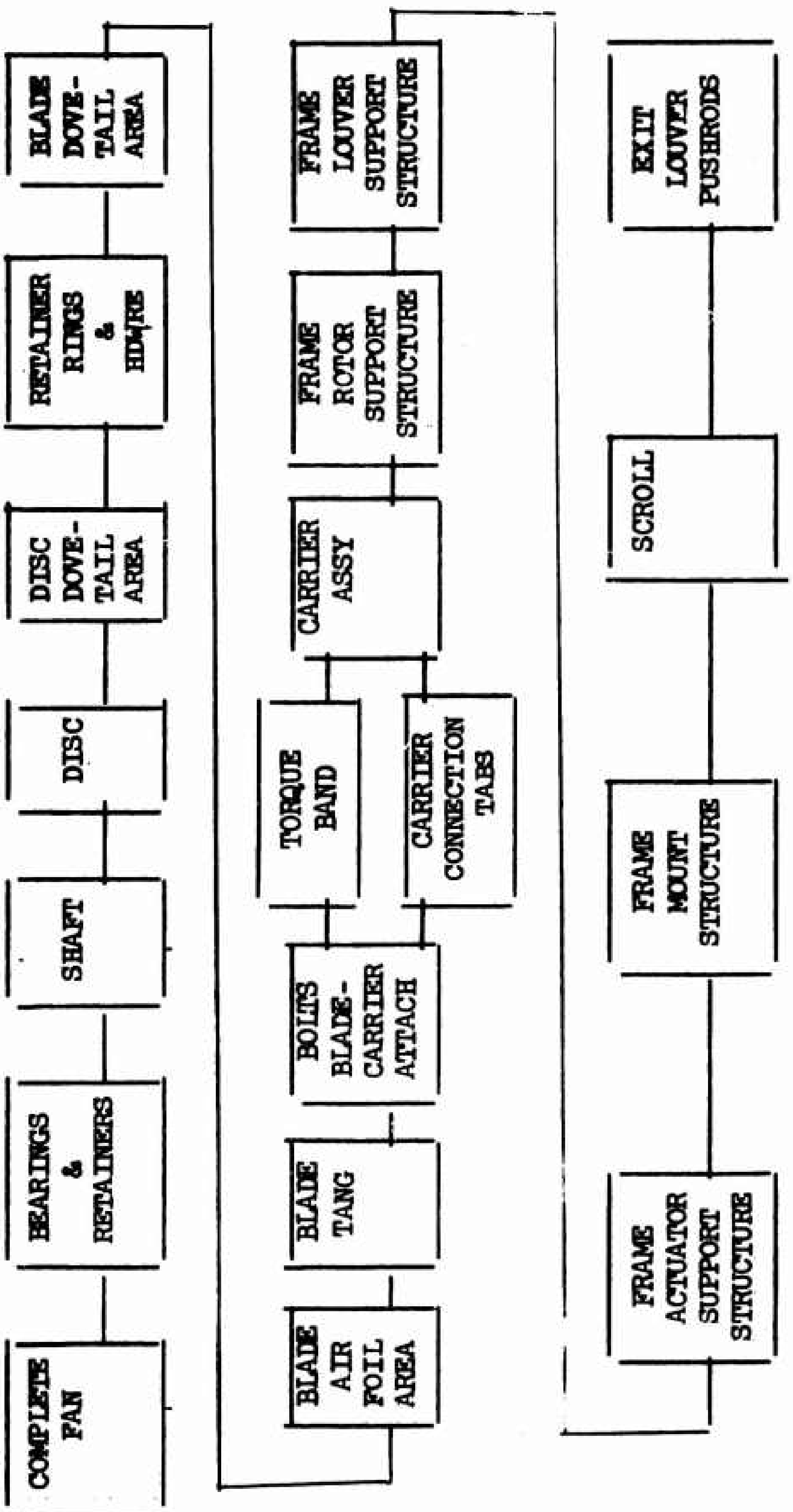
* measurement of rotating and stationary part stresses
* measurement of relative rotor - stator deflection during precession
* proof of redundancy in torque transmission system
* identification of wear tendencies
* demonstrating non-catastrophic consequences of turbine airfoil mechanical damage from upstream objects.

Component failure mechanisms - Each component has several mechanisms of failure. Typical examples are yield, deflection, fatigue, and rupture. Not all failure mechanisms are significant in any one component. Usually one mechanism becomes the controlling factor in a particular component design. Variation in load condition can shift the relative significance of the several failure mechanisms. Thus for each load condition a component must be analyzed for failure rate of each significant failure mechanism to provide a total failure rate for that component.

Stress deviation - The process of estimating a standard deviation of stress involves review of the designers' procedure to place values upon the variables affecting the design. Typical of these variables are the uncertainty of environment and stress analysis method, whether or not substantiating tests have been made, variation in weight manufacturing tolerances or

FIGURE 2

X353-5 LIFT FAN CRITICAL COMPONENT RELIABILITY DIAGRAM



C. 5. deviations, corrosion possibility and effect, practicality of inspection and effects of stalls and other transients. These factors then determine the estimated distribution of stress around the mean calculated or measured stress; assuming normal distribution, a standard deviation can be calculated.

Strength deviation - Two methods of obtaining a standard deviation of strength are available. In the first case strength data for some materials have been statistically evaluated in terms of mean strength and standard deviations of strength by the Large Jet Engine Department Materials Operation. In the second case, strength data also obtained from the Materials Operation, is presented as average strength multiplied by a deviation factor. The value of the expression (1 - dev. factor) is estimated as three standard deviations of strength based on materials evaluation experience. Both methods of defining material strength are described further in Appendix 1.

Probability of stress exceeding strength - Figure 3 graphically illustrates the relationship between mean stress and mean strength and the distributions of these parameters. The shaded area of overlap between distributions represents failure or stress > strength. Failure rate is then determined by calculating the relative area of overlap. The statistical mechanics will be illustrated in a sample calculation.

First example of analysis - The fan rotor disc is used as an example in this procedure. Design data for this component were obtained from Reference 2. First the disc is analyzed at the steady hover flight condition:

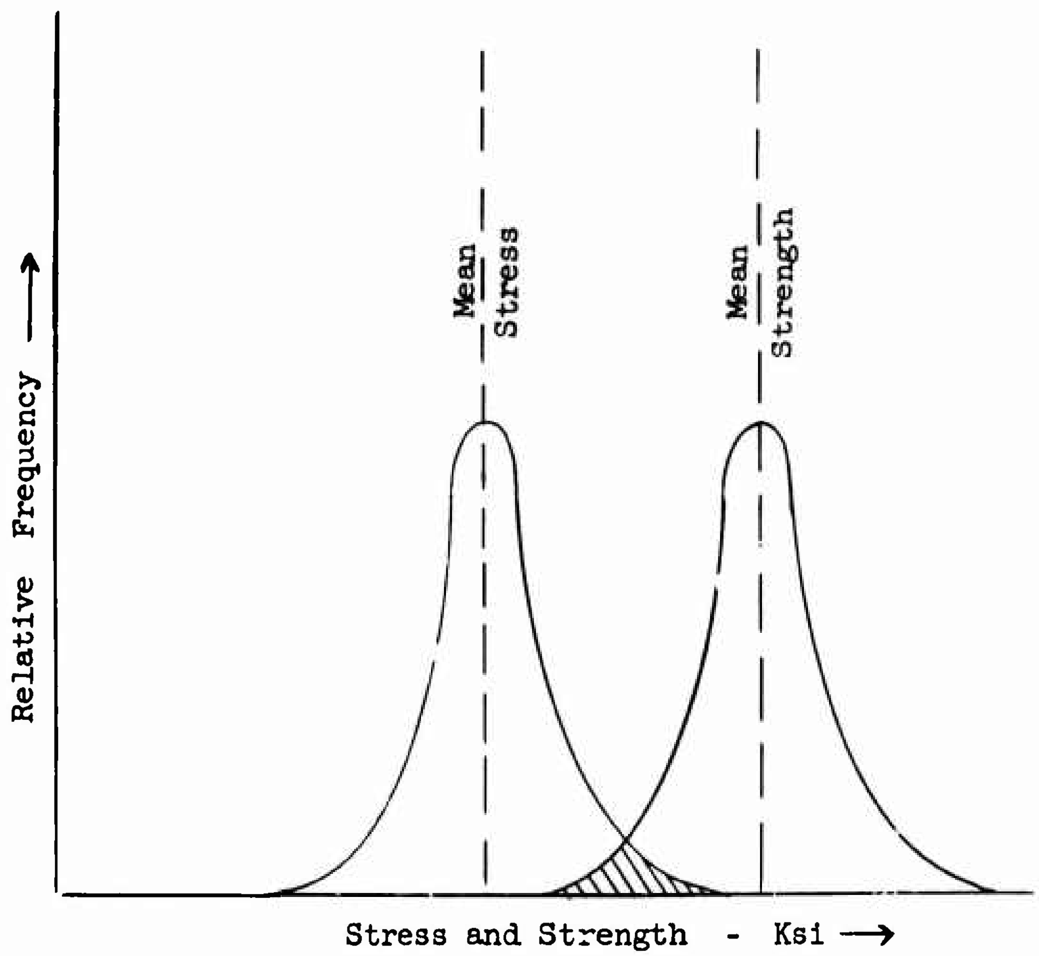
$\%N = 103\%$ (2719 rpm)

$W = 0$ radians / second

$Vp = 0$

FIGURE 3

STRESS & STRENGTH DISTRIBUTION



C. 5. Design data shows a steady centrifugal stress to be 63 Ksi maximum at 105% N and W = 0 = Vp.

Scaling to 103% rpm:

$$\text{Max stress} = 63 \text{ Ksi} \left(\frac{103}{105}\right)^2 = 60.6 \text{ Ksi}$$

Max temperature = 300°F

Assuming the .2% yield strength approximates the stress at which serious rotor - stator interference could result, solve for the probability of stress > strength.

From material data, .2% yield = 135 Ksi

Estimate standard deviation of stress,

$$\sigma_{stress} = \sqrt{\sum \left(\frac{\text{variations}}{3}\right)^2}$$

Steady stress variation		3σ variation	1σ variation
Load	* Weight supported	6%	2%
	* Transient overspeed	10	3
	* Acceleration torque	6	2
	* Impact-brg. free-play	10	3
Mfg	* Dimensional tolerance	6	2
	* Corrosion	5	2
	* Forging fault	10	3
	* Nicks and dents sensitivity	10	3

% (σ stress) rms

$$= \sqrt{(2)^2 + (3)^2 + (2)^2 + (3)^2 + (2)^2 + (2)^2 + (3)^2 + (3)^2}$$

$$= \sqrt{52} = 7.2\%$$

C. 5. and $\sigma_{stress} = 7.2\% \ (\mu \text{ stress})$

$$= .072\ (60.6 \text{ Ksi}) = 4360 \text{ psi}$$

From material data, $3\sigma_{strength} = 15\% \ (\mu \text{ strength})$

or, $\sigma \text{ strength} = 5\% \ (\mu \text{ strength})$

$$= .05\ (135 \text{ Ksi}) = 6750 \text{ psi}$$

Since extensive tests have shown fan hub area temperatures to be less than 300°F plus the fact that the XV-5A cockpit will readout bearing temperatures, assume that σ strength due to σ temperature is zero.

Now in effect we subtract the distribution of stress from the distribution of strength which yields a normal distribution of the population of differences as shown in Figure 4.

Then $$\sigma_d = \sqrt{(\sigma \text{ stress})^2 + (\sigma \text{ strength})^2}$$

$$= \sqrt{(4360)^2 + (6750)^2}$$

$$= 8000 \text{ psi}$$

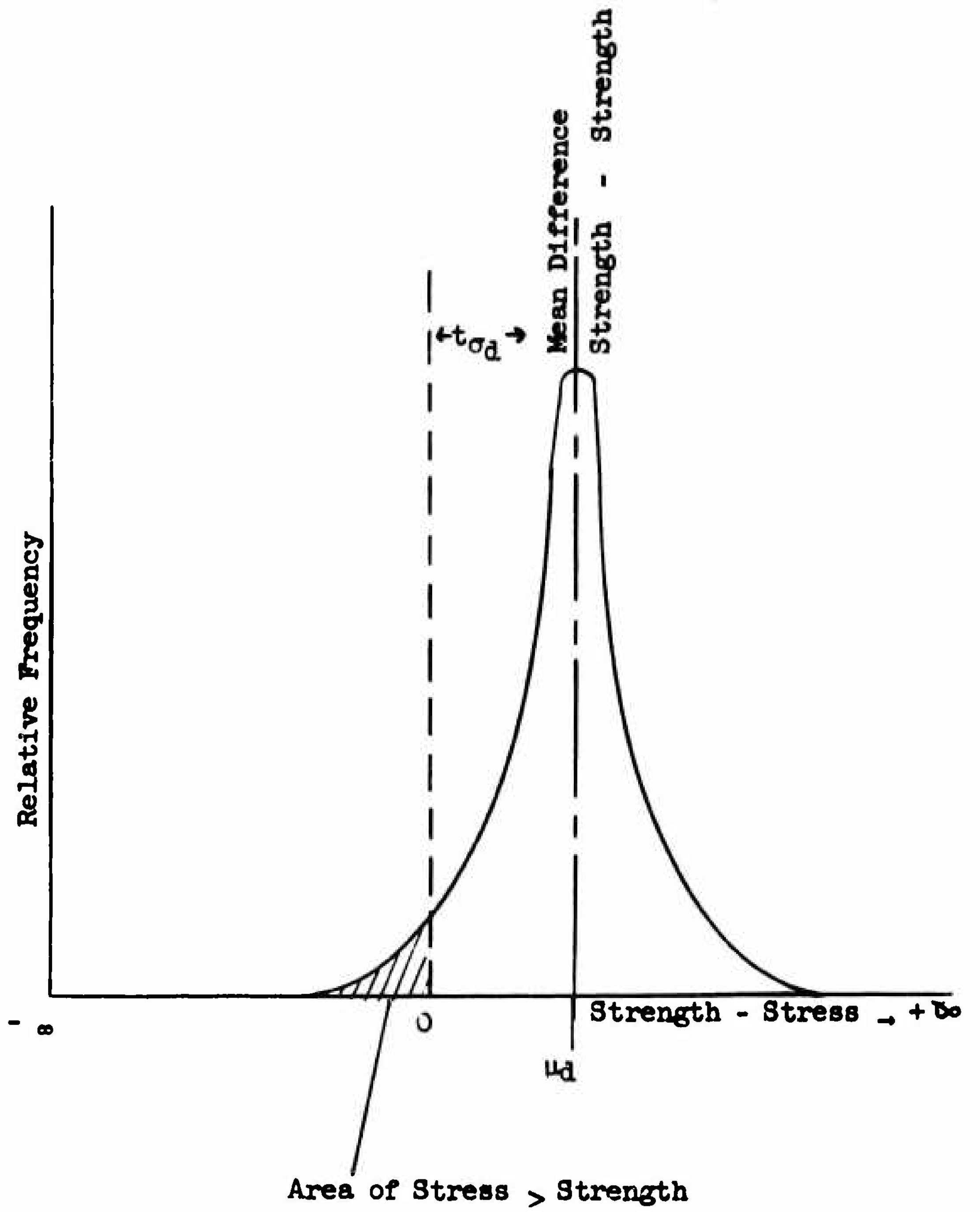
and $\mu \text{ strength} - \mu \text{ stress} = t\sigma_d$

or $$t = \frac{135 \text{ Ksi} - 60.6 \text{ Ksi}}{8000 \text{ psi}} = 9.3 \text{ standard deviations.}$$

From a table of normal distribution area versus standard deviations removed from the mean, the probability or failure rate of the disc yield failure mechanism is $\sim 1/10^{10}$. The rupture failure mechanism is even lower in rate. For a system with the simplicity of the X353-5, failure rates of $1/10^{10}$ can be neglected, i.e. assumed to be zero.

FIGURE 4

STRENGTH - MINUS - STRESS DISTRIBUTION



t = number of standard deviations from mean difference, μ_d

σ_d = standard deviation of the population of differences (strength - stress)

C. 5. The above calculations simply show that the fan disc is not designed to the steady centrifugal stress criteria of steady hovering flight. It is of course designed to the more stringent requirements of maneuver at conversionflight speed where fatigue and deflection become the significant failure mechanisms.

Second example of analysis - Consider now the fatigue mechanism failure probability of the same disc under max power operation at combined high forward flight speed and 0.8 radian / second precession.

From Reference 2 at 120 knots, 105% N and 2 rad / sec:

Steady centrifugal stress = 63 Ksi

Alternating gyro stress = 46.2 Ksi

Crossflow alternating stress = 4850 psi

Use of 0.8 rad / sec as max maneuver rate amounts to a de-rating of the fan for increased safety margins.

Converting loads:

Max steady stress = $63 \left(\frac{1.03}{1.05}\right)^2$ = 60.6 Ksi

Max crossflow alt. stress = 4850 psi

Max gyro alt.. stress = $46.2 \times \left(\frac{0.8}{2}\right)$ = 18.5 Ksi

Design of parts subjected to combined steady and alternating stresses involves use of a design tool known as a Goodman Diagram. Figure 5 is a typical Goodman presentation of allowable alternating stress - vs - steady stress for SAE 4340 alloy hardened to the condition used in disc manufacture. The curve shown represents average material combined load strength for 10^8 cycles of useful alternating stress life; however, the 10^8 cycle curve is essentially coincident with a curve of unlimited life.

Estimated failure rate of a part subjected to combined stresses is obtained by an iterative process of determining the amount of standard deviations of mean combined stress from the mean strength curve shown in Figure 5.

C. 5. Continuing the solution for a revised curve to determine amount of standard deviations from max. load point shown in Figure 5, assume first that alternating stress is zero and find ordinate intercept using procedure previously described:

μ strength = 147 Ksi

μ steady stress = 60.6 Ksi

Let t = 8 standard deviations as first estimate

From steady stress example, σ stress = 7.2% μ stress

From material data, σ strength = 1/3 (20% x μ strength)
= 9800 psi

Estimate σ_d = 6500 psi and iterate:

μ stress = μ strength - $t\sigma_d$
= 147 Ksi - 8(6500)
= 95 Ksi

Now σ stress = 7.2% (μ stress)
= .072 (95 Ksi) = 6840 psi

Iterating for σ_d:

$$\sigma_d = \sqrt{\sigma\ \text{stress}^2 + \sigma\ \text{strength}^2}$$
$$= \sqrt{(6840)^2 + (9800)^2}$$
$$= 11,900 \text{ psi}$$

This does not check assumed σ d of 6500 psi.

Re-estimate σ_d = 10,600 psi

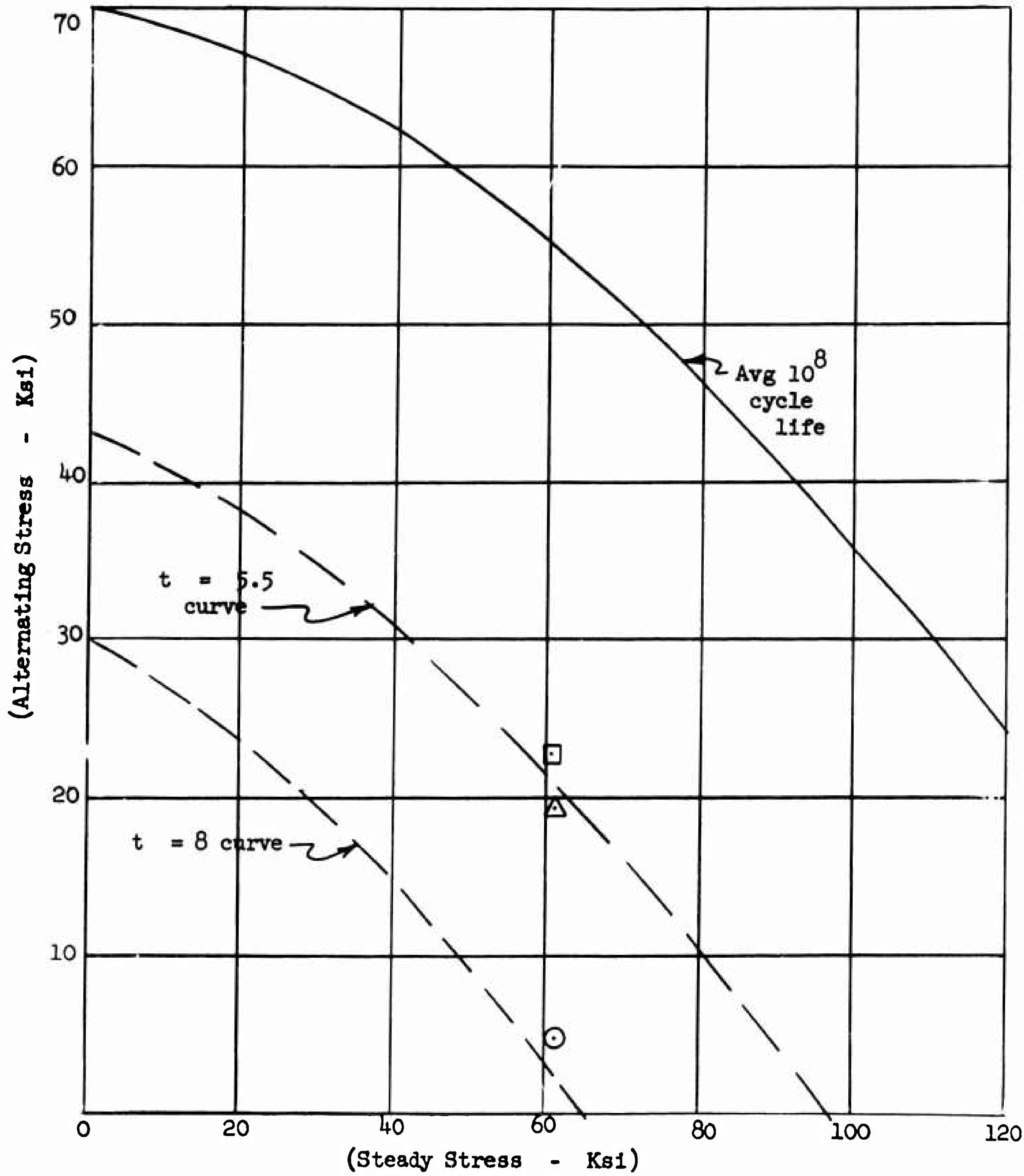
Then μ stress = 147 Ksi - 8(10,600)
= 62.2 Ksi

and σ stress = .072 (62.2) = 4480 psi

$$\therefore \sigma_d = \sqrt{(4480)^2 + (9800)^2}$$
$$= 10,780$$

FIGURE 5

GOODMAN DIAGRAM OF X353-5 DISC MATERIAL



⊙ Combined steady & crossflow stress
△ Combined steady & gyro load stress
⊡ Combined steady and in phase max. alt. stress

C. 5. This is a close agreement; therefore, the zero alternating stress intercept for μstress is 62.2 Ksi as shown plotted on Figure 5.

Now solve for the alternating stress intercept assuming zero steady stress.

Again let t = 8 standard deviations

μstrength = 70 Ksi

σstrength = 1/3 (20% x μstrength) = 4700 psi

σstress = 7.2% μstress

Let σd = 5200 psi and iterate for μstress at t = 8;

μstress = 70 Ksi - 8(5200) = 28.4 Ksi

σstress = .072(28.4) = 2040 psi

$\sigma d = \sqrt{(4660)^2 + (2040)^2}$

= 5095 psi which is below estimate

Additional iteration yields μstress = 29.2 Ksi which is plotted on Figure 5

By inspection of Figure 5, it is apparent that the original estimate of 8 standard deviations was too conservative since a curve approximation between the t = 8 intercepts does not enclose the combined stress data points.

The foregoing procedure is repeated for reduced values of t until the curve approximation encloses the max. combined load data point. Successive iteration yields a value of t = 5.5 or a probability frequency of events where stress exceeds strength for 10^8 cycle life of 1.9×10^{-8}

C. 5. Results of analyses - Table 3 summarizes the results of component failure analyses such as just described. Also included are the more important non-catstrophic types of failures.

TABLE 3 - FAILURE ANALYSIS SUMMARY

COMPONENT	FAILURE MECHANISM	PROBABLE FAILURE RESULTS	PROBABLE CONSEQUENCES: BASIC FLIGHT SAFETY	PROBABLE CONSEQUENCES: RESEARCH MISSION COMPLETED?	FAILURE RATE PER HOUR: 1. $V_p = 0$, Max. Pwr., Autostab, $\beta_s = 30°$, $\beta_{avg} = \pm 10°$	2. $V_p = 0$, Max. Pwr., No Autostab, $\beta_s = 30°$, $\beta_{avg} = \pm 10°$	3. $V_p = 120$ K, 103% $N_{L.F}$, No Autostab, $\beta_s = 10°$, $\beta_{avg} = 45°$	4. $V_p = 120$ K, 103% $N_{L.F}$, Autostab, $\beta_s = 10°$, $\beta_{avg} = 45°$
A. LIFT FAN	1. Rotor Burst and Support Structure Failure	• Loss of lift • Structural Damage • Uncontrollable Roll	•Catastrophic @ $V_p = 0$ •Major A/C Damage at V_p = max.	No	9×10^{-7}	1×10^{-5}	5×10^{-5}	5×10^{-6}
	2. Exit louver pushrod buckle or rupture	•Sudden Roll & Yaw Moments •Reduced Roll & Yaw Control Effectiveness	•Recovery Possible But Not Assured	No	1×10^{-5}	→	→	→
	3. Louver Actuator Mount Rupture	•Same as A.2.	Same as A.2.	No	1×10^{-6}	→	→	→
	4. Scroll Rupture	•Reduced Lift •Sudden Roll Moment •Fire or O/H Structure	Same as A.2.	No	1×10^{-5}	→	→	→
	5. Loss of Louver Supports or Torque Mechanism	•Reduced Control Effectiveness - Roll/Yaw/Altitude/Thrust	None	Yes 75%	1×10^{-8}	→	→	→
	6. Fan Mount Rupture	•Loss of Fan Positioning •Loss of Louver Control	•Catastrophic @ $V_p = 0$ •Major Damage or Catastrophe @ V_p = max.	No	1×10^{-8}	1×10^{-7}	→	1×10^{-8}
	7. All Other Failures	•Unscheduled Adjustment, Repair or Removal Maintenance	None	Yes 90%	1×10^{-1}	→	→	→
B. PITCH FAN	1. Rotor Burst and Support Structure Failure	•Uncontrollable Pitch •Cockpit Structural Damage	•Catastrophic @ $V_p = 0$ •Catastrophic at max V_p w/o cockpit guard	No	1×10^{-7}	1×10^{-6}	5×10^{-6}	5×10^{-7}
	2. Scroll Rupture	•Reduced Pitch Effect. •Reduced Lift •Sudden Pitch Moment •Fire or O/H Structure	•Recovery Probable	No	5×10^{-6}	→	→	→
	3. Mount Rupture	•Loss of Fan Positioning •Possible Loss of Pitch Thrust	•Possibly Catastrophic	No	5×10^{-7}	5×10^{-6}	5×10^{-6}	5×10^{-7}
	4. All Other Failures	•Unscheduled Maint.	None	Yes-90%	5×10^{-2}	→	→	→
C. DIVERTER VALVE	1. Body Rupture	•Reduced Lift •Fire or O/H Structure	•Recovery Probable	No	1×10^{-6}	→	→	M_p 0.7 S.L. 5 X 10[illegible]
	2. Actuator Linkage Rupture	•Doors Hold Position Until Conversion Attempted.	None - May cause single J85 failure at attempted conversion	No	3×10^{-6}	→	→	→
	3. Mount Rupture	•Loss of Engine Positioning •Possible duct rupture and fire consequences	•Recovery Probable	No	1×10^{-8}	→	→	1×10^{-6} @ 2 r/s
	4. All Other Failures	•Unscheduled Maintenance	None	Yes	1×10^{-2}	→	→	→

APPENDIX 1

Material Certification and Use of Material Property Design Data

1. Material Property Curves

These curves whether in terms of % creep, % yield, stress rupture, or ultimate (short time), are based on average test data usually from several specimens and several vendors (where possible). In terms of reliability, the probability of failure from any cause, whether time related or not, is one in two; therefore P = Q = 0.5

2. Deviation Factors

The material curves usually carry a number with them (from .75 to .95) called a deviation factor. The true and only definition of the material deviation factor is:

> The material curve stress values, be they creep, yield, rupture, etc. when multiplied by the deviation factor provide the designer with values of the minimum material strength which will be certified or accepted by the Material or Quality Control functions of the Manufacturing process.

Since the deviation factor forms a minimum level of material acceptance, it causes a discontinuity in the normal distribution of material strength (failures) and therefore has no co-relation with standard deviations from the mean failure level. Except for the fact that perfect homogeneity does not exist in a given batch or run of material, the use of deviation factors and material certification would provide minimum material strengths equivalent to an infinite number of standard deviations from mean failure strength. Since materials are actually not perfectly homogenous e , for example, between a test sample and the remainder used for manufacture, the combination of deviation

APPENDIX 1

2. factor and certification does not assure the necessary standard deviations from mean strength for required reliability and, more importantly, the designer has no quantitative knowledge of standard deviations in his design.

No standard procedure exists for establishing a material deviation factor; it is generally negotiated with vendors of the material and thus serves as an agreement as to what G.E. will and will not accept as "good" material. Factors influencing the actual value of a deviation factor include the complexity of an alloy - the difficulty in maintaining properties among batches, and the number of alternate vendors - lack of competition.

3. Material Certification and Sampling

Certification is, in effect, a notice to the user that a quantity of material has been tested and its properties meet or exceed minimum requirements of a designated G.E. material specification as evidenced by results of specified property tests.

a. Sheet Stock - Sheet stock is rolled from ingots. In some cases, several thicknesses of the same stock are rolled from a single ingot. It is general practice to certify the sheet material only once per ingot regardless of the number of stock sheets produced, so long as the sheets are the same thickness. Experience has shown that a single certification is justified by uniformity of the properties of single-thickness stock rolled from a single ingot. For the case in which several stock thicknesses are rolled from a single ingot, certification is generally required for each different thickness.

APPENDIX 1

3. b. Forgings - Even though the stock for, say 100 identical forgings may come from a single billet, the forging process so alters material properties that single certification cannot be used with confidence. Typical practice is to certify the material of groups of forgings which logically form during the forging process. For example, if units are heated and forged in groups of ten because of equipment limitations, a material certification will be made for each group of ten. Where large material flow takes place such as upsetting for blade dovetails, the lot certification may include samples of material from more than one area of a single forging.

4. Revised LJED Presentation of Material Property Curves

New material property curves issued through LJED Standards now include standard deviation data for calculating the probability of obtaining material with greater than or less than a specified strength. In general, strength curves not associated with time include a one standard deviation plot of stress reduction - vs - temperature. Time - temperature strength curves include a three standard deviation plot of strength. In either case, the number of standard deviations can be scaled as required.

APPENDIX 1

Application of Materials Safety Margins To Design Reliability

Percent safety margin practices yield little information as to the theoretical safety designed into the part. A part may be X% of stress safe or Y psi safe but there is no quantitative analysis possible which says that the part is designed to be 1 failure in 100 safe or 1 failure in 1 million safe. Using the standard deviation information will allow the design engineer to select a stress level which will correspond to a stated sureness that P% of the material will be at least that strong.

Briefly, the standard deviation data can be used to:

A. Aim for a designated degree of reliability in the part.

B. Compare the risk between alternate designs (i.e., the cost in terms of reliability of a weight saving or the relative reliability of two design approaches against a given weight bogey).

C. Construct a safety margin which is tailormade to the specific situation. To do this, assign numbers of standard deviations to each element of the design situation, then take the square root of the sum of the squares to get the overall factor. This method is advocated by Lusser and other authorities in the reliability field.

D. Appraise design approaches by comparing actual part failure history against the safety margin values initially used in the design. If approach C (above) is used, the validity of the overall margin and its elements can be appraised.

DETERMINATION OF ALLOWABLE DESIGN STRESS

To determine the tolerable short time stress level for the part:

1. Select a number (N) of standard deviations from Table I which best meets the reliability criteria for the part.

APPENDIX 1

2. Multiply the psi value for 1 standard deviation (for the temperature under consideration) by the number (N) of standard deviations determined in item 1

3. Subtract this from the average strength at the same temperature

To determine the allowable design creep, rupture, or relaxation stress level or parameter:

1. Select a number (N) of standard deviations from Table I as before.

2. From the data curve drawing determine the numberical value of 1 standard deviation in terms of stress (at a selected parameter) or in terms of parameter (at a selected stress) by dividing the numerical difference between the average and 3 standard deviation lines by 3.

3. Determine allowable stress or parameter as above

   Design Parameter = Avg. Parameter - N x Standard Deviation (Parameter)

   Design Stress = Avg. Stress - N x Standard Deviation (psi)

   NOTE: The presentation of the 3 standard deviation line on the parameter plot does not infer that 3 standard deviations should be used in every case. The 3 standard deviation line was chosen because it is far enough from the average to avoid confusion in reading the two lines.

   The need to determine easily either Design Parameter or Design Stress makes the format used for tensile properties impractical as a format of the stress vs. parameter presentation.

APPENDIX 1

PRECAUTIONS:

For the most realistic results, it is highly recommended that certain precautions be observed (otherwise a more conservative than necessary design may result):

1. Where the environmental conditions are fairly accurately known and the design calcuations correlate well with actual stresses the use of "nominal" design conditions (dimensional tolerances, temperature variations, etc.) is recommended. This requires only the comparison of one stress to the strength for a given safety measure. (A above).

2. Be sure the design claculations are stripped of as many arbitrary multipliers, additional "materials deviations," etc., as possible. Where operating environment is poorly defined in terms of temperature, loads, etc; or where accurate analysis of the stresses is difficult because of design complexity or probable manufacturing variations (i.e., complex sheet metal weldment), approach C above should be used. Use "nominal" stress calculations and assign additional units of the standard deviation to cover the undefinable variables, i.e., 3 standard deviation overall reliability target plus 1 sigma for poorly defined environment plus 2 for manufacturing variability. Taking the square root of the sum of the squares given an overall safety margin of 3.75 standard deviations to achieve the target of 3 standard deviations.

3. More precision is possible if the overall distribution (variability) of stresses with respect to variable G loading, dimensional tolerances, and other design variables can be determined. A more accurate analysis of the reliability or risk target is then possible than can be obtained by using nominal stresses. To do this, the standard deviation of the design stress must be determined. The overall risk with respect to a

APPENDIX 2

TABLE 1

N = Number of Standard Deviations
P = Probability of Strength $\geq$ Than
Q = Probability of Strength $\leq$ Than

$$P + Q = 1.0$$

N	P	Q	Q Fractional
0	.500	.500	1 in 2
1.0	.8413447	.15[illegible]3	1 in 6
1.5	.9331928	.0668072	1 in 15
2.0	.9772499	.0227501	1 in 44
2.5	.9937903	.0062097	1 in 160
3.0	.9986501	.0013499	1 in 740
3.5	.9997674	.0002326	1 in 4300
4.0	.9999683	.0000317	1 in 31,500
4.5	.999996602	.000003398	1 in 300,000
5.0	.9999997136	2.864×10^{-7}	1 in 3,500,000
5.5	.999999981	1.9×10^{-8}	1 in 500 Million
6.0	.999999999014	9.86×10^{-10}	1 in 1 Billion

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3. given stress level then becomes the product of the probability of a material strength value that low or lower (Q from Table I and Figure 1) times the probability of a stress value as high or higher than that under consideration. This is simply the Conditional Probability Law restated.

In the analysis of the materials property data from which the average and standard deviation values are derived, due consideration for the effectiveness of quality control in rejecting material below the specification minimum has been made. To the best of our knowledge the reliability or risk values will be representative of the true distribution of properties in finished parts after all specification acceptance testing and rejections. Variation of properties caused by manufacturing procedures (i.e., sheet metal fabrication) after acceptance are <u>not</u> included.

XV-5A AIRPLANE RELIABILITY PROGRAM

A. SUMMARY

1. Systems Analysis

Hydraulic, electrical, auto-stabilization, and flight control systems have been continuously evaluated for means of improving reliability and assessing consequences of design changes. Several component and sub-systems redundancies and special safety provisions have evolved from the quest for improved reliability.

2. Propulsion System

Initial efforts have produced failure probabilities of installed J85 turbojets. These results are summarized in Figure 8.

3. Mathematical Model

The thousands of components comprising an XV-5A have been mathematically related in a digital IBM program. The program can be used to compute XV-5A or major sub-system reliability for any specific time period and flight condition or it can be reversed to compute the component reliabilities needed to provide a specific required XV-5A reliability. The program will be used extensively as estimated component reliabilities become available.

4. Test Plans

The most important source of component data will be flight controls and hydraulic simulator. Preparation for this test is discussed. Other tests which will be useful in reliability evaluation are the airframe static load test, ground vibration test, and ground checkout tests.

B. PURPOSE

Part III of this report outlines the accomplishments to date of the Reliability and Failure Analysis Program for the U.S. Army Model XV-5A Lift Fan Flight Research Vehicle. Final quantitative predictions of aircraft flight safety and mission reliability are not set forth in this report. These data will be submitted as part of the Flight-Worthiness Report.

C. OBJECTIVE

The primary objective of the XV-5A Airplane Reliability Program is to produce an experimental aircraft possessing high inherent design reliability.

Given unlimited time and resources during the operational checkout phase of any experimental complex system, unknown but expected deficiencies will be detected. These deficiencies can be corrected by redesign and modification, but the process is too expensive, time consuming, and inefficient to be relied upon as the sole means for attaining an acceptable system. However, by application of accepted quantitative and qualitative reliability analysis techniques during the design phase, many of these deficiencies can be identified, and either eliminated or modified to bring the adverse effects of failures within acceptable limits, early in the design phase of development programs.

Other objectives include estimation of expected probability of detailed mission success, determination of operational flight safety criteria, determination of functional and preflight checkout requirements, and determination of maintenance requirements.

From every inherent weakness diagnosed and rectified during the design phase multiple benefits have accrued. These benefits include increased flight safety; reduced down time; reduced costs for changes; and, of greatest importance, a reduction of the likelihood that the research program objectives will be jeopardized by random failures, or subtle design deficiencies.

D. DISCUSSION

1. General Reliability Mathematical Model

For the purpose of Reliability Evaluation, the aircraft has been divided into 10 major systems. Each of these systems has been assumed to be statistically independent of the other systems. However, final quantitative analysis of the probability of catastrophic failure of the aircraft or of successful completion of a given mission will include factors resulting from systems interactions.

A general reliability mathematical model of the complete aircraft has been developed. This model assumes components have exponential failure density distributions with respect to time. The model further assumes that these components will not be used beyond their useful life. In other words, that they will be removed from the system and discarded or repaired before wear out failures occur.

A series system is defined as a system made up of a group of elements arranged such that if any one element fails, the system will fail. A parallel system is defined as a system made up of elements arranged such that all elements must fail before the system fails. Sample reliability block diagrams and mathematical equations for these types of systems are shown in Figure 6, where Q = probability of failure, λ = part failure rate, t = operating time, and e = the base of the natural of Napierian system of logarithms (2.718 ...).

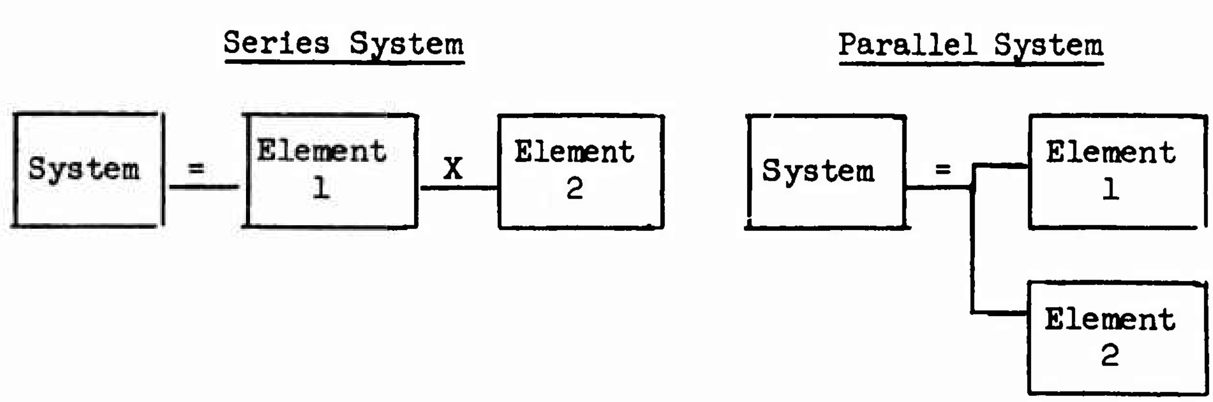


FIGURE 6

D. DISCUSSION

1. Once the probability of failure "Q," at a given time, has been calculated for either a series or parallel system, a single element with equivalent characteristics may be substituted for that system. Figure 7 shows one series system and one series parallel system and their relative positions in the detailed block diagram of the aircraft.

   The probability of failure equation of the abbreviated system, as shown in Figure 7 is as follows:

$$Q_1 = 1 - \left[e^{-t\left((\lambda_{19}+\lambda_{20}+\ldots\lambda_{27})+\lambda_{18}+(\lambda_{14}+\lambda_{15}+\lambda_{16})+(\lambda_3+\lambda_4+\lambda_6+\ldots\lambda\ldots\lambda_{12})\right)} \right]$$

$$\left[1 \left(1-e^{-t(\lambda_{30}+\lambda_{31}+\lambda_{32}+\ldots\lambda_{37})} + (\lambda_{39}+\lambda_{40}+\lambda_{41}+\ldots\lambda_{43})\right) \left(1-e^{-t(\lambda_{44}+\lambda_{45}+\lambda_{46}+\ldots\lambda_{63})}\right) \right]$$

The general solution of this equation has been programmed for execution by an IBM 704 Digital Computer. The data plotted in Figure 8 are the results of a typical subsystem computation.

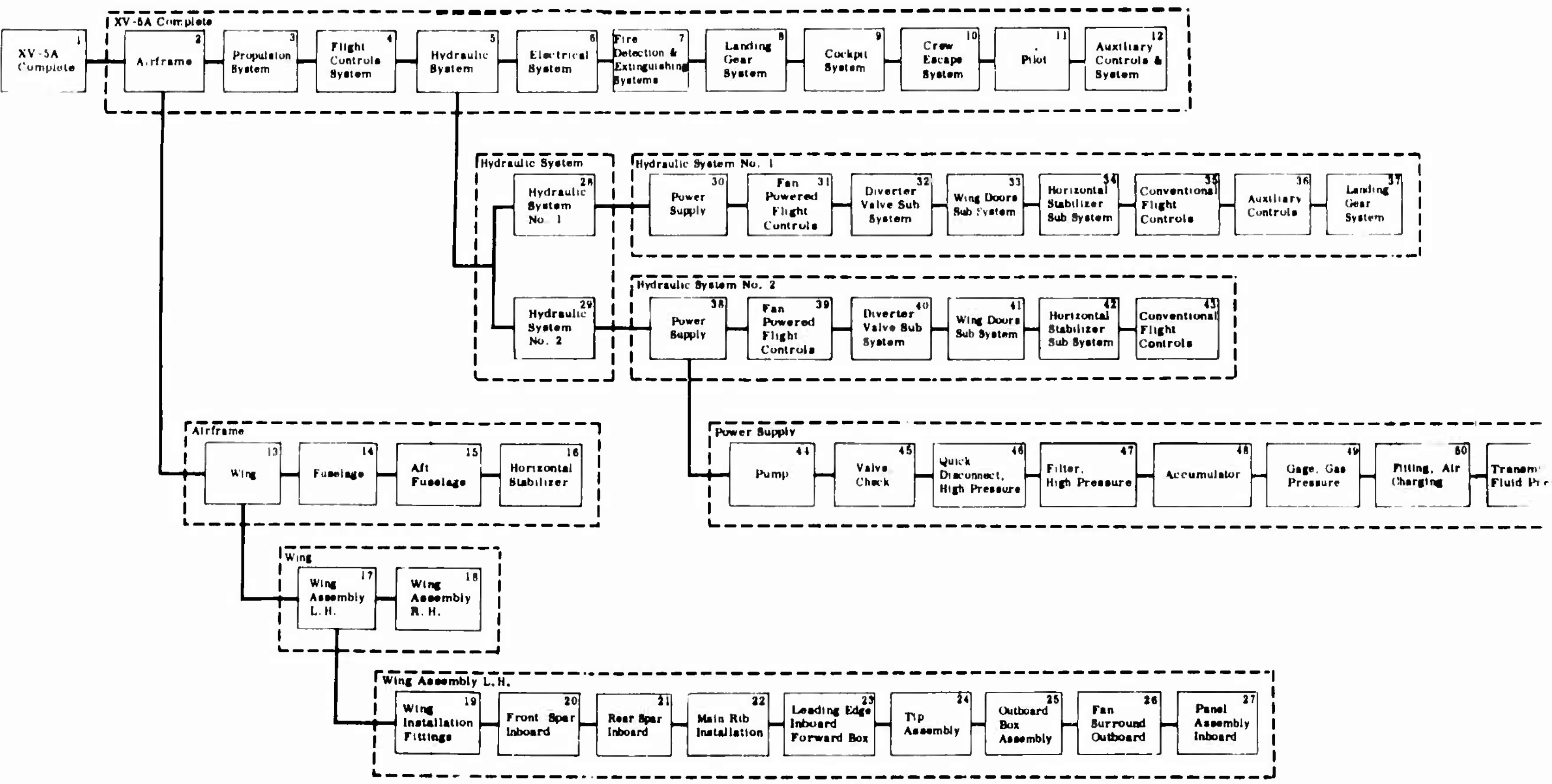
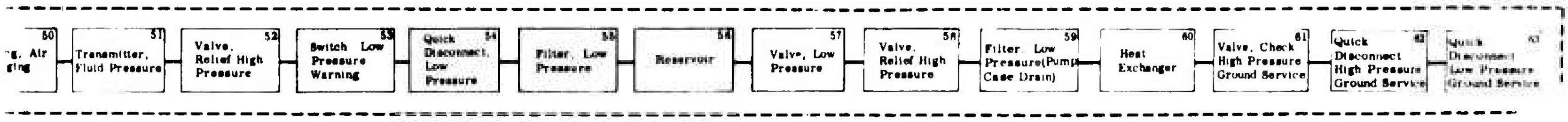


Figure 7. Abbreviated Airc...

A



ted Aircraft Block Diagram

B

D. 2. Systems Reliability Analysis

Both quantitative reliability analysis, as described above, and qualitative effects of failure analysis have been employed to evaluate and maximize the inherent reliability of the complete aircraft system. Many detailed reliability improvements have been incorporated as aircraft and subsystem design progressed. Some of the major studies and results are described in this section.

* Propulsion systems

Throttle Cut-Back System

Reliability aspects of a single gas producer automatic throttle cut-back system were investigated. Double throttle cut-back system was retained to minimize time required to regain 100% power after cut-back.

	Reaction Time
single unit	5.2 seconds
double unit	2.9 seconds

Minimum Recovery Envelope

Minimum Recovery envelope criteria is being developed based on the conclusion that the likelihood of simultaneous engine failures (within two minutes of each other) is too low to consider, but the engine out procedures (both engines fail more than two minutes apart) will be defined. (Ref. Figure 8)

* Flight Controls Systems

Ailerons

Aileron hydraulic actuators were originally proposed as single actuators having the left and right hand ailerons powered from the number one and number two hydraulic systems respectively, to save weight and still retain aileron function in event of single hydraulic system failure. System failure analysis, after incorporation of aileron droop requirements, showed

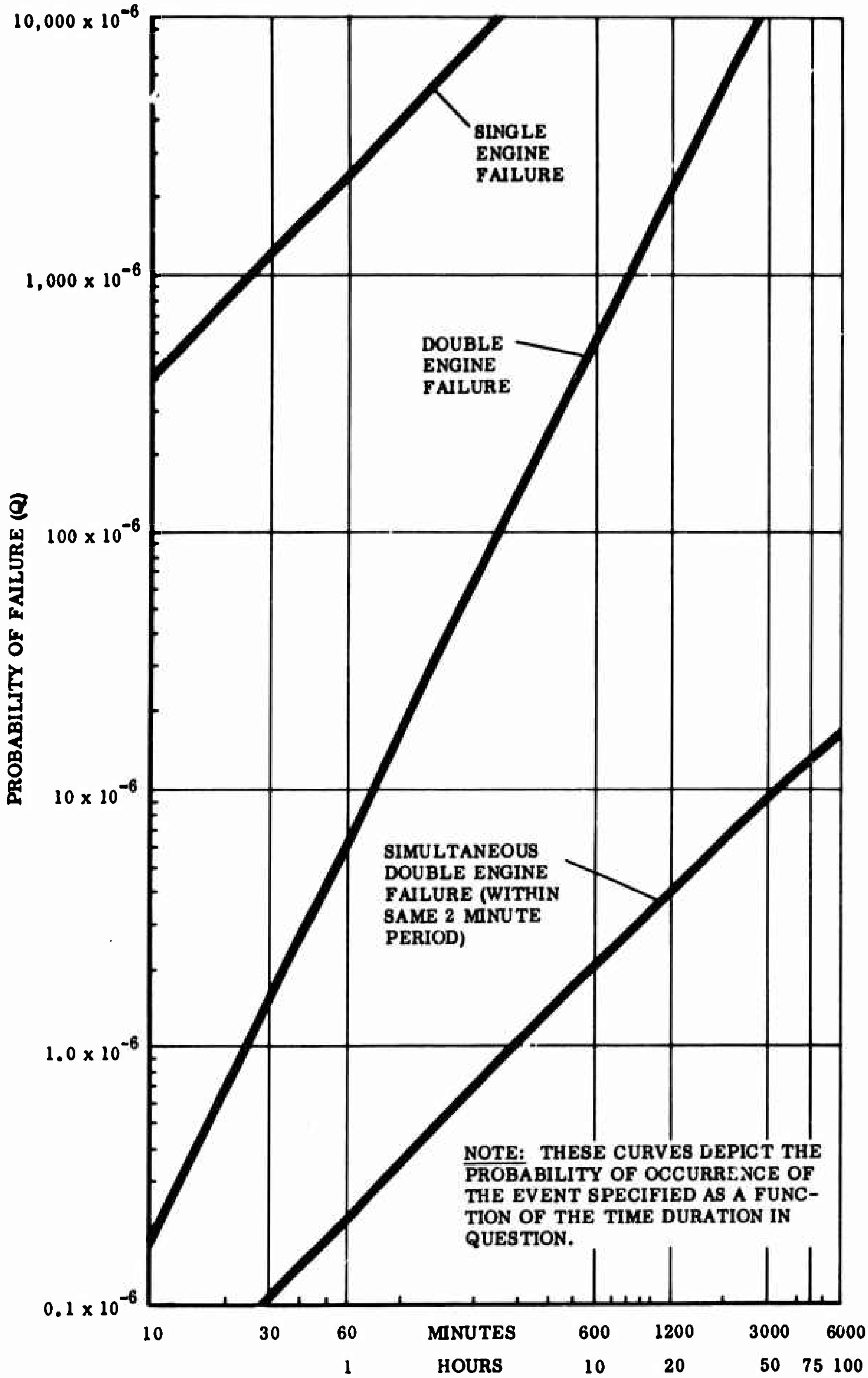


Figure 8 Propulsion Subsystem Failure Probabilities

D. 2. Ailerons (Cont'd)

uncontrollable stick reaction would occur during VTOL transition flight if one hydraulic system failed. Consequently tandem actuators were incorporated in the aileron system, resulting in full pilot control should either hydraulic system fail. Reduced, but still effective pilot control is provided by the aero-dynamic servo-tabs should both hydraulic systems fail.

* Hydraulic system

The preliminary design hydraulic system was composed of a primary and secondary power source and single actuators with automatic shuttle valves to shift to secondary power if the primary system failed. This system was more reliable than a single system, but a single fluid leakage failure in any actuator could cause loss of both systems. As a result of reliability analysis, all critical functions have been redesigned to incorporate tandem actuators, and all shuttle valves have been removed, increasing the reliability of the hydraulic system approximately three orders of magnitude.

* Electrical systems

Two single 28 volt DC generator systems were evaluated in terms of reliability versus weight savings with the following results:

System	Relative Chance of Flight Safety Failure	Ratio	Weight Savings
Original Dual System	1:1347	1:00	----
Single Generator with Single Drive	1:955	.71	32 lb.
Single Generator with Dual Hydraulic Motor Drive	1:1280	.95	12 lb.

D. 2. * <u>Electrical Systems</u> (Cont'd)

The amount of weight saved was not considered adequate justification for reduced reliability.

Further electrical system improvements include a three bus 28V DC system, a two bus 115V 400 cycle AC system, primary and standby control circuits for fan powered flight controls and conversion control systems. The 28V DC electrical loads are divided into three categories; 1) Non-Essential loads; 2) Essential loads; and 3) Emergency loads, and are fed from buses identified accordingly. Two independent generators, one on each engine, provide power to the three buses. In addition to the generators, a silver-zinc battery, sized to supply five minutes power demand on the emergency bus, including one conversion mode change, is provided. Generator fault detectors are provided such that if either generator should fail, the non-essential bus drops out of the circuit, and if simultaneously or subsequently the second generator should fail, the essential bus drops out of the circuit and only the emergency bus powered by the battery remains operative. Cockpit indicators warn the pilot of generator failures. The 115V 400 cycle AC loads are divided into two categories: 1) essential loads, and 2) non-essential loads. As long as both inverters function properly, both buses are supplied power. If either inverter should fail, the non-essential bus is dropped out, and the essential bus is powered by the remaining inverter. Warning lights on the cockpit annunciator panel warn the pilot of inverter failures.

* <u>Landing Gear</u>

Emergency actuation. The landing gear is normally extended and retracted by one hydraulic system. Since free fall, down locking of the gear is considered marginal due to gear configuration, a high pressure gas operated emergency system has

D. 2. * Landing Gear (Cont'd)

been incorporated. The high pressure gas is stored in the main landing gear struts and is manually controlled by the pilot to effect emergency "conventional gear down" operation.

* Auto-stabilization System

The Auto-stabilization system is composed of three basic subsystems: 1) three axis rate gyros, 2) amplifiers, and 3) electro-hydraulic servo valves, and operates only during VTOL flight mode.

The rate gyro and amplifier subsystems each have two independent redundant systems, the primary system and the standby system. The servo valves are controlled by the primary or standby systems on manual command from the pilot. An indicator light in the cockpit notifies the pilot of amplifier hardover command failures in any axis, or all axes. The control authority of the auto-stab system is limited to 25% of pilot control authority and can be overpowered by the pilot. A hardover failure has the effect of an out of trim condition and can be eliminated by transfer to the other system.

Preliminary flight simulator studies indicate that loss of one of the three axis control loops is readily compensated for by the pilot. Further investigations in this area are planned.

Independent redundant electro-hydraulic servo valves were investigated but proved to be undesirable since they would degrade the reliability of the dual input mechanical servo actuators. Several precautions have been taken to alleviate the effects of failure of the auto-stab electro-hydraulic servo valves.

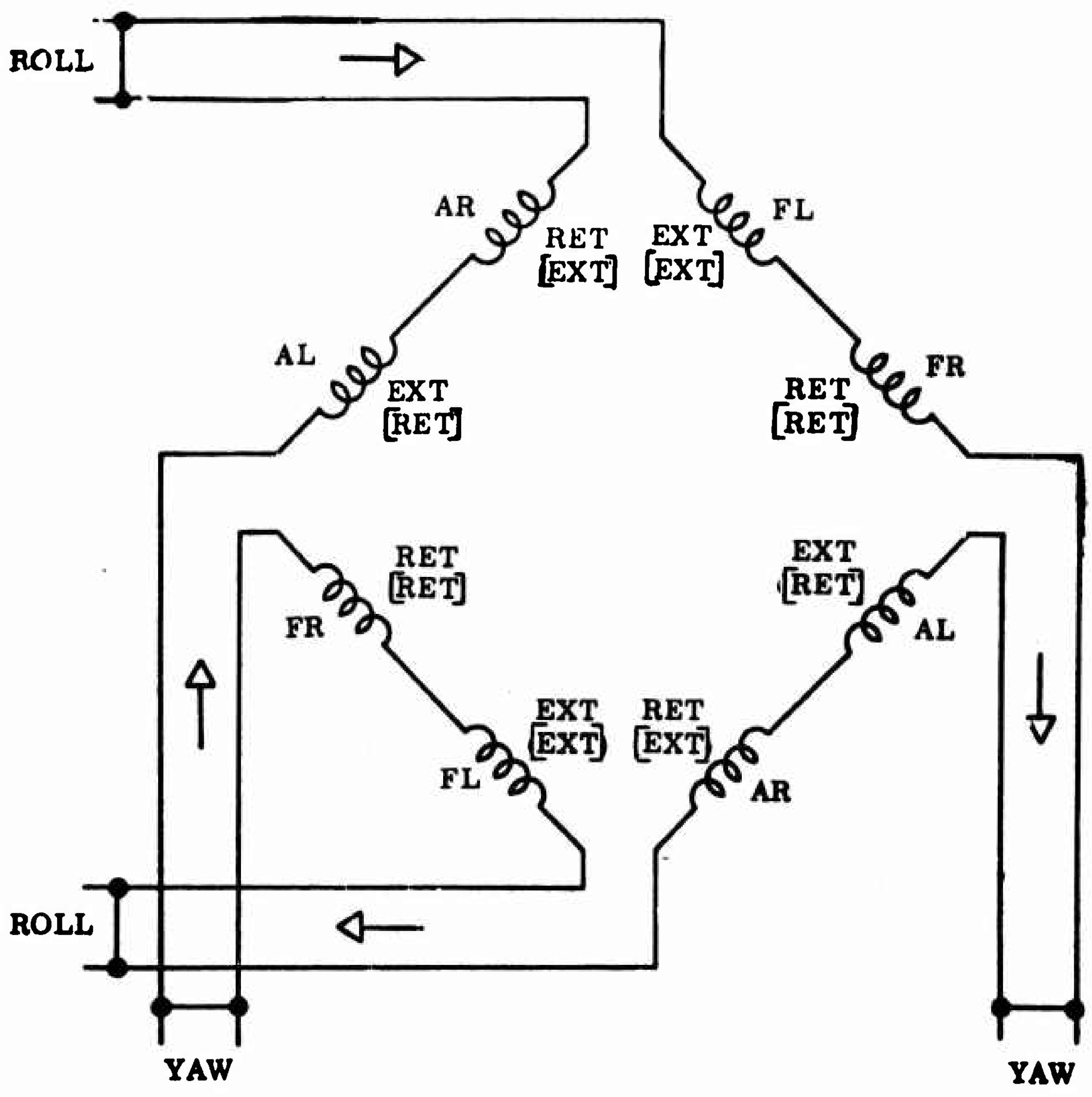
D. 2. * Auto-stabilization system (Cont'd)

Each servo valve torque motor has two coils, each capable of functioning alone. The roll/yaw servo valves are wired in a bridge circuit such that at least two coils must fail before the roll/yaw function is lost. The pitch servo valve coils are paralleled for the same reason. The coils and amplifiers have been designed to operate at the higher voltage and current requirements of operation after a single coil failure. See Figure 9 for Auto Stabilization Servo Valve Wiring Diagram.

In order to prevent complete loss of the auto-stab function due to failure of a single hydraulic system, the four louver actuator auto-stab servo valves are paired off on the two hydraulic systems.

Three combinations of pairs are possible and during normal system operation are equally satisfactory. However, considering single hydraulic system failures, each combination produces control system cross-coupling. Two of these combinations produce adverse roll/adverse yaw and yaw/adverse roll effects. The adopted combination of left fore/right aft servos on one hydraulic system and left aft/right fore on the other hydraulic system results in a small amount of yaw/lift or roll/thrust coupling. This coupling is of small magnitude and is expected to be practically negligible since the auto-stab system will have about 50% of its normal authority in this failure mode.

Connector Mismatching - In order to maintain interchangeability, the rate gyro packages for primary and secondary auto-stab systems have the same connectors. This makes it possible to cross-connect either the power or signal cables to the gyro packages. A cursory checkout of the system would not disclose this defect. Under normal flying conditions the fault could still go undetected. However should a malfunction within either the primary or secondary systems cause a power loss for that system, the result would be



-▷ - DIRECTION OF CURRENT FLOW
AR - AFT RIGHT LOUVER ACTUATOR VALVE COIL
AL - AFT LEFT LOUVER ACTUATOR VALVE COIL
FR - FORWARD RIGHT LOUVER ACTUATOR VALVE COIL
FL - FORWARD LEFT LOUVER ACTUATOR VALVE COIL
EXT - EXTEND
RET - RETRACT
[ ] - ACTUATOR DISPLACEMENT DUE TO YAW SIGNAL

TWO COILS ARE CONTAINED IN EACH SERVO VALVE.

Figure 10 Auto Stabilization Servo Valve Wiring Diagram

D. 2. * Auto-stabilization system (Cont'd)

the complete loss of both auto-stabilization systems. In order to preserve independent parallel redundancy, matched pairs of power and signal cables will be potted together for a sufficient distance from the gyro packages so that crossconnecting will not be possible.

* Conversion control system

The conversion control system provides for sequencing and interlocking the three interdependent conversion functions. They are wing fan closures, horizontal stabilizer (programmed) and diverter valves. See Figures 10 and 11 for complete interlock diagrams.

CTOL To VTOL Conversion

Flight simulator studies indicate that the ideal sequence of these functions is (1) wing inlet doors start open, (2) horizontal stabilizer start "programmed" trim, (3) diverter valves operate. The horizontal stabilizer and diverter valves are interlocked to prevent operation until after the wing inlet doors reach a preset position.

If the diverter valves and horizontal stabilizer get out of phase with each other, the induced pitching moments are more than the pilot can reasonably control. The conversion functions have been further interlocked to preclude this event. The horizontal stabilizer program starts first and is monitored by the diverter valve command circuit. If the horizontal stabilizer does not start its programmed motion, the diverter valves will not cycle, and the conversion sequence ends; but the aircraft is still in a controllable configuration.

Assuming normal horizontal stabilizer operation, diverter valve function is then monitored; and if the diverter valves fail to operate, the horizontal stabilizer motion is stopped. This

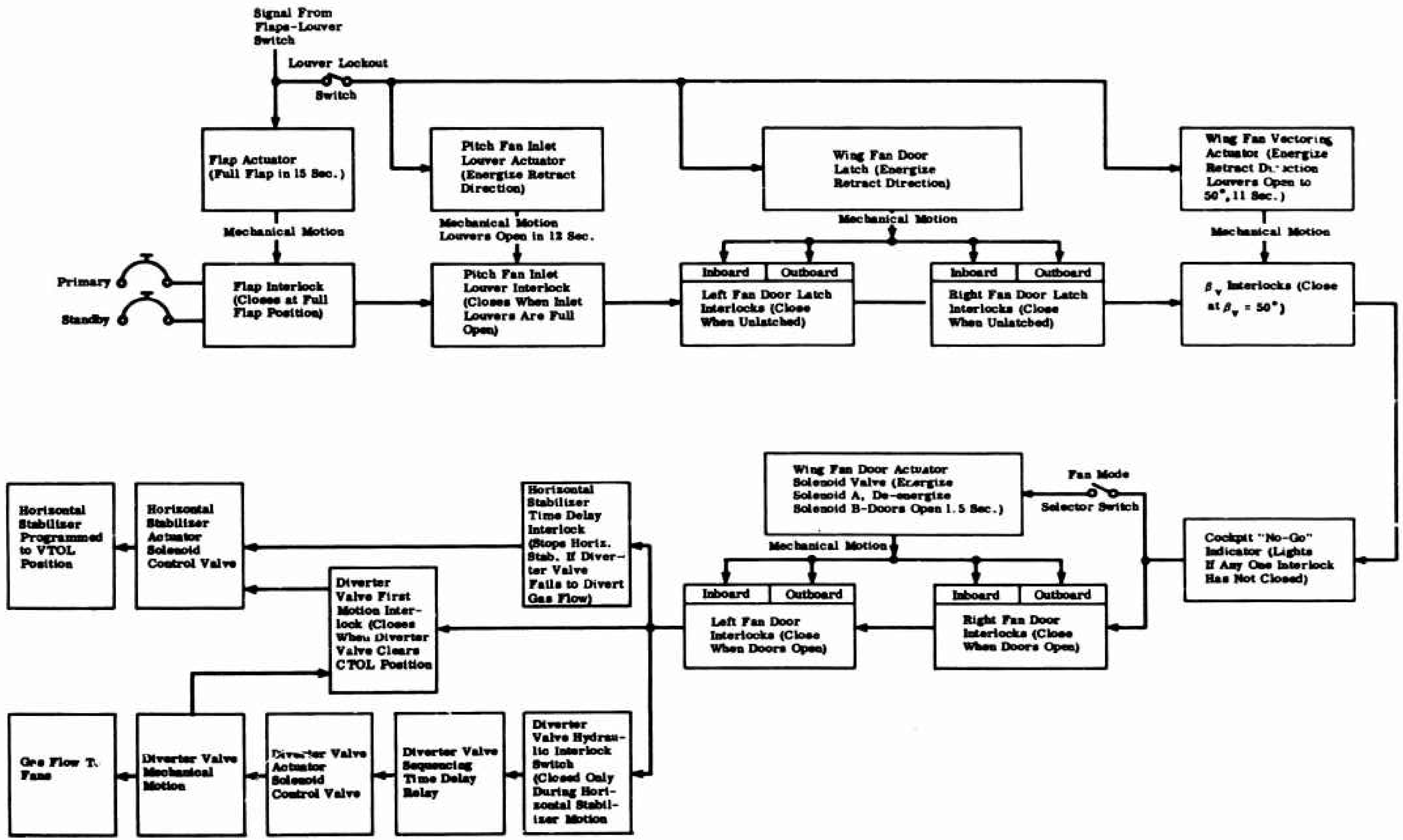


Figure 10 Conversion Control Interlocks for CTOL to VTOL Flight Mode

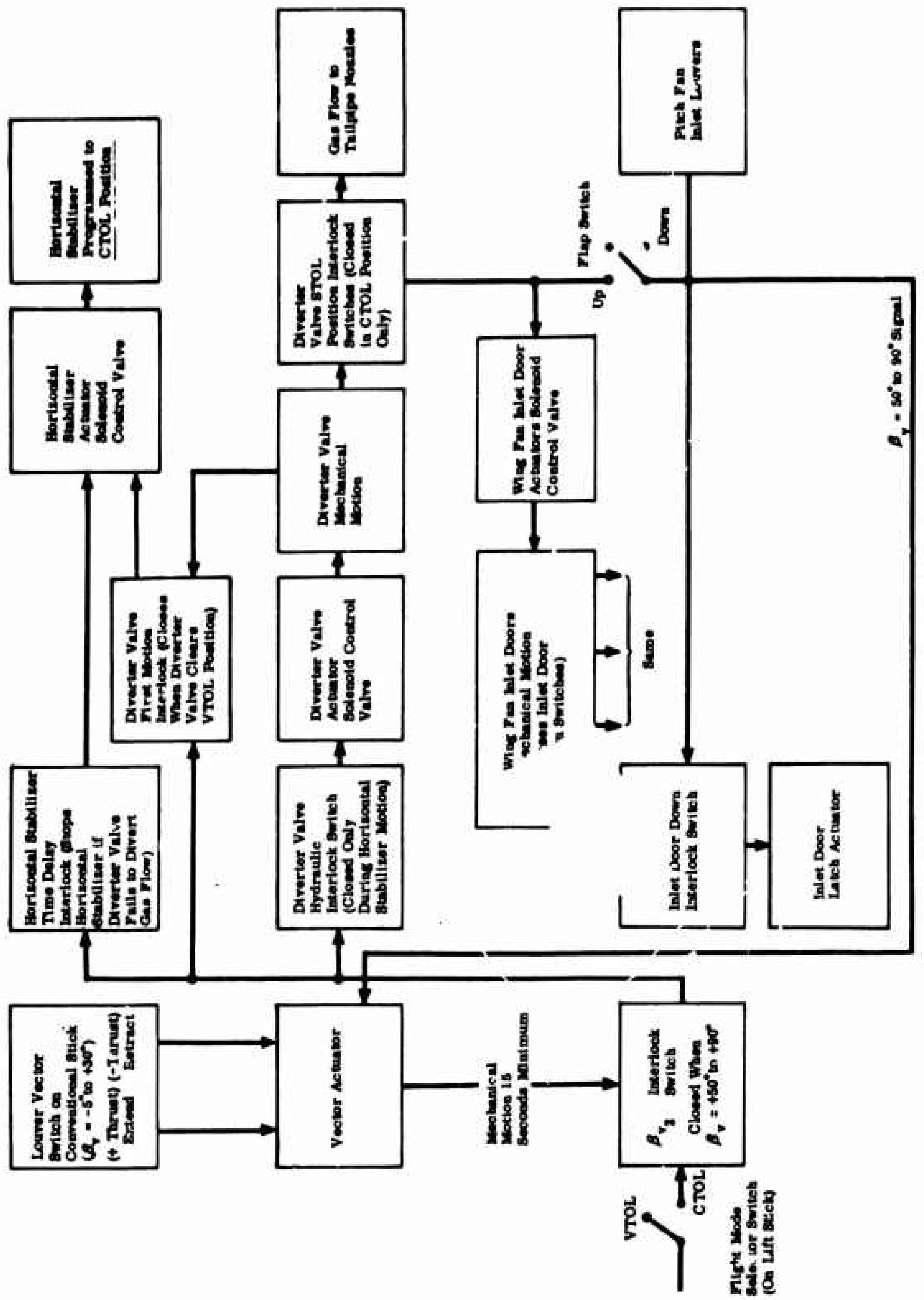


Figure 11 Conversion Control Interlocks for VTOL to CTOL Flight Mode

D. 2. CTOL To VTOL Conversion (Cont'd)

point is such that induced adverse pitching moments are still controllable by the pilot. Further, if the diverter valve "no function," signal stopping the horizontal stabilizer is a fault (the diverter valve functioned normally but the monitor circuit malfunctioned) the resultant aircraft pitching moments are still controllable.

VTOL To CTOL Conversion

VTOL to CTOL conversion sequence, as optimized with the flight simulator, is accomplished by starting the diverter valve cycle simultaneously with the horizontal stabilizer "programmed" trim; and after a specific time delay, the wing inlet doors are closed.

In order to provide the same fail safe approach to this conversion sequence, a similar automatic functional cross check is performed. The conversion sequence is initiated by the horizontal stabilizer "programmed" trim. If the horizontal stabilizer does not start, the sequence is stopped. If the horizontal stabilizer "programmed" trim functions, the diverter valves cycle starts. This cycle is also monitored; and if the diverter valve cycle does not start, the horizontal stabilizer motion is stopped. Thus the horizontal stabilizer is never permitted to be at such an angle of attack that uncontrollable pitching moments would be introduced, regardless of the position of the diverter valves.

The wing inlet doors are interlocked with the diverter valves and cannot be cycled closed until both diverter valves have completed the cycle to CTOL position.

* Hydraulic simulator plans

The hydraulic simulator will be used as a full scale test stand to collect functional and performance data for the aircraft hydraulic and control systems. Complete operating histories will be recorded for the entire system and each component.

D. 2. * Hydraulic simulator plans (Cont'd)

This data will include functional, performance, loads, environments, operating time or operating cycles, during installation, checkout, and simulated flight operations. From reliability analysis of this data, component and system failure rates and failure modes will be estimated. In addition, this data will be utilized to establish aircraft operational and maintenance criteria, update the aircraft reliability mathematical model, and provide preliminary data to the flight worthiness reliability report.

Some of the major elements of the hydraulic simulator are illustrated in Figures 12 through 15.

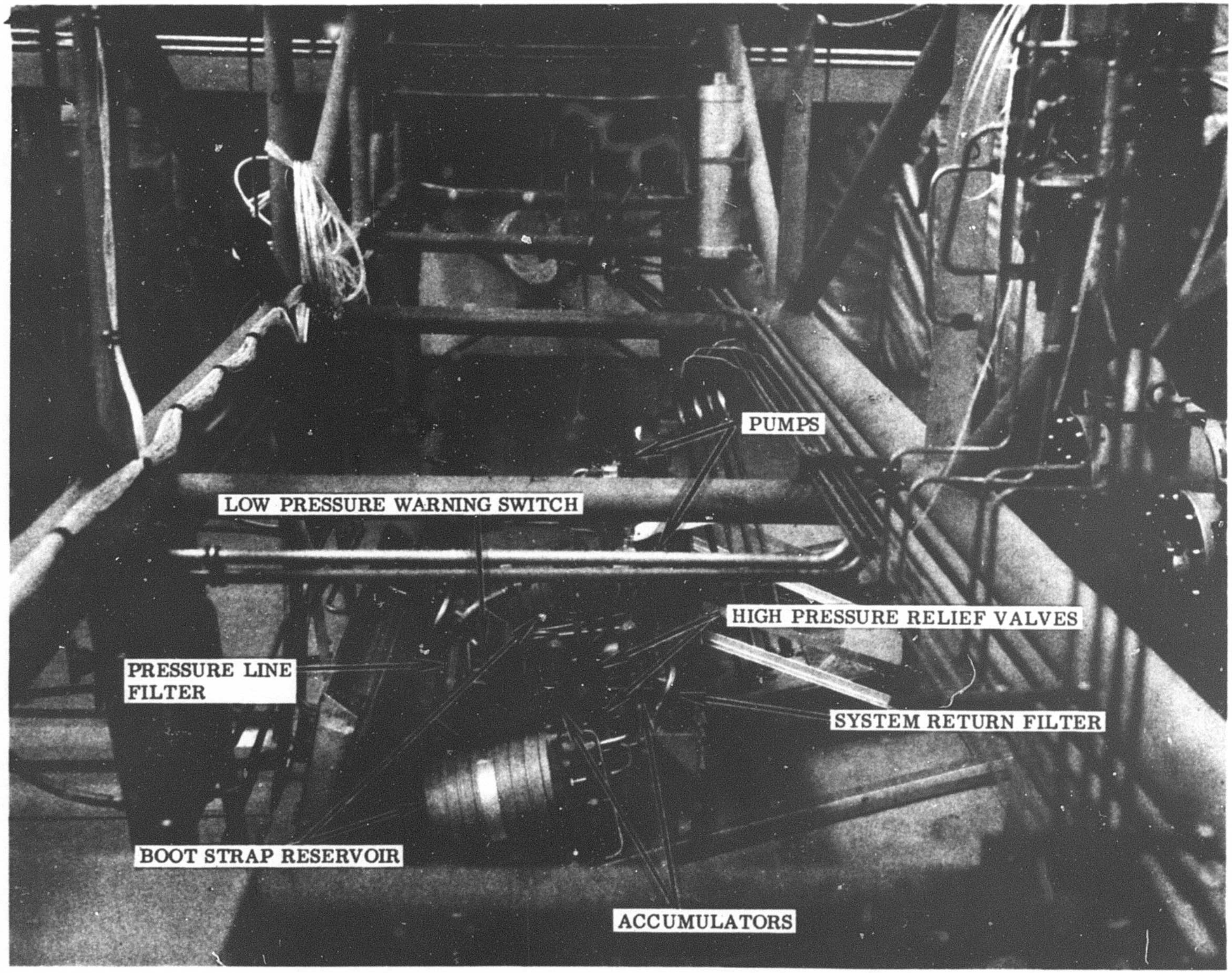


Figure 12. Simulated Aircraft Hydraulic Compartment

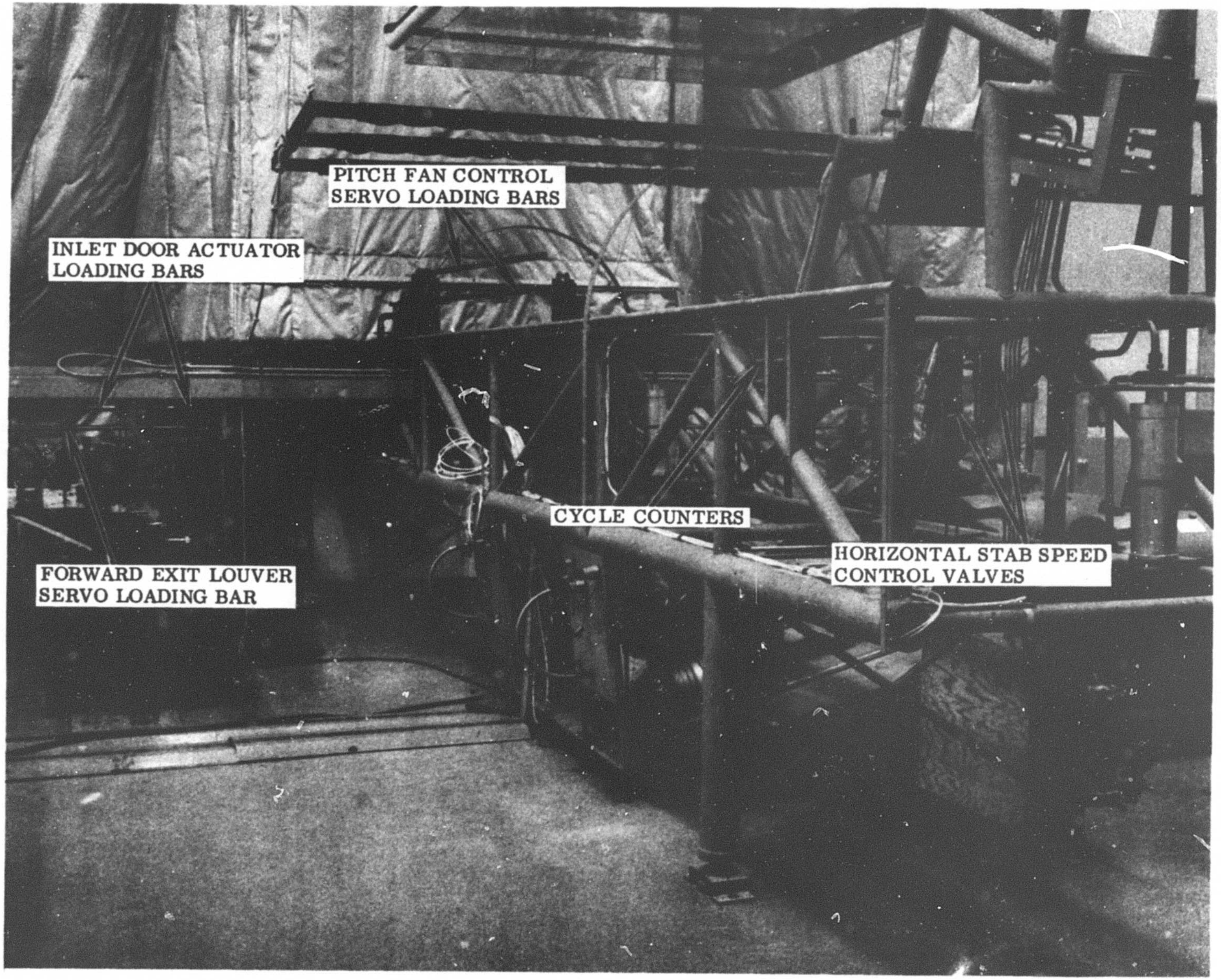


Figure 13. Loading Bars

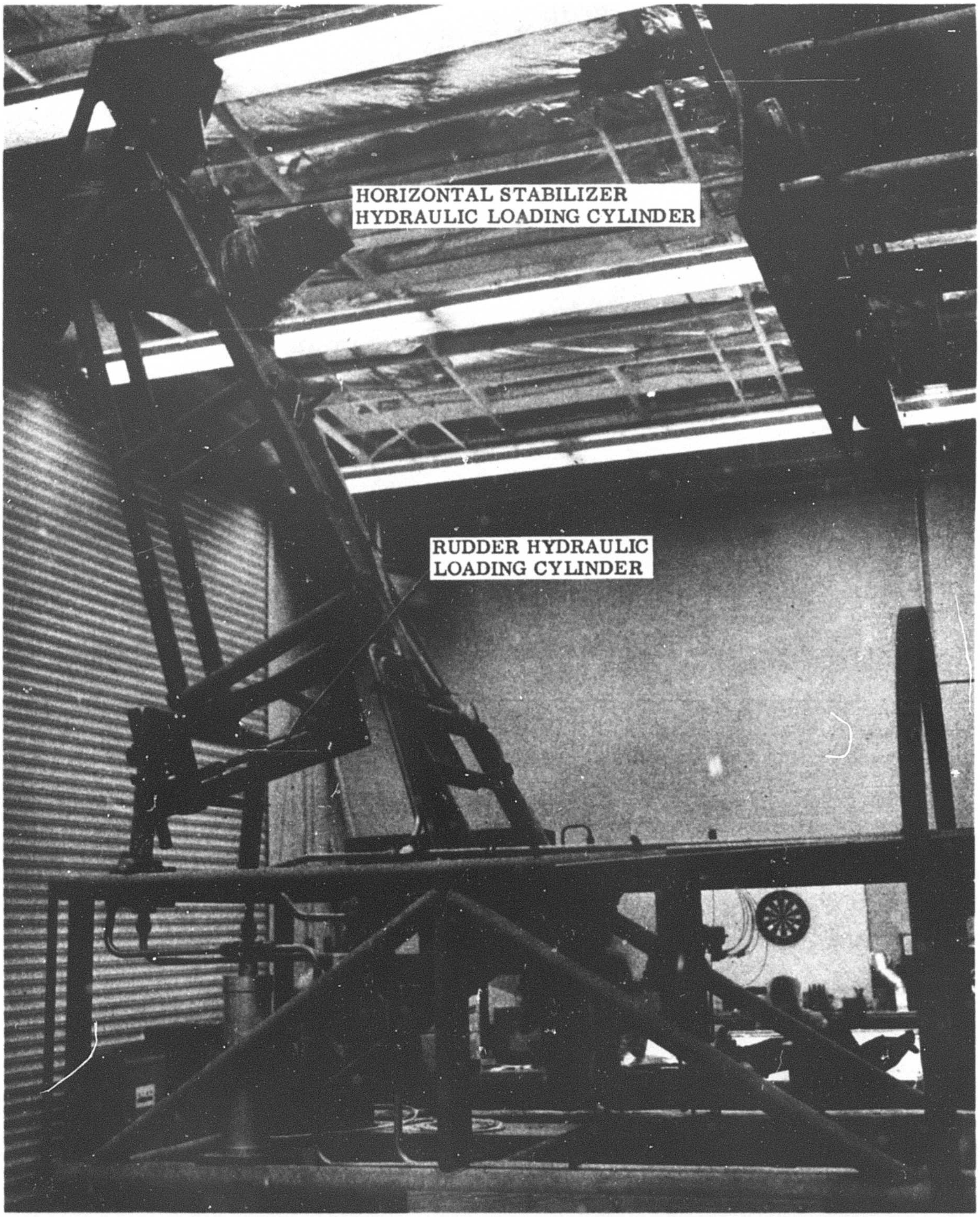


Figure 14. Loading Cylinders

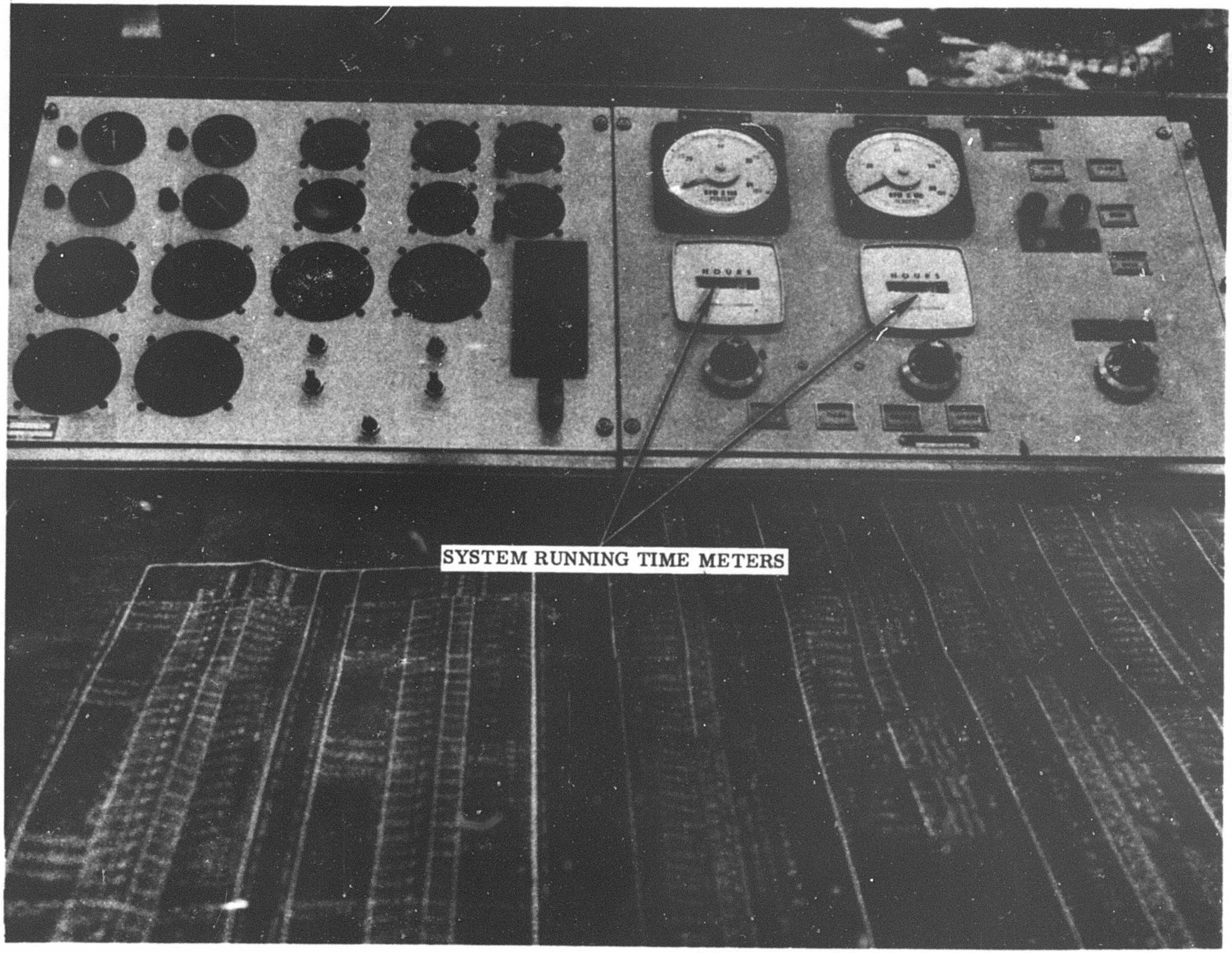


Figure 15. Remote Indicating Panel