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Follow-On Studies for Design Definition of A Lift/Cruise Fan Technology V/STOL Airplane

Boeing Military V/STOL Group

Final Report

Volume 1

Technical Report

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Prepared Under Contract NAS2-9277

THE BOEING COMPANY
Military Airplane Development Organization
P. O. Box 3999
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for

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National Aeronautics and Space Administration

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16. Abstract

A three-engine, three-fan V/STOL airplane was designed for use as a Research Technology Airplane in proof-of-concept of a candidate configuration for use as a Navy multimission airplane. Use of mechanically interconnected variable pitch fans is made to accommodate power transfer for flight control in hover and to provide flight capability in the event of a single engine failure. The airplane is a modification of a T-39A transport. Design definition is provided for high risk propulsion components and a development test program is defined.

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SYMBOLS AND ABBREVIATIONS

A	aspect ratio - b^2/S; nozzle area-ft^2 (m^2)
A/P	airplane
ASW	antisubmarine
b	wing span-ft and (m)
BCAV	best cruise altitude and velocity
BPR	bypass ratio-fan airflow to primary airflow
BS	body station
$\bar{c}$	mean aerodynamic chord
c.g.	center of gravity
C_D	drag coeffficient, D/qs
C_L	lift coefficient, L/qs
$C_{L_{max}}$	maximum lift coerfficient
C_{L_α}	lift curve slops
CSAR	combat search and rescue
D	drag, lb (N)
DDA	Detroit Diesel Allison
e	Oswald efficiency factor on drag due to lift ($C_D = C_{D_o} + CL^2/\pi Ae$)
F	thrust, lb (N)
F_g	gross thrust, lb (N)
$F_{g\ max}$	maximum gross thrust, lb (N)
$F_{L/C}$	lift cruise fan thrust, lb (N)
F_N	net thrust, lb (N)
F/W	thrust-weight ratio, lb/lb (N/N)
G.E.	General Electric Company

gpm	gallons per minute
GW	gross weight
HLH	heavy lift helicopter
HP	horsepower
IOC	initial operating capability
Ix Iy Iz	moments of inertia, slug ft^2 (kgm^2)
kn	knot
KEAS	knots equivalent airspeed
KTAS	knots true airspeed
L	lift, lb (N)-characteristic length
LE	leading edge
M	Mach number; pitching moment ft lb (N m)
MAC	mean aerodynamic chord
MH	maximum horizontal flight Mach number
N	number of engines; yawing moment ft/lb (N/m)
OEW	operating empty weight, lb (N)
P	pressure lb/ft^2 (N/m^2)
$P_{T_{max}}$	maximum total pressure, lb/ft^2 (N/m^2)
$P_{T_{min}}$	minimum total pressure, lb/ft^2 (N/m^2)
q	dynamic pressure, lb/ft^2 (N/m^2)
R_c	compressor pressure ratio
R_F	fan pressure ratio
S	wing or reference area, ft^2 (m^2)
SA	surface attack

SAS	stability augmentation system
SFC	specific fuel consumption, lb/hr/lb (Kg/hr/N)
STOL	short takeoff and landing
T	temperature, deg.
TAS	true airspeed
t/c	thickness/chord ratio
T_{SLST}	temperature, sea level std. temp.
T.O.	takeoff
TOGW	takeoff gross weight, lb (N)
V	velocity, kn (m/s)
V_H	level flight maximum speed
VL	vertical landing
V_o	freestream velocity
VOD	vertical onboard delivery
V/STOL	vertical/short takeoff and landing
VTO	vertical takeoff
VTOGW	vertical takeoff gross weight, lb (N)
VTOL	vertical takeoff and landing
W	weight, lb (N), airflow, lb/s, N/s, watts
WBL	wing body line
W_F	fuel flow, lb/hr (N/hr)
Greek:	
α	angle of attack, deg (rad)
γ	flightpath angle, deg (rad)
δ	pressure ratio P/P_{SL} std; flap deflection, deg (rad)
θ	pitch angle, deg (rad); temperature ratio, $\frac{T}{T_{SLST}}$

λ gross thrust vector angle relative to the horizontal body reference line: when thrust is horizontal and forward, $\lambda = 0^o$, when thrust is vertical and up, $\lambda = 90$ deg (rad)

$\Lambda_{c/4}$ sweep of the quarter chord line, deg (rad)

ϕ roll angle, deg (rad)

ψ yaw angle, deg (rad)

0.0 INTRODUCTION AND SUMMARY

The objectives of the follow-on study, NAS2-9277 are to refine the modified T-39 Sabreliner research and technology conceptual design for a truly low-cost aircraft; and to conduct sufficient design effort on power transmission and control systems to identify the technical risks, and to obtain a more detailed estimate of the design, fabrication, and testing costs. Primary emphasis of the study is on two full-mission modified T-39A aircraft. Component and system designs reflect the needs of the Research Technology Airplane (RTA), and not particularly the needs of an operationally oriented design. Technology Demonstrator design priorities are safety, program costs and mission performance.

Areas of the propulsion and control systems identified as "risk" items have been studied to a depth sufficient to determine that development can be accomplished within the program projected costs and schedules.

The study includes three tasks:

A) Task I is a refinement of the research and technology aircraft design for low cost and risk.

B) Task II includes design details, analysis, and development methods and costs for the propulsion system components identified as critical items.

C) Task III is an iteration of the Task I design, and a departure of the Task I design. Task III is accomplished to determine the degradation in research and technology demonstration capability of an aircraft having a cost reduction of 20% over the Task I aircraft.

The approach to Task I included:

A) Design and fabrication of the technology aircraft shall follow the "experimental shop approach".

B) Control systems will be designed with minimum sophistication, but will not compromise safety. Existing surfaces, cables, rods and control parts to be retained whenever possible and available off-the-shelf hardware and software systems to be identified where possible.

C) Development testing to be lumped into a single static test rig capable of qualifying all propulsion , mechanical interconnect, flight control and interface systems prior to flight test.

D) Two and three engine configurations driving three fans to be examined with the intent of producing a cost effective design. Thrust and weight margins, performance, mechanical interconnect simplicity and engine availability are all factors in the propulsion assessment.

E) Maximum use of the existing airframe to be a primary design requirement. Areas of major modification to be identified early in the study to allow enough time to properly evaluate the impact of these changes on the remaining structure.

F) A minimum avionics suite to be selected, using as many of the existing components as practicable. Additional systems being those associated with the V/STOL flight mode.

G) A number of egress systems have been identified as potentially satisfactory for the demonstrator airplane. Of these, only one system was believed capable of safely ejecting pilot and co-pilot over any part of the flight regime with minimum qualification.

H) Extensive analysis of control power requirements over a variety of flight situations was completed to determine a fan beta stop as a method of simplifying and therefore reducing the cost of the fan control limiting system to protect the drive train components from overload.

Under Task II, detail design and analysis of components of the power transmission system considered to have "risk" were completed. The T-box, drop box, engagement clutch, and the gearbox cooling system were studied in detail. Results and conclusions were formulated and appear in the text of this report.

In conclusion, the Task I design, Model 1041-135-2A demonstrator airplane will provide a suitable lift/cruise fan V/STOL test bed, and will meet or exceed the design guidelines as written or implied in the work statement. The mechanical interconnect system as design is completely within the state-of-art, parallels helicopter mechanical interconnect design, and will easily allow for engine-out flight in all modes.

0.1 CONTEXT OF THE STUDY

The work reported herein is a follow-on to a Design Definition Study of a Lift/Cruise Fan/Technology V/STOL Airplane - Summary, NASA CR-137749, August 15, 1975, Contract NAS2-6563.

Other currently funded studies providing design basis and definition information in support of this study are:

- o Large Scale Variable Pitch Lift/Cruise Fan Tests for the 40 x 80 Foot Wind Tunnel, RFP 2-25950 (FVF) dated 31 October 1975. Contract Number NAS2-9215.
- o Design and Fabrication of a Lift/Cruise Fan V/STOL Model for Wind Tunnel Test, RFP 2-25781 (FVF): Contract Number NAS2-9178 dated July 11, 1975.
- o Preliminary Design of the Flight Control System for Boeing V/STOL Technology Demonstrator Airplane, RFP 2-26234, Contract Number NAS2-9177.

0.2 DEMONSTRATOR OBJECTIVES

This demonstrator is designed to permit evaluation of total airplane configuration concept, the basic propulsion concept and how all of the requisite elements function as a unit.

Objectives include:

- o Definitive operating techniques for V/STOL aircraft - to include practical take-off and approach corridors
- o Use of integrated propulsion/aerodynamic flight control system
- o Fly-by-wire flight control system for the VTO and STOL modes
- o Use of variable pitch fans for hover control
- o The ability to perform conventional maneuvers and meet specified VTOL and STOL and CTOL Test missions
- o Demonstration of the high speed mechanical drive system including engagement clutch, 'T' box, drop box and shaft interconnect system

Factors for evaluation will include: (Relative to Operational Airplane)

- o Fan pressure ratio simulation
- o Thrust-to-weight margins
- o Control System Capability
- o Handling Qualities
- o Pilot workload
- o Possible use of advanced system (i.e., STOLAND)
- o Operational considerations: (i.e. shipboard compatibility as a demonstrator)
- o Induced Aerodynamic Effects

1.0 TASK 1 - MODIFIED T-39 CONCEPTUAL DESIGN REFINEMENT

In 1975, under contract NAS2-6563, a baseline modified T-39, Model 1041-135-2 was defined. A general description of the necessary changes to propulsion, structures, flight control systems, etc., was written. Preliminary weights and performance were calculated.

The modifications identified at that time included: a new nose to accommodate the lift fan installation; new vertical and horizontal tails; canopy replaced with a lightweight enclosure fixed in place; adaption of the existing mechanical flight control system; increased flap deflection for rotating lift/cruise fan propulsion pod clearance, a new hydraulic system; a landing gear system utilizing modified existing main gear with a new nose gear installation, and a new propulsion system consisting of two engines driving three fans (2 lift/cruise and 1 lift) through a mechanically interconnected drive system.

The two turboshaft engines were identified as Allison T-701's using water injection, a contingency rating, and incorporating minor turbine changes to achieve suitable single engine emergency performance.

The operating weight of the modified T-39A, Model 1041-135-2 was estimated in 1975 to be 17,100 pounds resulting in a net weight increase to the T-39A operating weight of 7300 lbs.

1.1 DESCRIPTION OF MODEL 1041-135-2R

In order to proceed with the Conceptual Design Refinement Study of 1976, Contract NAS2-9277, an update of the Model 1041-132-2 design was necessary. This necessity was primarily the result of two causes:

a) Additional detailed design data on the T-39A being made available to assess more precisely the extent of structural modifications necessary.

b) Weight growth resulting from a more detailed assessment of structural changes, including the need for an alternate landing gear due to the higher operating weights; and the introduction of "contingency weight" - that being an allowance for "probable" weight growth of the system.

This newly defined modification of the T-39A, with its improved structural change definition, and updated weight statement, warranted a new model number. Subsequently, Model 1041-135-2R (Revised) was chosen to designate the new baseline, two engine-three fan modified T-39A from which to begin Task I.

The -2R operating weight was calculated to be 19530 lbs., or 2430 lb. heavier than that for the 1041-135-2 model.

1.2 DESCRIPTION OF MODEL 1041-135-2A

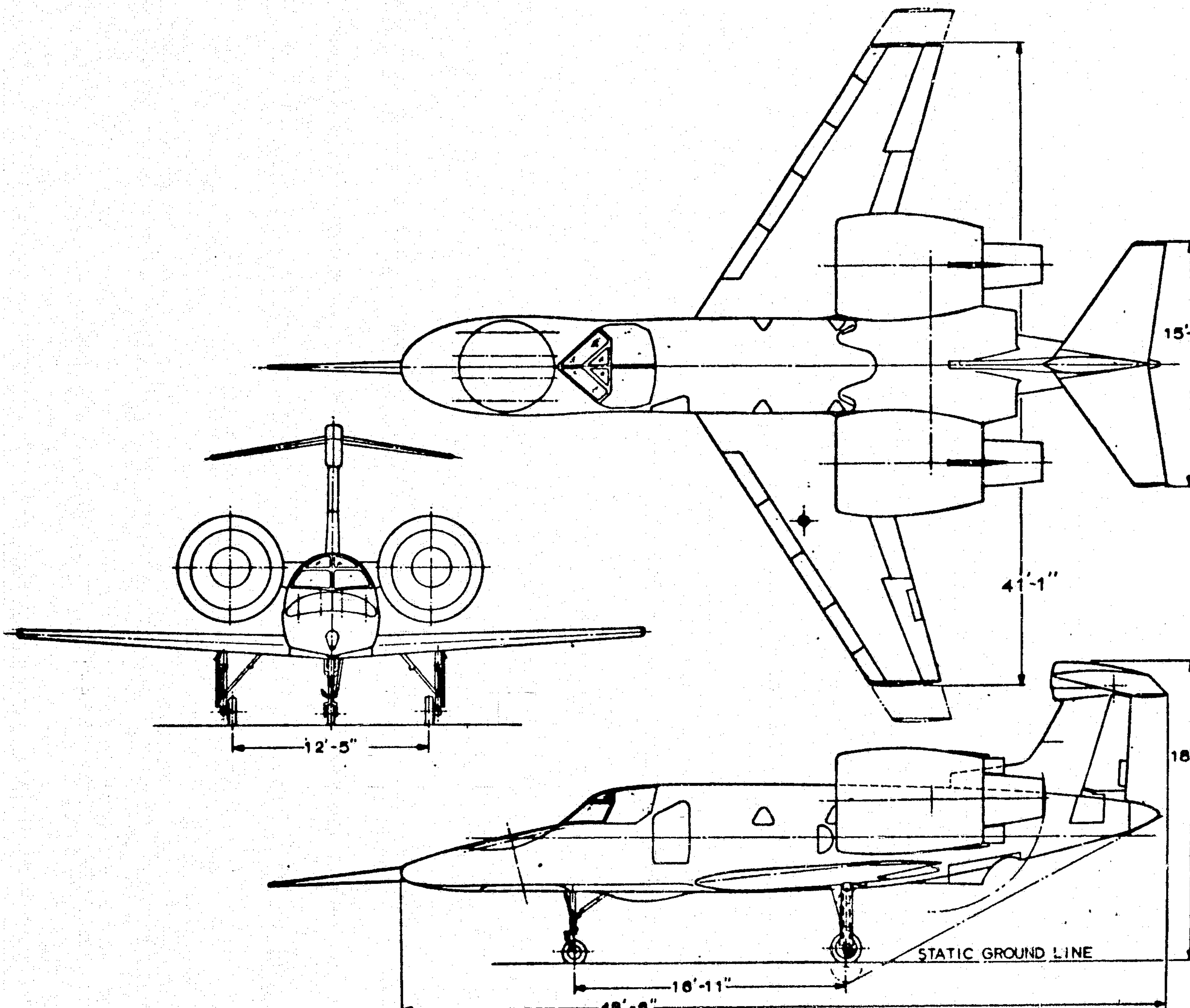
Propulsion performance calculated for the Model 1041-135-2R showed that engine out thrust-to-weight margins for a two engine T-39 V/STOL demonstrator were unacceptable for the predicted operating weight. A third Allison XT-701 engine was then added to the baseline Model-135-2R, resulting in the three engine, three fan Model 1041-135-2A.

The third engine is installed inside the fuselage and aft of the center wing section. Additional major modifications to the -2R, as a result of the third engine, included:

a) addition of a drop box (helical gear) and drive shaft connecting the third engine to the drive system drop box.
b) addition of all subsystems pertinent to the third engine (i.e., fuel, control, mounting, etc.).
c) addition of fuselage inlet and exhaust ducting.

A three-view of Model 1041-135-2A is shown in Figure 1.2-1. The configuration retains as much of the basic T-39A structure as possible. Some areas are strengthened locally while other areas are of completely new design. The propulsion system, nose section, horizontal tail center section, and vertical fin fall into the latter category.

Landing gear, flight deck canopy, and wing are modified or exchanged for a suitable alternative as required. Specifically, the landing gear is changed to incorporate a modified A-4 system; a portion of the flight deck canopy structure is changed to allow for installation of a frangible section allowing for crew egress; and the wing tips are cropped 22 Inches. making the wing suitable for loads at the higher weights. The operating weight of the -2A configuration is calculated to be 21600 lbs., or 2070 lbs. heavier than the 2R baseline.



Model 1041-135-2A

FIGURE 1.2-1

The resulting modified T-39 V/STOL demonstrator airplane, Model 1041-135-2A, is a low wing, high T-tail configuration. Two aft body mounted lift/cruise fans and one nose lift fan driven by three turboshaft engines power the airplane. The two lift/cruise fans are capable of pivoting through an angle of 100° about a common axis perpendicular to their thrust centerlines. Two of the three engines are mounted inline and aft of each of the lift/cruise fans. Each of these engines is supercharged by the efflux of the fan immediately ahead of it. The third engine, interbody mounted, remains unsupercharged.

A mechanically interconnected system of shafting, gearboxes and clutches permit engine-out operation and provide means of control power transfer in the VTOL mode.

1.3 DESCRIPTION OF MODEL 1041-135-2B

An effort was made during Task I to determine the feasibility of an alternate third engine. The engine selected for comparison was the T-56 A-14 producing approximately 4500 SHP. The label identifying this 3 engine T-39 V/STOL derivative is Model 1041-135-2B. The installation of the single spool T-56 is similar to the third engine XT-701 installation in the -135-2A.

Possible cost reduction, slightly higher operating weight and the fact that the T-56, like the XT-701's, is an Allison engine were all reasons for addressing, to some degree, this derivative. The operating weight of this configuration was calculated to be 21810 lbs., up 210 lbs. from the -2A.

1.4 CONFIGURATION COMPARISONS

The three designs, Models 1041-135-2R, -2A, and 2B are all basically the same airplane. Variations in propulsion system and propulsion system interface equipment make up the physical differences. Impacts of these differences are felt on:

- o Installed static thrust/weight margins for VTO performance
- o RTA mission performance
- o Program risk
- o Program costs

Figure 1.4-1 illustrates these impacts on the three designs. The weight ranges between minimum and maximum hover weights allude to the built in weight growth allowance or lack of it for each of the designs. Where no growth allowance is present, the airplane would have to be restricted from flight in a "dead man zone".

1.5 INTERNAL ARRANGEMENT AND MODIFICATIONS

The inboard profile, Figure 1.5-1, shows airframe and landing gear modifications and drive train additions required to modify the T-39A to a Technology Demonstrator non-pressurized Lift/Cruise Fan Airplane.

1.5.1 Significant Airframe Modifications

The nose contour forward of the cockpit is extended and re-faired to enclose a lift fan duct unit including high speed duct closure panels and duct support structure without alteration of the T-39A windshield, see Figure 1.5-2. Control sticks replacing T-39A control yokes are shown. Left and right frangible cockpit covers replace T-39A overhead cockpit structure and the upper portion of the bulkhead immediately aft of the cockpit is sloped aft to provide for the ejection system. The T-39A nose gear support beams are modified to support the higher energy capacity nose gear, lift fan drive shaft bearing supports, drive shaft cover and new nose gear closure doors. Fiberglass fairing will be attached to the body skin to fair the new nose section into the body ahead of the entrance door. Space for payload is reserved inside the body aft of the cockpit to the wing front spar attachment bulkhead. Speed brakes are removed and replaced by structural skin panels. Support provisions for the centrally located unitized drop box, lift fan engagement clutch, alternator transmission heat exchanger and blower, and the third engine overrun clutch truss are added at the wing rear spar support frame, and at the frame immediately aft of the wheel well. Third engine inlet ducts are faired into each side of the midbody and connect to a plenum at the face of the engine.

A notch in the upper body contour with local longeron re-routing is for installation of the integrated left and right power pods, and box beam interconnect structure which houses the 'T' gear box, pod support bearings

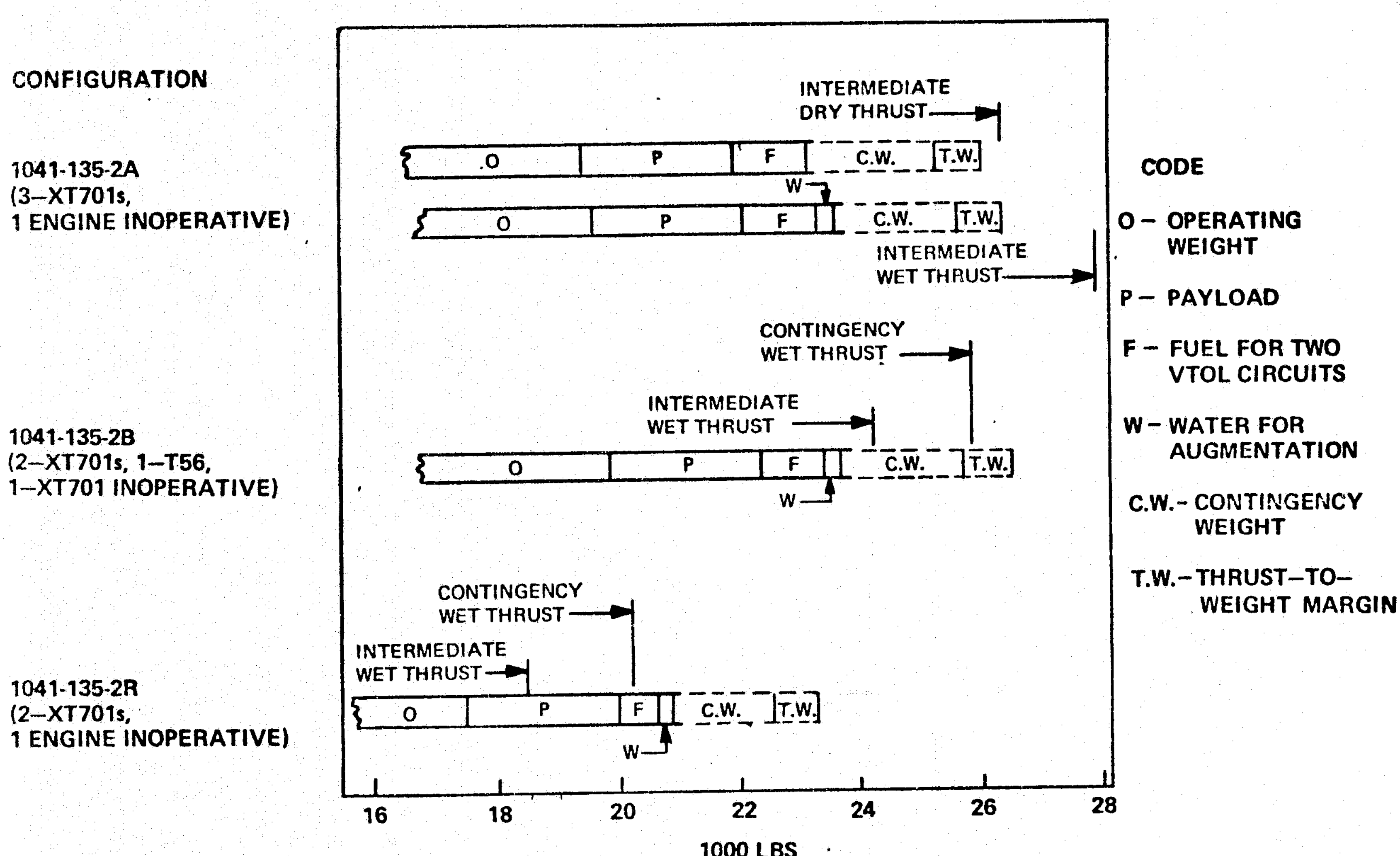
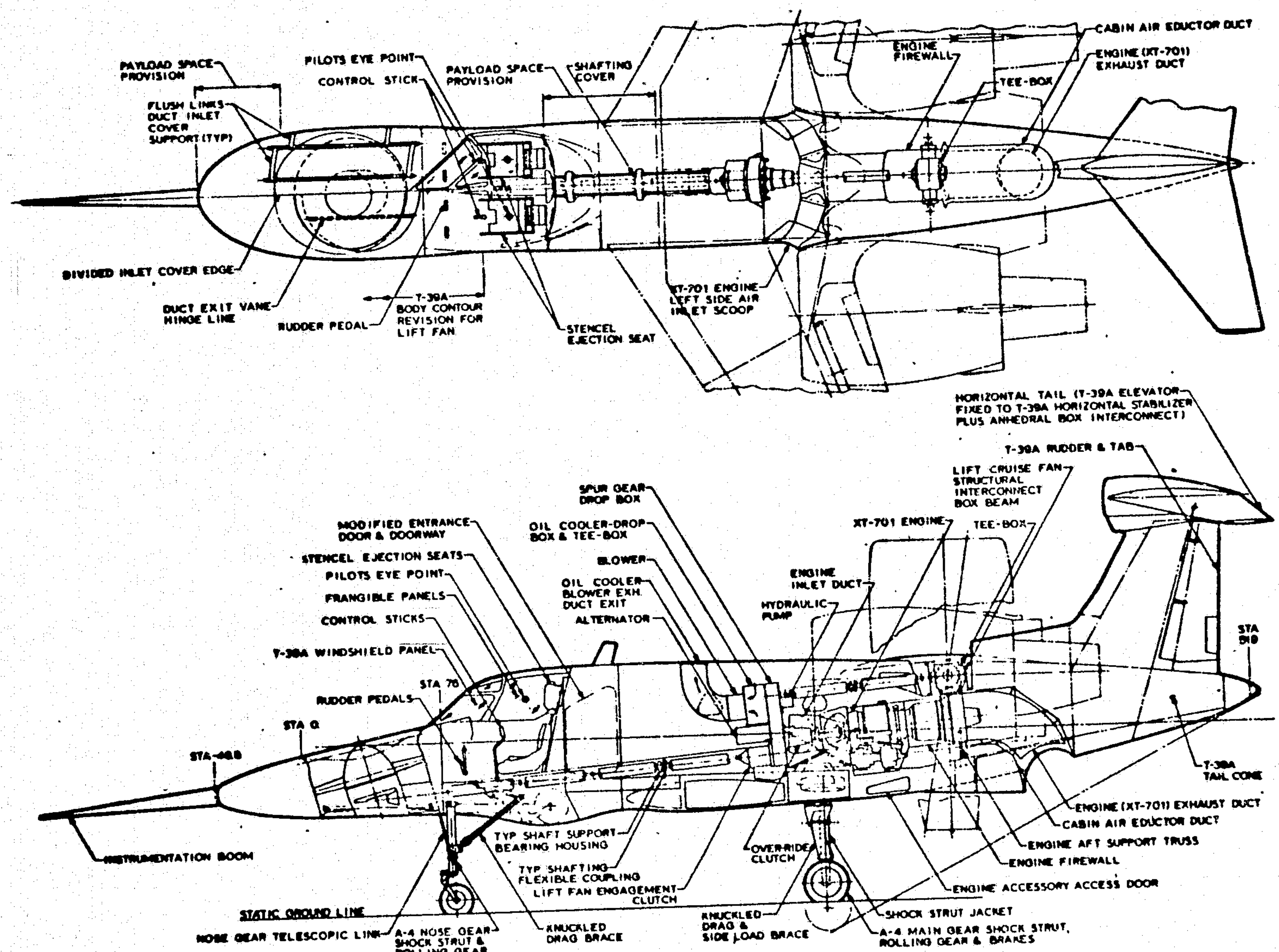
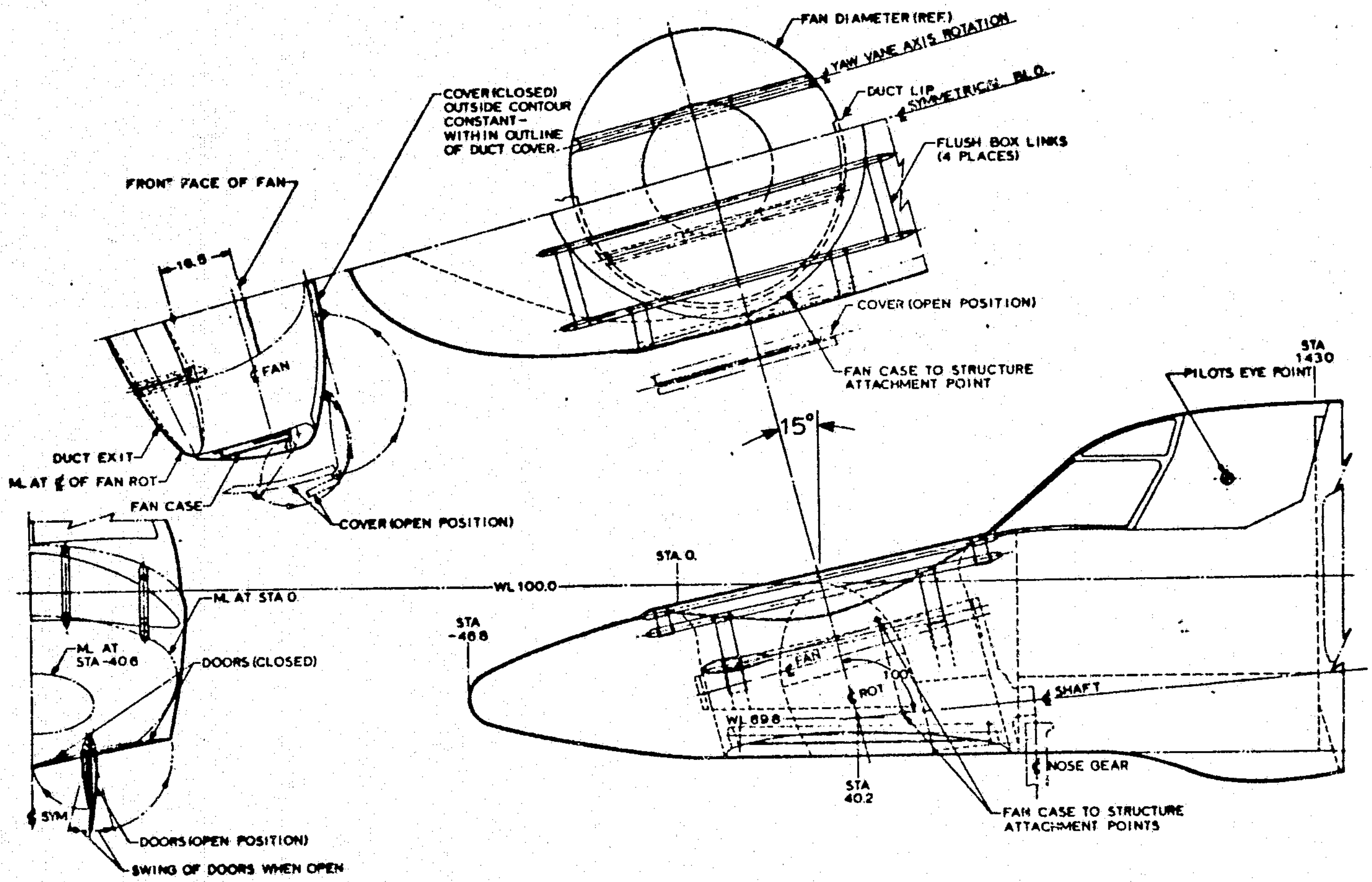


FIGURE 1.4-1 . Thrust and Weight Comparisons, Baseline Configurations Emergency Take Off Weight, One Engine Inoperative SLS, 90°F Day



Inboard Profile

FIGURE 1.5-1



Lift Fan Installation

FIGURE 1.5-2

and pod harmonic drive system. The body torque box structure is completed by a removable "V" brace above the interconnect box. It provides a transverse shear and longitudinal axial load path across the notch replacing the original upper skin panel (Figure 1.5-3). An additional frame forward of the pod interconnect box structure is added to assist in transferring vertical pod loads into the body structure. The T-39A rear fuselage access door opening, between the lower longerons, is lengthened forward and aft to permit hoisting of the engine into the body. Cabin air is exhausted through an eductor duct enclosing the engine exhaust duct. A partial bulkhead is added to the base of the fin to provide a torsion and bending load path from the fin front spar into the body skin. The fin front spar has a transverse bending tie in place of the single pin tie on the T-39A. Appreciable re-skinning of the body at the base of the fin to accommodate increased loads from the 'T' tail configuration is contemplated.

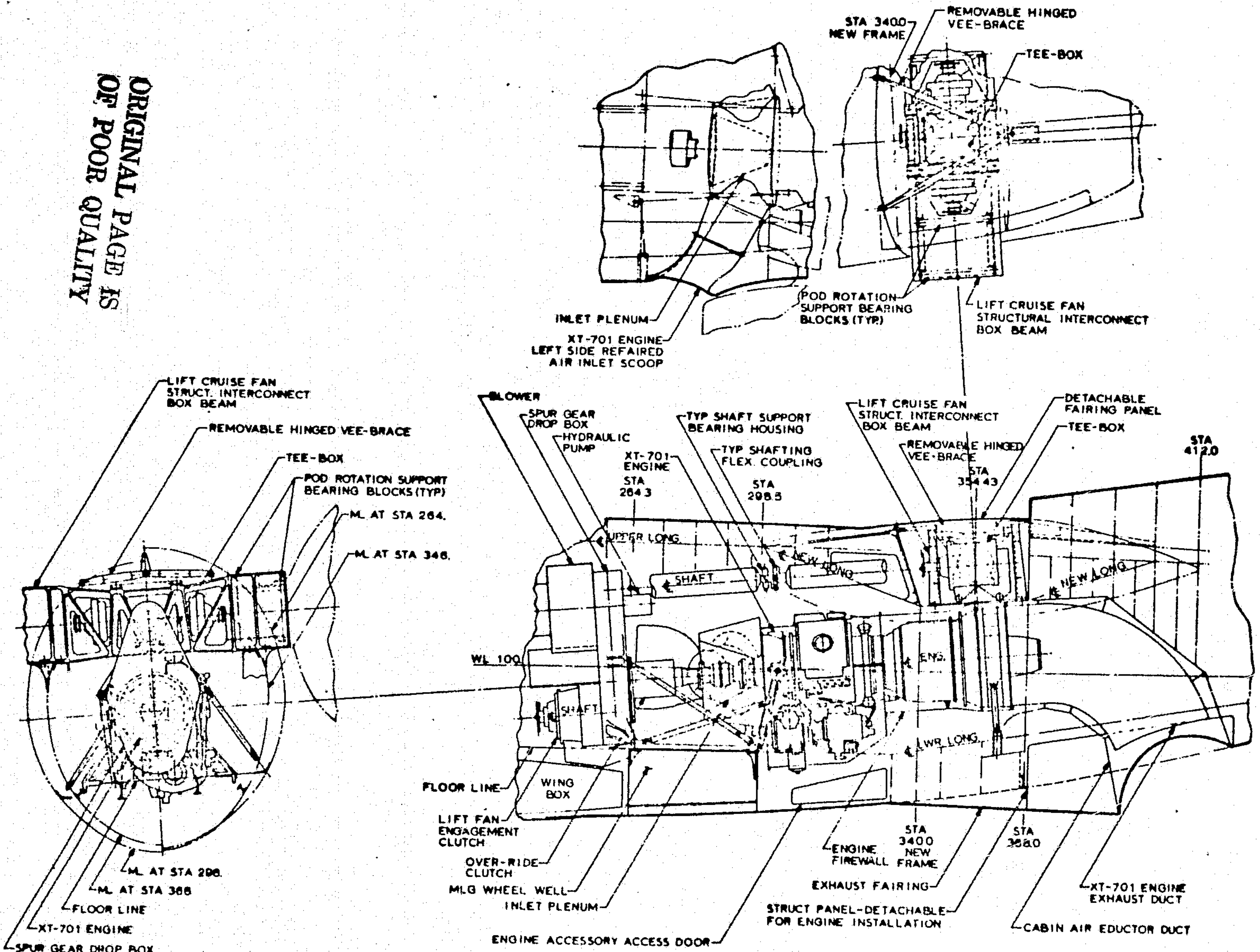
The fin structure is new. It has a front spar and removable leading edge skin panel to provide for access and installation of the horizontal stabilizer power control system. A machined upper rib including stabilizer surface hinge fittings for the stabilizer actuators are enclosed in a surface intersection fairing at the top of the fin. Fairing is to be functional over normal stabilizer high speed throw angles used for control, but not over full trim range required for V/STOL.

The horizontal stabilizer is of the slab type with annhedral to minimize flutter structural penalties. Flight control requirements do not indicate the need for an elevator at this stage in the design cycle. Should subsequent analysis, wind tunnel testing, or simulation evaluation show a desirability for additional control power, it will be provided.

The rudder and tail cone are T-39A components with closure skin added over the tail cone openings provided on the T-39 for the horizontal stabilizer carry through structure.

1.5.2 Wing Modifications

Wing modification consists of a wing root trailing edge fillet of sufficient depth to include the main landing gear support auxiliary beam



Third Engine Installation

FIGURE 1.5-3

and a small hinged panel in the flap that can be deflected to avoid fan flow impingement in the VTOL mode. Fittings on the wing rear spar at juncture with the auxiliary beam are required. The high lift system consists of the existing leading edge slat installation with slight tailoring of the outboard segment to accomodate a wing tip cut-off, existing inboard flaps, and droop of the existing aileron installation to a maximum 30 degree droop setting.

1.5.3 Landing Gear Modifications

The tri-cycle landing gear is composed largely of A-4 gear components. A-4 nose and main gear shock struts and rolling gear are used. The nose gear new drag links straddle the drive shaft in gear-up position. The A-4 nose gear telescopic links are shortened to suit required gear geometry. The main landing gear legs are composed of A-4 shock struts in new design jackets for lateral retraction supported at their lower end by a folding V-brace which carries gear drag and side loads from the lower end of the jacket to the wing rear spar and the auxiliary transverse beam.

1.5.4 Drop Box Arrangement

Parallel power train shafting is shown in Figure 1.5-3 to provide for use of spur gears at the drop box which permit low shafting location through the cockpit and bring the third engine into the power train through an HLH overrun clutch mounted on the aft face of the box. The lift fan engagement clutch is mounted on the front face of the box as are the lubrication cooler/blower unit for the clutches, gear boxes, and alternators. Common shaft segments are interconnected by flexible couplings located adjacent to shaft bearing supports to allow for airframe structural deflection.

1.5.5 Fuselage Engine Installation

The third engine is completely enclosed by a protective fire-wall. The engine exhaust duct is shrouded by a fire resistant eductor duct.

1.6 STRUCTURES

1.6.1 Structural Capabilities of the T-39 Sabreliner

No technical documentation on the T-39 was available to The Boeing Company during this contract, so the available strength of the airframe had to be deduced from published flight restrictions in Sabreliner specifications, the T-39A Flight Manual, and a detail weight statement. One of the largest unknowns was the strength margin above design load built in due to other considerations, such as fatigue or ease of manufacture. This information would normally be found in the Strength Summary and Operating Restrictions Report.

1.6.1.1 T-39A Structural Design Criteria. A summary of the T-39A structural design criteria is found in Table 1.6-1. There appear to be several variants of the airplane with gross weights up to 18,650 lbs. and the T-39A Flight Manual quotes a maximum zero fuel weight of 11,675 lbs; but the numbers in Table 1.6-1 appear to be representative of the T-39A.

TABLE 1.6-1

T-39A STRUCTURAL DESIGN CRITERIA

- o Structural design meets FAR Part 25
- o Design weights -
  - Maximum Design Weight = 17,760 lbs.
  - Design Flight Weight = 16,527 lbs.
  - Maximum Zero Fuel Weight = 10,896 lbs.
- o Design Load Factors
  - +4g to -1g at Design Flight Weight
- o Design speeds
  - Limit Speed/Mach = 450 KEAS/.85
  - Operating Speed/Mach = 350 KEAS/.80
- o Design Gust Velocity = 50 ft./sec. at operating speed
- o Design Sink Speed = 10 ft./sec. at maximum design weight

1.6.1.2 Design Wing Loads. A wing load survey was conducted to establish a wing strength level. The most likely critical conditions used in the survey are shown in Table 1.6-2. The survey used the Boeing computer program ORACLE. This program sizes a stress-designed wing structure

TABLE 1.6-2

DESIGN CONDITIONS FOR T-39A WING LOAD SURVEY

WEIGHTS

1. 16527 Lbs. c.g. at 17.5% $\bar{c}$ full wing fuel
2. 10896 Lbs. c.g. at 17.5% $\bar{c}$ zero wing fuel

MANEUVER LOAD FACTORS

n_z = +4.0 and -1.0

SPEEDS

1. Stall speed ($C_{L_{max}}$ = 1.0)
2. Limit speed

ALTITUDES

1. Sea Level
2. 20,000 Ft.

GUST CONDITION

50 ft./sec. equivalent gust velocity
350 knots equivalent airspeed
Mach = .8
Altitude = 21,100 ft.

LANDING

10 ft./sec.
Spin-up and spring-back.

which is compatible with the aeroelastic loading. Wing box geometry and material allowable stresses are input to the program. The aeroelastic solution is that of NASA TN 3030 using a rigid aerodynamic loading from the Kuchemann semi-empirical lifting surface theory. No body or nacelle interference effects were used. The weight analysis of the detail weight statement indicates that the T-39 has a loss of lift at the wing root due to the presence of the engine nacelles which results in increased root bending moments. This effect was ignored for both the basic T-39A and the demonstrator where the effect may be even greater due to the larger nacelles. Aerodynamic interactions such as these necessitate pressure model data be made available for predicting final demonstrator loads.

Conventional monocoque theory is used to predict the wing box stresses and size the bending and shear material. The survey indicated that the wing is designed by a 50 ft./sec. gust at 350 knots at 21,000 ft. The center of gravity load factor is 5.011 with zero wing fuel. The resulting wing loads are shown in Figure 1.6-1.

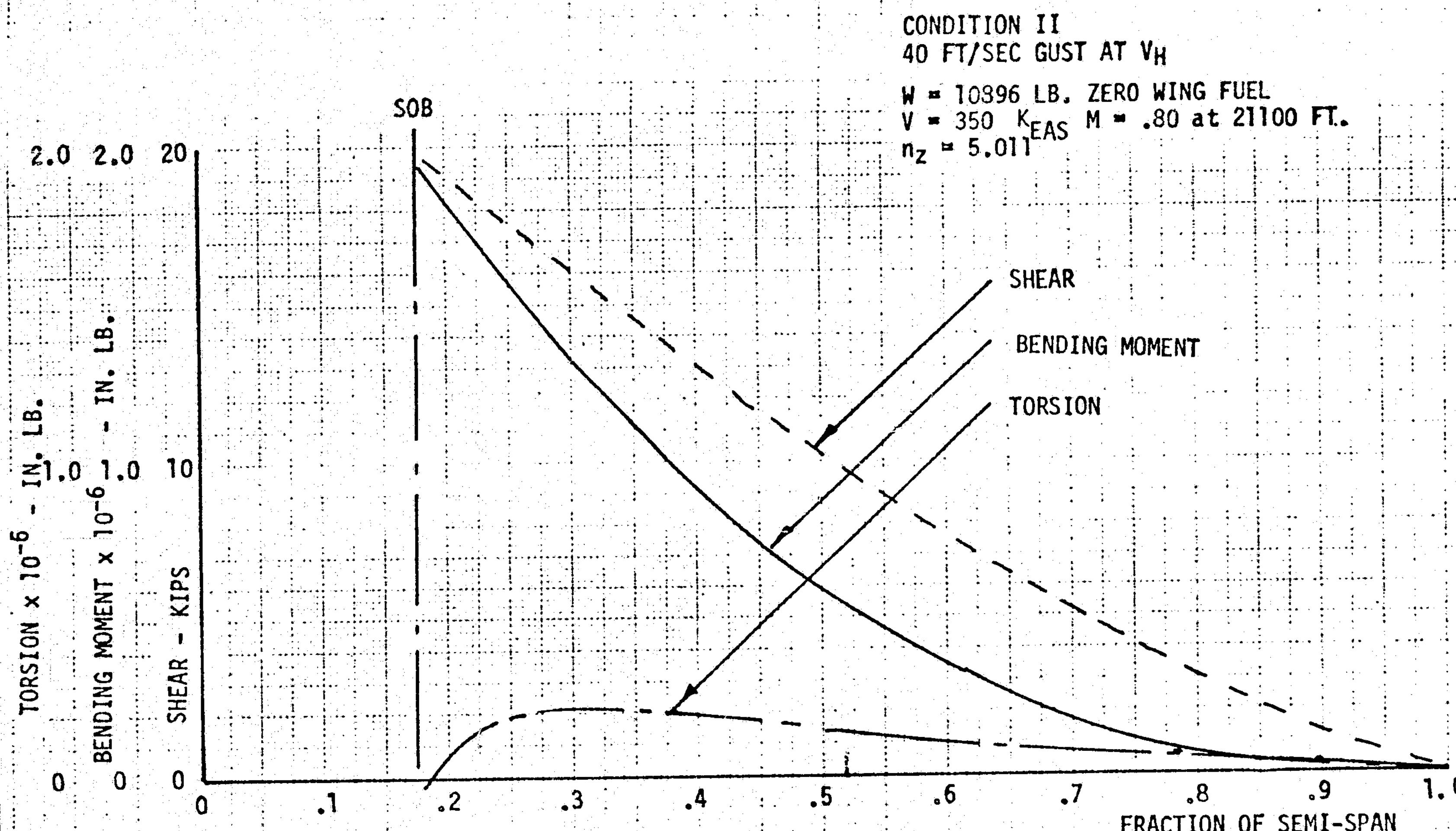
1.6.1.3 Design Body Loads. A load survey was carried out on the body to establish the probable design loads using the Boeing computer program BODYEMP. This is similar to ORACLE in that it will size the structure of both the body and empennage. Empennage load distributions are calculated using a full three-dimensional vortex representation of the vertical and horizontal tails. Conditions included in the survey are listed in Table 1.6- 3.

The maximum vertical bending loads occur during the upgust and are shown in Figure 1.6- 2. Maximum aft body torsion occurs during a lateral gust.

1.6.1.4 Design Empennage Loads. Without wind tunnel data on such a complex aerodynamic shape as the T-39A, it is not possible to calculate tail loads. Estimated tail loads are given and used in Paragraph 1.6.5.

1.6.2 Demonstrator Design Criteria

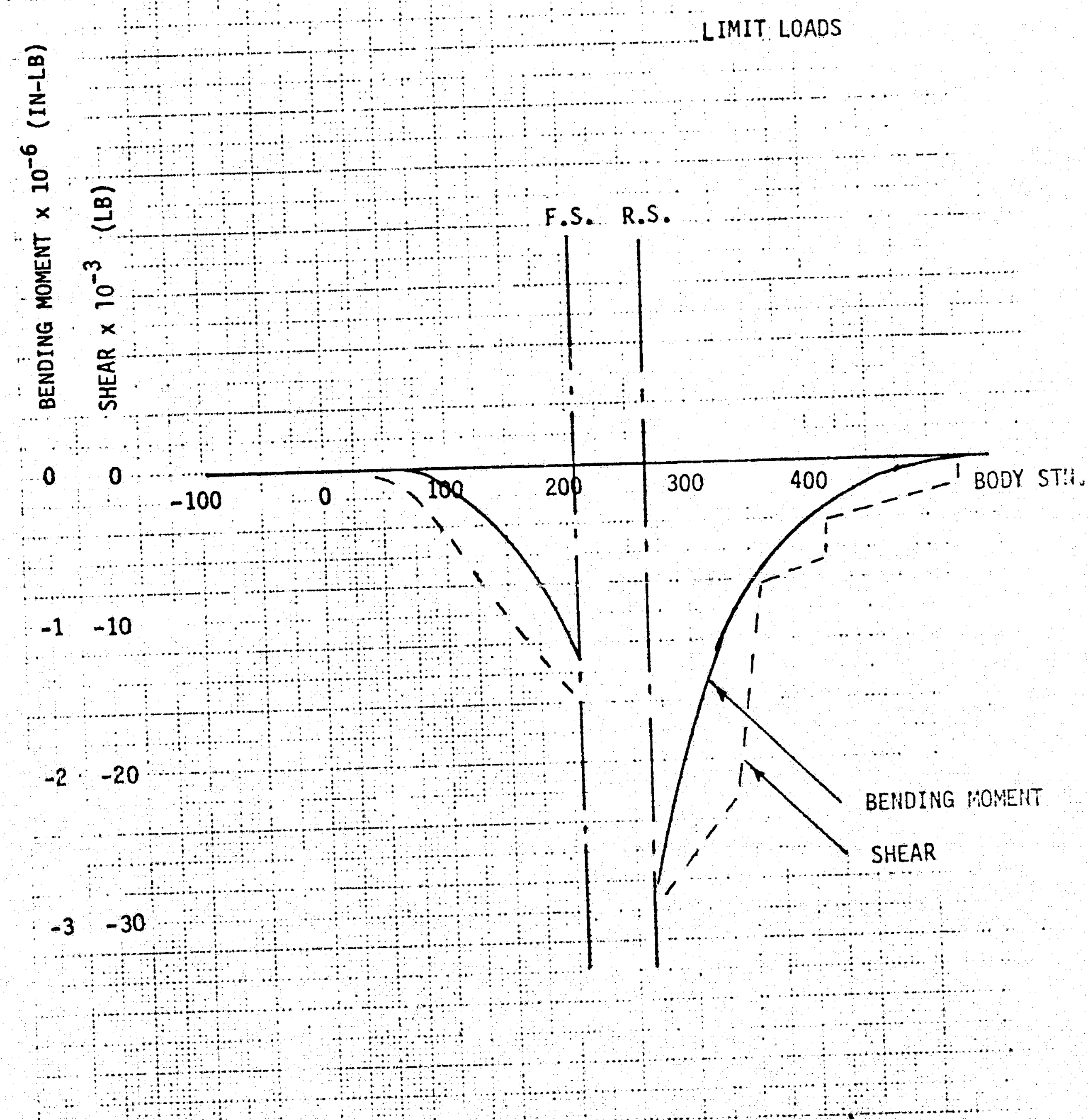
Structural design criteria have been developed for the V/STOL demonstrator in sufficient depth for evaluation of the T-39A demonstrator. The criteria permit evaluation of the airframe weight and structural flexibility. More detailed criteria must be developed in the preliminary



MAXIMUM POSITIVE WING LOADS TO ESTABLISH WING STRENGTH

FIGURE 1.6-1

CALC		REVISED	DATE
CHECK			
APR			
APR			



CALC			REVISED	DATE
CHECK				
APR				
APR				

MAXIMUM BODY BENDING
VERTICAL GUST $N_Z = 5.01$ g

FIGURE 1.6-2

T-39 TECHNOLOGY DEMONSTRATOR OPERATING ENVELOPE

(1041-135-2A)

GROSS WT = 25050 LB.

90^{o}F DAY

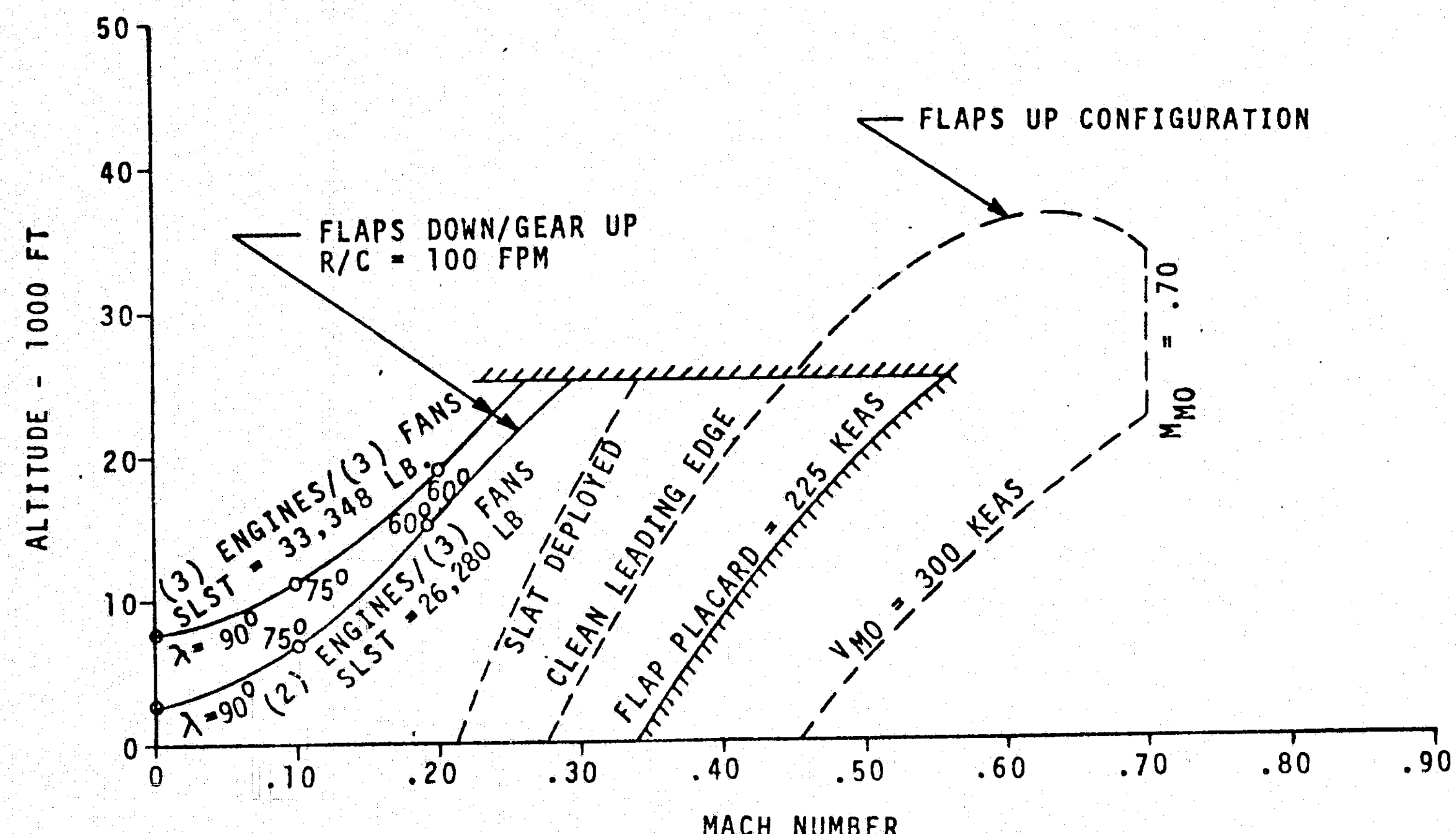


FIGURE 1.6-3

design phase and several outstanding requirements resolved. Two of these concern landing. First, the short landing and conventional landing impact conditions must be defined, so that spin-up and spring-back conditions can be evaluated. Second, drift landing conditions, both for vertical and short landings, must be defined. Suggested criteria are found in Reference 3 which is still the most comprehensive study of V/STOL landing criteria.

1.6.2.1 Basic Structural Philosophy. The demonstrator modifications will be strength checked to ultimate loads which are 1.5 times limit loads predicted from the design criteria. This applies to structure which has demonstrated a strength level in previous static tests. New structure will be designed to an additional 15% margin. A Safety Review Board will convene to ascertain design suitability for airworthiness. This approach was used successfully on the Buffalo modification and is being used on QSRA.

TABLE 1.6-3.

DESIGN CONDITIONS FOR T-39A BODY LOAD SURVEY

Upgust	$n_z = 5.011$
Downgust	$n_z = -1.858$
2g Taxi	
3 Point Braked Roll	
Lateral Gust (U_{de} = 50 ft./sec.)	
M = .8 at 21,100 ft.	

1.6.2.2 Flight Criteria. Maneuver and gust requirements are summarized in Table 1.6.4 . The speed/altitude envelope is found in Figure 1.13-1. A 20% flutter margin above limit speed will be used. Flight flutter tests will demonstrate positive damping out to limit speed.

1.6.2.3 Ground Criteria. Landing criteria are summarized in Table 1.6-5. Ground operation criteria follow the requirements first promulgated in ANC-2 and currently found in MIL-A-8863A.

It can be seen from Table 1.6-5 that the STOL landing could be more severe than VTOL because the STOL mission requires more fuel for the 11 circuits than the VTOL 5 circuit mission, and the gear is subject to spin-up and spring-back loads not present in vertical landings. This is discussed further in Paragraph 1.6.6.

TABLE 1.6- 4

FLIGHT DESIGN CRITERIA FOR T-39A DEMONSTRATOR

Limit Load Factor = 2.5 to -1.0 at Flight Design Weight
Flight Design Weight = 26410 lbs.
(Includes payload, crew, 60% STOL and fuel)

Limit Load Factor = 2.0 to -1.0 at Maximum Design Weight
Maximum Design Weight = 30190 lbs.
(Includes payload, crew, full fuel)

Design Gust Velocity = 50 ft./sec. (EAS) at Cruise
Speed (V_H)

1.6.3 Demonstrator Loads

1.6.3.1 Wing Loads. A load survey was conducted to establish the maximum loads resulting from the above criteria. Conditions considered are given in Table 1.6- 6. The aeroelastic loadings were based on the stiffness given by the load survey on the basic T-39A. The critical condition proved to be a 2.5g maneuver with zero wing fuel. The resulting loads are found in Figure 1.6- 4. Gust loads are of similar magnitude and are not so critical as for the T-39A because of the reduced cruise speed and Mach number. This situation could change when T-39A wind tunnel data becomes available since the present study is based on the Prandtl-Glauert compressibility correction.

A comparison of figures 1.6-1 and 1.6-4 shows a 28% increase in wing root bending moment for the demonstrator. Required reinforcements are discussed in Paragraph 1.6.4, but an alternate solution would be to reduce the wing span so that adequate bending strength would be available in the T-39A structure. Wing loads were, therefore, calculated on the T-39A wing with two feet and four feet removed from each wing tip. From this,

TABLE 1.6- 5

GROUND DESIGN CRITERION FOR T-39A DEMONSTRATOR

Vertical Landing - Translation Velocity TBD.

Sinking Speed = 12 ft./sec. at Design Vertical Landing Weight.

Design Vertical Landing Weight = 25500 Lbs.

(Includes payload, crew, VTOL mission fuel less one takeoff)

Sinking Speed = 8 ft./sec. at Maximum Vertical Landing Weight

Maximum Vertical Landing Weight = 29840 lbs.

Short Landing - Translational velocity TBD

Sinking Speed = 12 ft./sec. at Design STOL Landing Weight

Design STOL Landing Weight = 27650 Lbs.

(Includes payload, crew, STOL mission fuel less one takeoff)

Conventional Landing - Forward Velocity = 1.3 V_{SPA}

(V_{SPA} is the stall speed in the powered-approach configuration.

Sinking Speed = 10 ft./sec. at Design CTOL Landing Weight

Design CTOL Landing Weight = 26536 lbs.

(Includes payload, crew, 40% internal fuel)

Sinking Speed = 6 ft./sec. at Maximum CTOL Landing Weight

Maximum CTOL Landing Weight = 29840 Lbs.

(Maximum Design Weight less fuel used in one takeoff & circuit.)

TABLE 1.6-5

WING DESIGN CONDITIONS CONSIDERED FOR THE T-39A DEMONSTRATOR

WEIGHTS

1. Flight Design Weight = 26410 Lbs.
2. Zero Fuel Weight = 24100 Lbs.

MANEUVER LOAD FACTORS

n_z = 2.5 and -1.0.

SPEEDS

1. Stall Speed ($C_{L_{max}}$ = 1.0)
2. Limit Speed

ALTITUDES

1. Sea Level
2. 20;000 Ft.

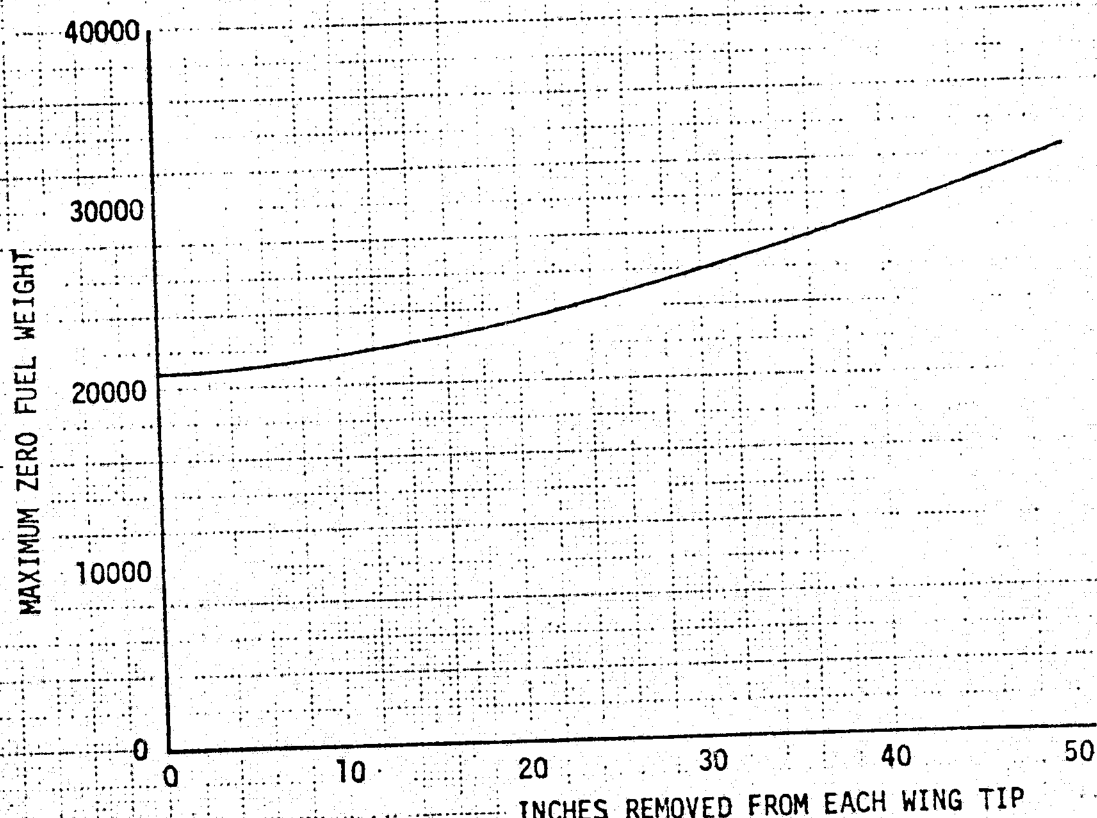
GUST CONDITION

50 ft./sec. equivalent gust velocity

300 knots equivalent airspeed

Mach = .7

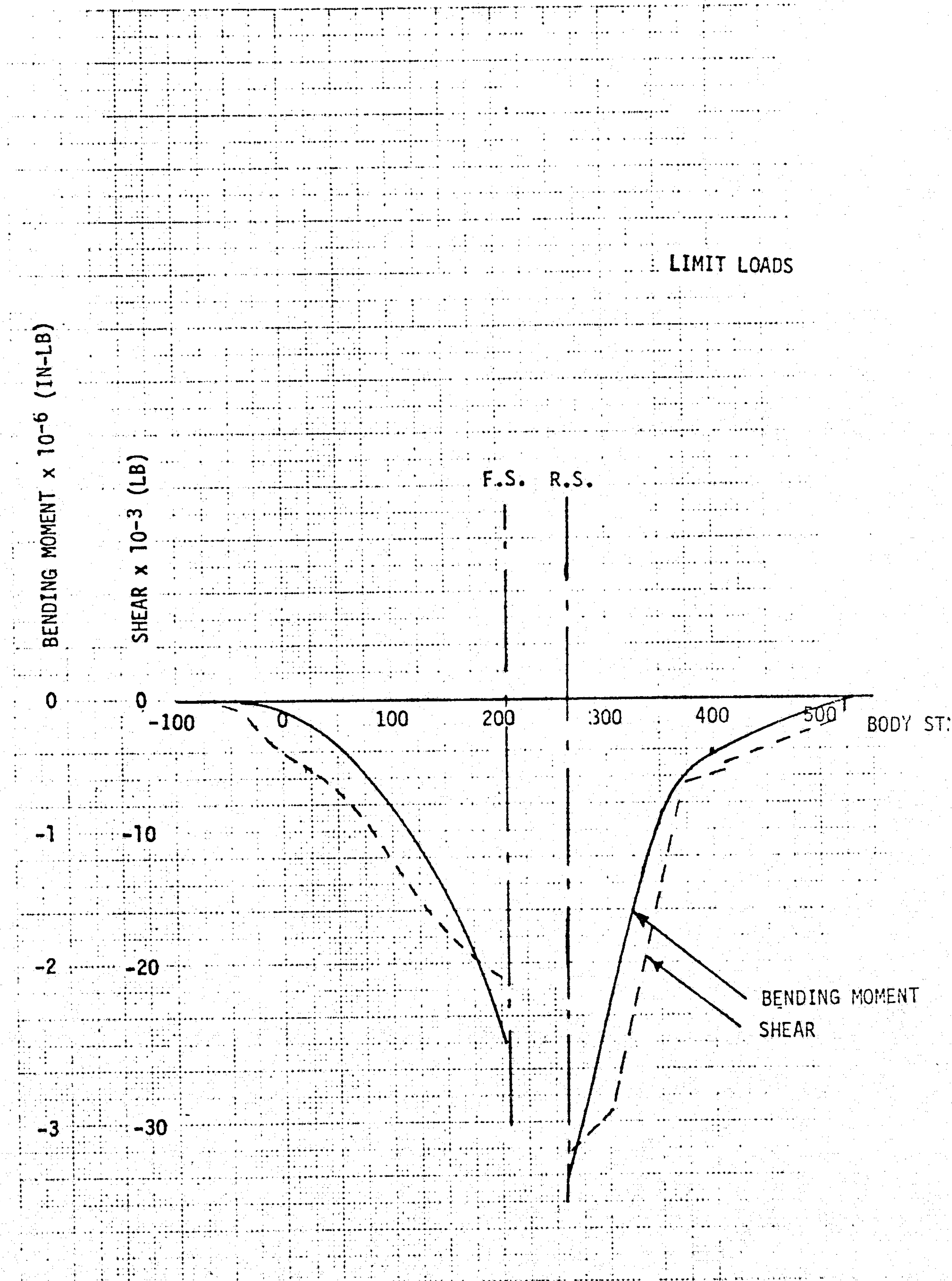
Altitude = 22,100 ft.



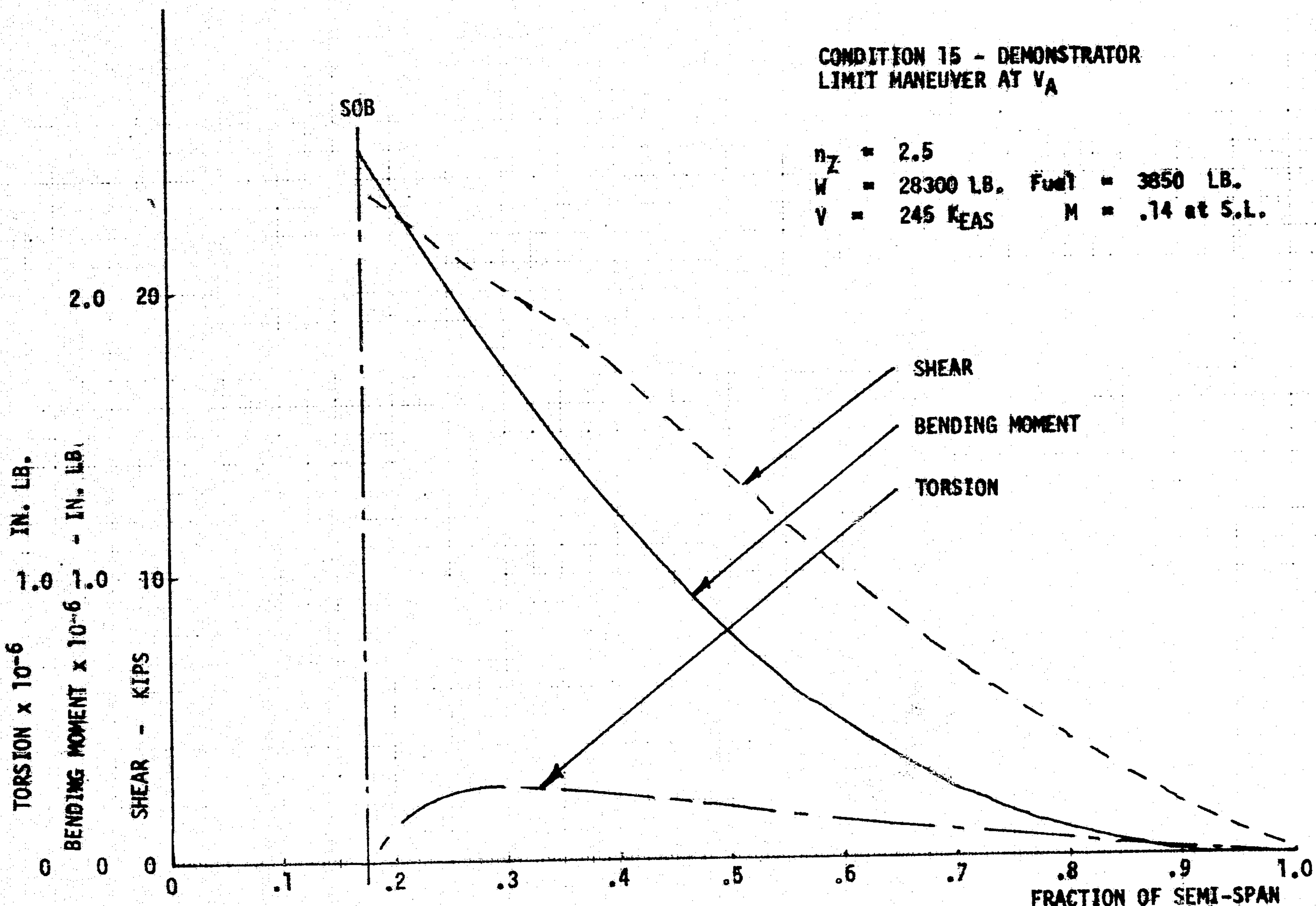
			REVISED	DATE	EFFECT OF SEMI-SPAN ON ZERO FUEL WEIGHT TO EQUAL T-39A ROOT BENDING STRENGTH AT 2.5 g
CALC					
CHECK					
APR					
APR					

FIG. 1.6-

K-E



CALC			REVISED	DATE	MAXIMUM BODY BENDING VERTICAL GUST $N_Z = 2.61$	FIGURE 1.6-5
CHECK						
APR						
APR						



MAXIMUM POSITIVE WING LOADS REQUIRED FOR DEMONSTRATOR

FIGURE 1.6-6

the maximum allowable zero fuel weight at which a 2.5g maneuver could be performed without exceeding the T-39A wing root bending strength was computed. The results are shown in Figure 1.6-5.

1.6.3.2 Body Loads. Body load survey conditions are presented in Table 1.6-7. Critical vertical bending loads still occur during gust encounter and are shown in Figure 1.6-6. It should be noted that the 15 ft./sec. landing conditions are lower than the gust loads provided that the lift fans are still maintaining the 1g upward force. The impact of these loads on the body structure is discussed in Paragraph 1.6.4. The aft body torsion due to fin gust is increased by 157% in spite of the reduced cruise speed since the induced rolling moment on the horizontal tail is additive to the fin torsion when a T-tail configuration is used.

TABLE 1.6-7

BODY DESIGN CONDITIONS CONSIDERED FOR THE T-39A DEMONSTRATOR

Upgust. $n_z = 2.61$

Negative Maneuver $n_z = -1.0$

Landing Impact - 2 point - $n_z = 2.9$*

Landing Impact - 3 point - $n_z = 2.9$*

2g Taxi

3 Point Braked Roll

Unsymmetrical Braking

Lateral Gust ($U_{de} = 50$ ft./sec.)

M = .7 at 10,000 ft.

* Additional conditions run at 4.0g to simulate 15 ft./sec. landing.

1.6.4 Structural Modifications

1.6.4.1 Wing Structural Description. The wing is a full-cantilever, semi-monocoque, structure tapering in planform and depth. The primary structure is a single cell box beam which forms integral fuel tanks. The upper and lower skin panels are integrally tab stiffened panels machined from an aluminum plate. The spars are a combination of forgings and machined plate. The structural capability of the wing was determined by using the "ORACLE" computer program.

1.6.4.2 Wing Modification Alternatives. The increased gross weight of the V/STOL demonstrator necessitates changes to the wing or to the flight envelope. The following alternatives were studies:

A) Wing structural reinforcement to meet the increased flight loads (Figure 1.6-1)
B) Reduce wing area and span by cutting off wing tips
C) Placard the maneuver load factor and the cruise speed

1.6.4.2.1 Reinforcement Method. A study was made to determine the feasibility of reinforcing the wing primary structure to carry the increased bending moments, shears, and torsions. The increased bending moment could be accommodated by an external strap at one or both of the spar locations. However, at the wing root, a more difficult modification is necessary at the terminal fittings. The increased shear and torsion also requires increased spar gages. Generally, this approach, of structural modification, appears to be difficult and necessitates the tear-down of the wing primary structural box.

1.6.4.2.2 Tip Removal Method. By removing a portion of the wing tips, the wing loads are reduced. Figure 1.6-4 shows a plot of the length cut off the tip versus the allowable gross weight assuming a 2.5g (limit) maneuver load factor. The plot shows approximately 22 inches off each tip result in wing loads not exceeding the structural capability. This alternative may require modification of the wing aileron.

1.6.4.2.3 Placards Method. Two approaches to flight placards were studied. First, by reducing the maneuver load factor to 2.0g (limit) the gross weight could be increased to 25,394 pounds. However, at the higher gross weights, the airplane will become gust critical and may have to be limited in cruise speed or restricted to flying in smooth air. Retaining the 50 ft./sec. gust capability would require restricting the cruise speed to 185 knots at a zero fuel weight of 25,000 pounds.

1.6.4.3 Fuselage Structural Description. The body is a four longeron type structure using chem-milled skins stiffened by frames at 8 - 11 inch spacing. Bulkheads are located forward of the crew compartment, forward and aft of the main entry door, at the wing front and rear spar locations, at the aft pressure bulkhead (also supports the engine) and in the aft fuselage section, supporting the vertical and horizontal stabilizers.

1.6.4.3.1 Fuselage Modification. Incorporating the engine installation, the forward fan, and the backup structure necessary for the vertical tail will require significant body structural modifications. In addition, the body loads shown in Figure 1.6-2 have increased. Additional strength will be incorporated into the modified areas as a result of the redesign. In the body sections that can be salvaged, the increased bending moments will be taken by using external longerons back-to-back with the existing longerons. The shear and torsion load increases will be handled by skin age increases, as required. Since body pressurization is no longer a requirement, excess shear material is available which may take the increased loads. Without a detailed structural analysis of these body sections, skin gage specifications cannot be made at this time.

1.6.4.4 Empennage Structural Description. The T-39 empennage consists of a vertical tail and low horizontal tail, both mounted off the aft body. The primary structure of the vertical tail is a single cell torque box. The two-span structure has moment continuity at the rear spar and a shear correction at the front spar. The side skins are lightly loaded.

1.6.4.4.1 Empennage Modification. The horizontal tail has been relocated to the top of the vertical tail on the T-39 V/STOL demonstrator to avoid interference with the aft lift fans. This T-tail configuration results in significantly higher loads in the vertical tail structure. Several design modifications have been investigated to accommodate the required structural reinforcement.

A finite element structural model of the vertical tail primary structure was prepared. The model was used to determine the distribution of reactions at the front and rear spars assuming various boundary conditions (Case I, moment capability - front spar and rear spar; Case II, moment capability - rear spar only, shear capability - front spar only; and Case III, moment capability - front spar, shear capability - rear spar). The results of this analysis are presented in Tables 1.6-8 and Figures 1.6- 7, 8 and 9. With these results, and using the root bending moment on the basic T-39 as the structural capability, one can conclude that Cases I or III above will not require modification of the rear spar to body bulkhead attachment. However, Case III does show high reactions and does not use the existing structural capability of the aft body between B.S. 412 and 457.4. Therefore, it is recommended to provide moment continuity at both the front and rear spars, and to provide reinforcement to the present T-39 inspar structure.

1.6.4.5 Landing Gear Modification Description. The demonstrator will not use the T-39A gear. A longer stroke gear from the A-4E is proposed to absorbe the increased sink rate at higher gross weights. The T-39 gear attachment points cannot be used since larger wheel wells are needed for the longer gears. Lacking drop test data on either gear, a qualitative assessment of the practicality of the substitution was made to determine suitability.

1.6.4.5.1 Landing Gear Structural Description. A preliminary evaluation was made of the landing gear system for the Model 1041-135-2A. The conclusions were that the A-4 Nose and Main Landing Gears provide major components for a gear of adequate length and stroke for providing Power Pod ground clearance and a sink rate of 12 ft./sec. without undue beef-up of the basic T-39 airframe.

The proposed nose gear consists of A-4 nose gear fork, wheel, axle, tire, shimmy damper or steering cylinder and, shock strut with metering pin modification mounted in reinforced existing T-39 nose gear support beams. Aft retraction is to be accomplished by knuckled drag bracing struts which straddle the front fan drive shaft when retracted. A modified length A-4 gear telescoping link is to be included to reduce gear length in the retracted position, thus minimizing airframe modification.

The proposed main landing gear concept consists of the A-4 wheel, tire, tube, brake, and shock strut assembly with modified metering pin, installed in a new design "support jacket." The A-4 gear swiveling

REACTIONS - VERTICAL TAIL SAMECS* MODEL Table 1.6-8

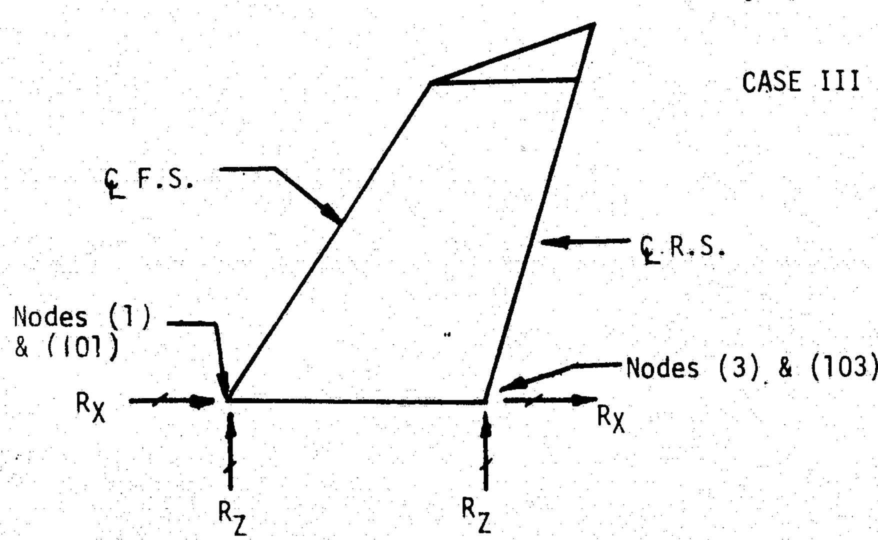
(T-39, Modified)

NODE NO.	POSITION	R_X	R_Y	R_Z	
1	F.S. - Right Side	36726	-2449	51264	CASE I
101	F.S. - Left Side	-45096	-3227	-62060	
3	R.S. - R.S.	5694	1811	48758	
103	R.S. - L.S.	3309	3865	- 7689	
1	F.S. - CTR	- 6890	- 951	-10659	CASE II
3	R.S. - R.S.	22466	- 577	112907	
103	R.S. - L.S.	-11942	1528	-71975	
1	F.S. - R.S.	53968	-3629	76077	CASE III
101	F.S. - L.S.	-61832	-4504	-86649	
3	R.S. - CTR	11497	8133	40845	

CASE I - F.S. & R.S. - Fixed

CASE II - F.S. - Pinned
R.S. - Fixed

CASE III - F.S. - Fixed
R.S. - Pinned



* Structural Analysis Method for Evaluation of Complex Structures

Table 1.6-8 Continued

REACTION MOMENTS

$$M_X = (\text{Depth}/2)\ (R_{Z_{(1)}} + R_{Z_{(101)}})$$

$$M_Z = (\text{Depth}/2)\ (R_{X_{(1)}} + R_{X_{(101)}})$$

M_X

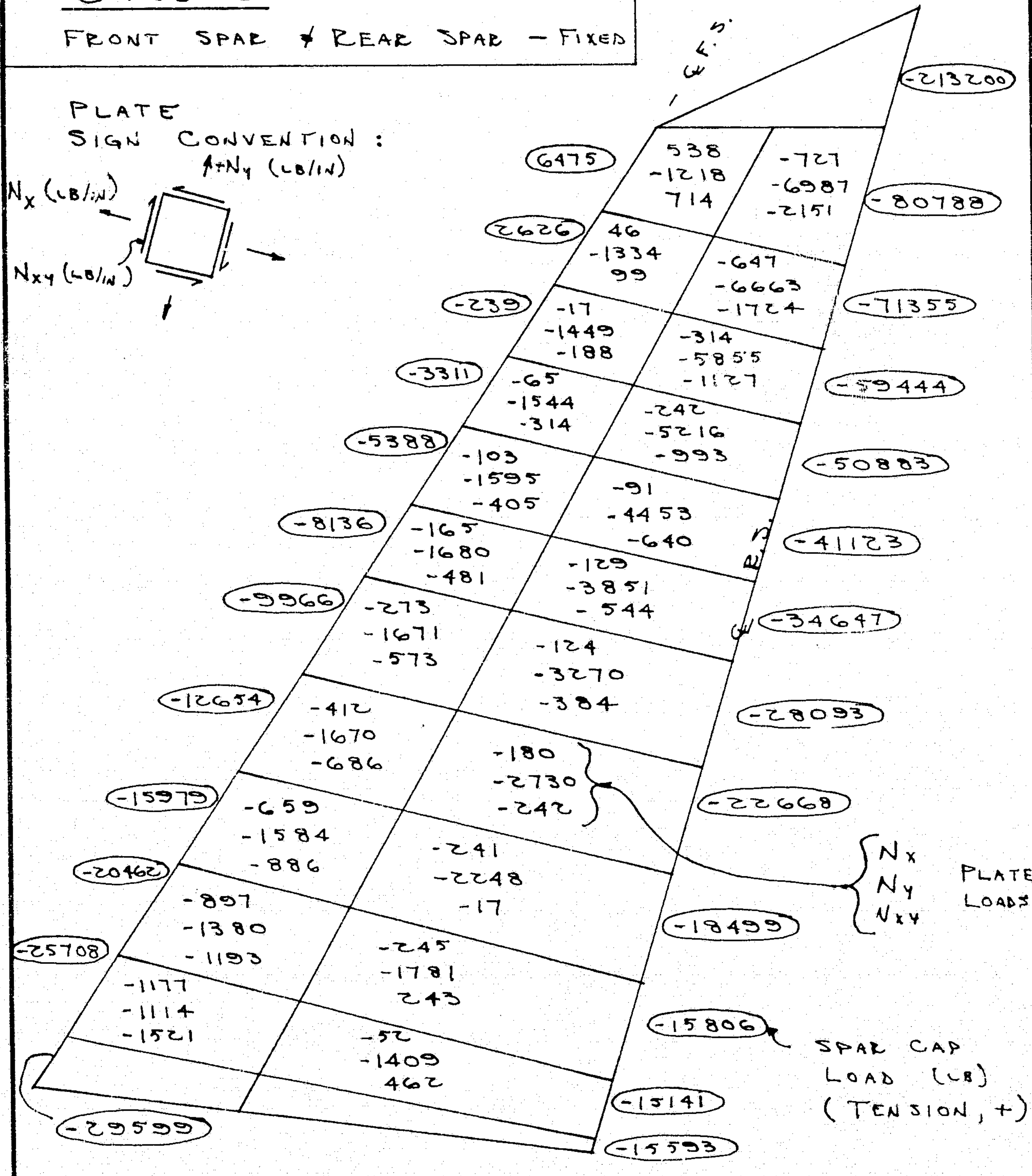
CASE	LOCATION	DEPTH/2	M_X (In.-Lb.)	
I	F.S	3.8	430631	$\left.\begin{matrix} \\ \end{matrix}\right\}$ 628195
I	R.S.	3.5	197564	
II	R.S.	3.5	647087	
III	F.S.	3.8	618359	

M_Z

CASE	LOCATION	DEPTH/2	M_Z(in.-Lb.)
I	F.S.	3.8	310924
I	R.S.	3.5	31510
II	R.S.	3.5	120428
III	F.S.	3.8	440040

CASE I

FRONT SPAR & REAR SPAR – FIXED



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE		MODEL
CALC	GK	9/7/76			MODIFIED T-39	FIGURE 1.6-7	
CHECK					VERTICAL TAIL		
APPD.					SAMECS OUTPUT		
APPD.							

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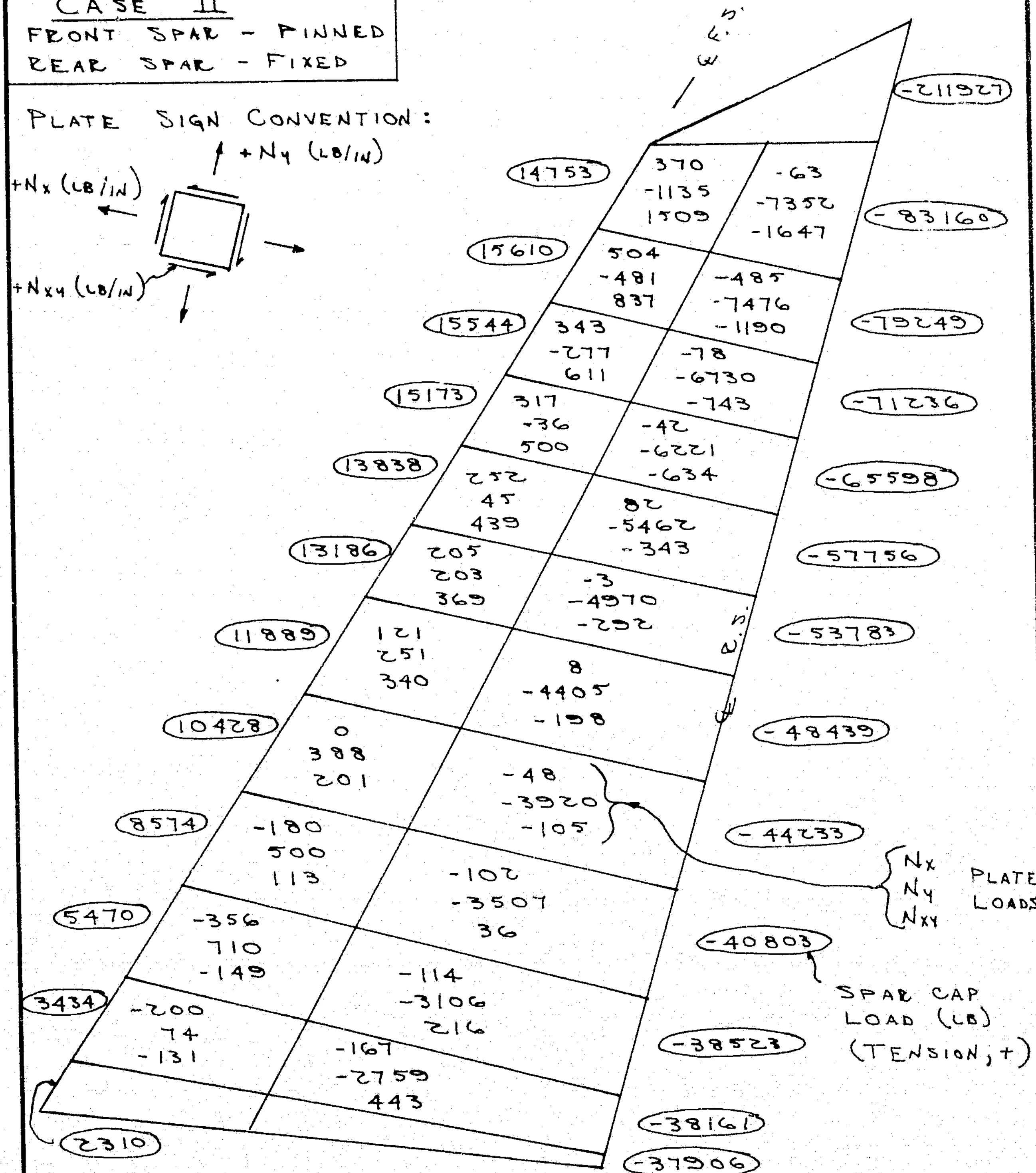
CASE II

FRONT SPAR - PINNED

REAR SPAR - FIXED



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE	MODEL
CALC	GK	9/7/76			MODIFIED T-39 VERTICAL TAIL SAMECS OUTPUT — FIGURE 1.6-8	
CHECK						
APPD.						
APPD.						

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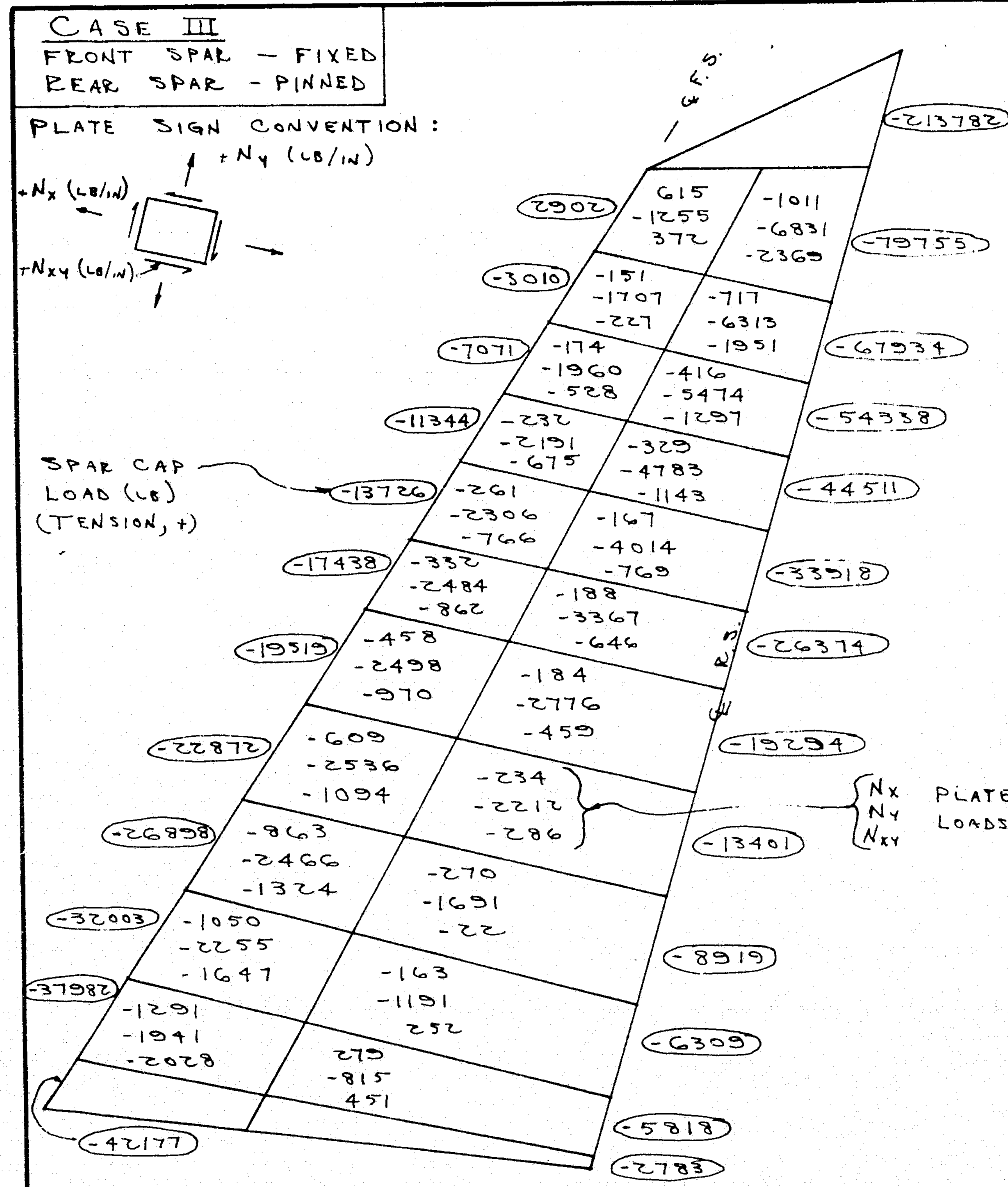
OF POOR QUALITY



	INITIALS	DATE	REV BY INITIALS	DATE	TITLE		MODEL
CALC	GK	9/7/76			MODIFIED T-39 VERTICAL TAIL SAMECS OUTPUT	FIGURE 1.6-9	
CHECK							
APPD.							
APPD.							

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mechanism is to be eliminated as inboard retraction to fully enclose the wheels within the T-39 wing/body contour is proposed.

The support jacket is to be hinged at an upper end fitting, sandwiched between the swept rear spar and an added auxiliary transverse beam extending through the body to the opposite wing rear spar. A trailing edge wing/body fillet is to be added to provide reasonable depth for the auxiliary beam. The lower end of the support jacket is to be supported at the apex of folding a vee-brace which transfers gear side load drag load and torque to the body structure.

The following functions were considered in the evaluation:

- o Shock absorption
- o Braking
- o Steering
- o Retraction/Extension

The A-4E landing gear parameters and the pertinent requirements of the RTA are shown in Table 1.6-9 . The shock strut stroke versus airplane rate of sink is shown in Figure 1.6-10. At (12 fps) the estimated peak landing load factor is under 2g's; at (15 fps) the peak load factor is just under 3g's. The T-39 airframe is designed for a load factor of 3 indicating this gear may be adequate for sink speeds to 15 fps. However, because of the greater length of this gear, the limit in this case will be influenced by spin-up loads and method of attachment of the gear to the wing structure.

The static load-stroke curves for the main and nose gear struts are shown in Figures 1.6-11 and 1.6-12 respectively. These curves indicate that the pressure in the main gear strut (fully extended) will have to be increased from the (25 psig), used on the A-4E and that of the nose gear should be reduced from the current (210 psig). Optimum charge pressures and metering pin shape will be determined by dynamic analysis during future detail design.

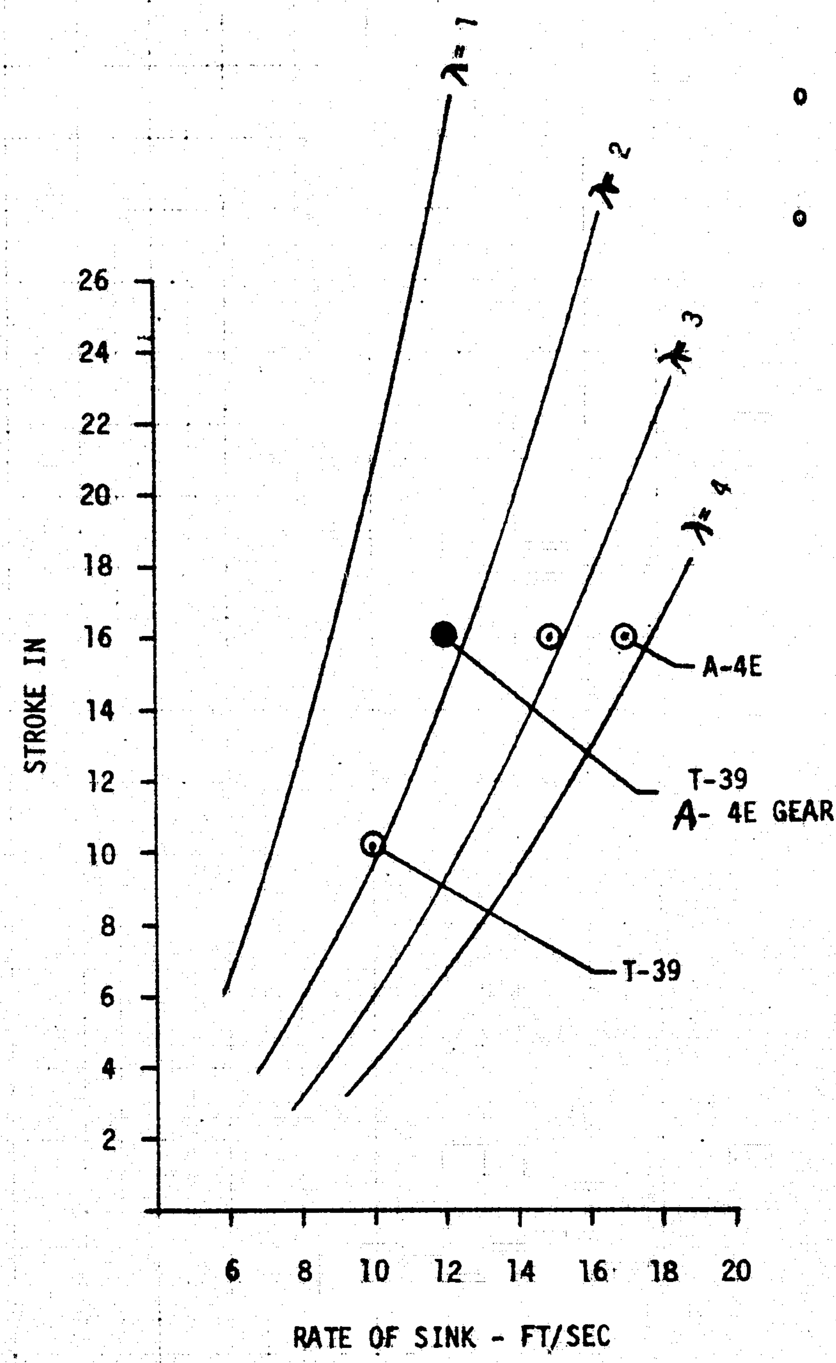
Two different wheels and brakes are available for the A-4E; a single rotor brake with a shallow cavity wheel and a two rotor brake with a deep cavity wheel. Either brake has adequate energy capacity for the RTA.

TABLE 1.6-9 LANDING GEAR PARAMETERS

	A-4E			RTA MODEL 1041-135-2A	
Max. T.O. GW	27,420 Lb			27,500 Lb.	
Max Ldg Wt.	11,556 Lb			25,050 Lb.	
Max Rate of Sink	17 Fps			12 Fps (Consider 15 fps)	
Shock Strut	Main		Nose	Main	Nose
o Stroke	16 in		17.88 in.		
o Piston D/A	3.75 in		3.75 in		
o Extend Press.	25 psig		210 psig		
	PD 476-15	PD 824-8			
Wheels					
o P/N	3532234	9541181	BFG3-1128		
o Static Load	8300 Lb	1,523		12000 Lb	4720 Lb
o Weight	36.2 Lb	29 Lb.			
Brake:					
o P/N	9542024	9541299	-	Taxi Energy	-
o Normal KE	2.46 MFP	3.65 MFP	-		
o Max. RTOKE	6.85 MFP	9.3 MFP	-	3.0 MFP	-
o Operating Press.	600 PSIG	600 PSIG	-	3000 PSIG	
o Static Braking Force	?	4400 Lb	@ 588 Psi	6000 Lb ("Hold" Airplane with one fan at full the	
o No. of Rotors	One	Two	-		
o Weight	31.6 Lb	47.9 Lb	-		

TABLE 1.6-9 LANDING GEAR PARAMETERS (Continued)

	Main	Nose	Main	Nose
Tire				
o Type	24 x 5.5	18 x 5.7		
o Ply Rating	16	14		
o Static Load/ Press	11,500/1355 PSIG	6200/1215	12000	4720
o Bottoming Load	32,000 Lb	17,300 Lb.		
o Max Dynamic Load		9,200	-	11250 Lb

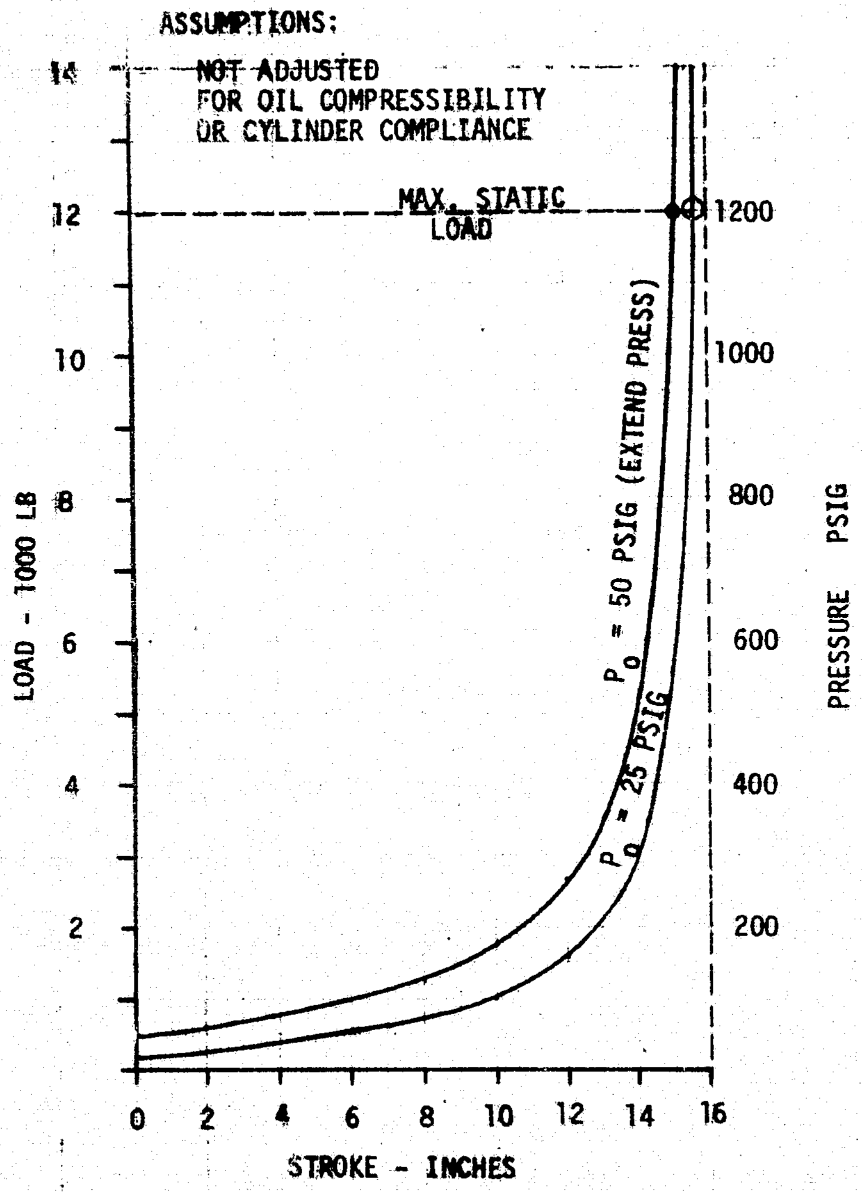


ASSUMPTIONS:

- STROKE INCLUDES 1 INCH ALLOWANCE FOR SERVICING ERROR
- 4 INCH TIRE DEFLECTION ASSUMED
- AIRPLANE LIFT EQUALS WEIGHT STRUT EFFICIENCY IS 85%

			REVISED	DATE		
CALC					SHOCK STRUT STROKE VS RATE OF SINK	FIGURE 1.6-10
CHECK						
APR						
APR						PAGE

MAIN GEAR
A-4E GEAR ON
T-39 TECH DEMO



CALC			REVISED	DATE	MAIN GEAR LOAD - STROKE STATIC CONDITION	FIGURE 1.6-11
CHECK						
APR						
APR						PAGE

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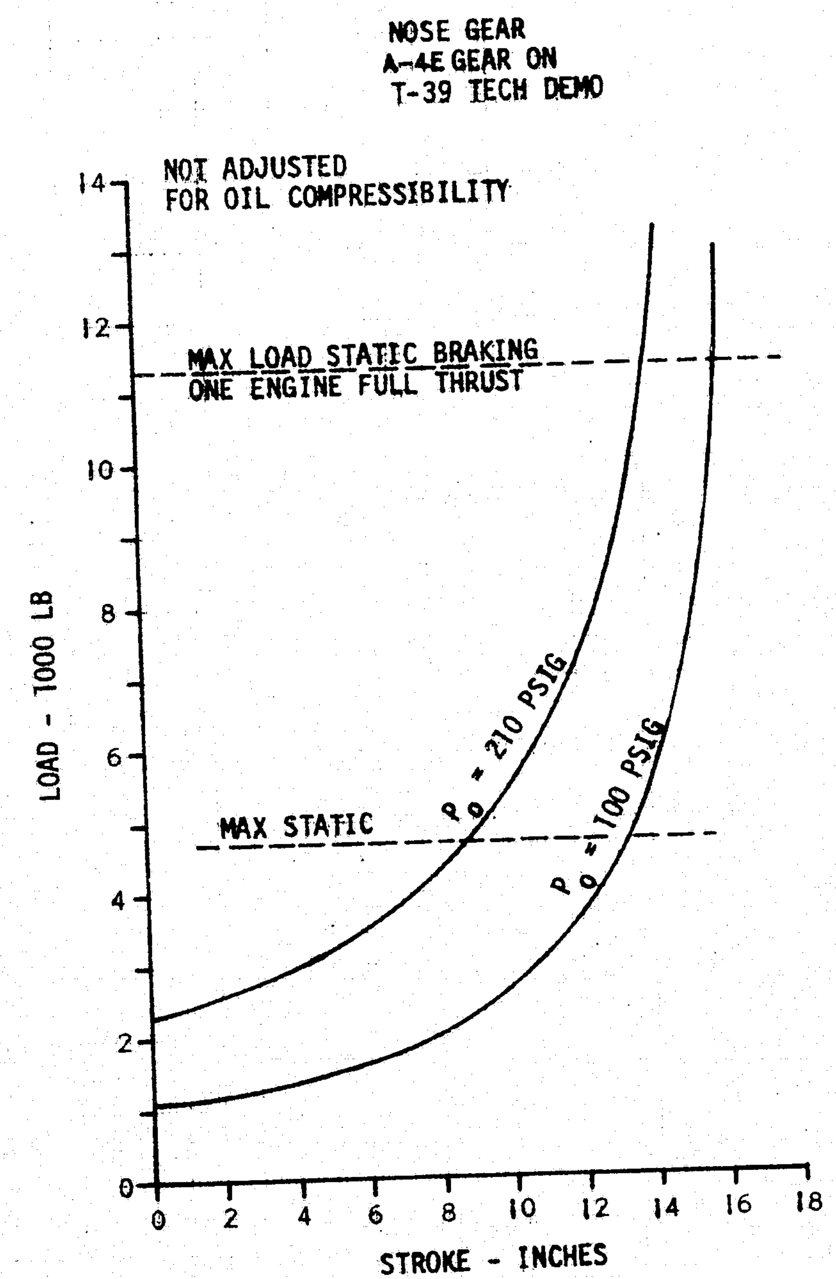

			REVISED	DATE		FIGURE 1.6-12
CALC					NOSE GEAR LOAD - STROKE STATIC CONDITION	
CHECK						
APR						PAGE
APR						

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However, the higher torque capacity of the two rotor brake and the higher static load rating of the deep cavity wheel may be required. Also, the two rotor brake may have to be qualified to a higher torque level, i.e., increased pressure, assuming the brakes must hold the airplane static with one fan at full thrust.

Deboost valves will be required in the brake hydraulic lines since the A-4E brakes are designed for low pressure (less than 1000 psig) and the T-39 system is (3000 psig). An alternate means of achieving a match between the brake and the hydraulic system would be to reduce the brake piston size. This can be done if the piston housing can be qualified to a working pressure (3000 psi).

The A-4E does not have a nose gear steering system (steering is accomplished by differential braking). However, a steering actuator can be installed in place of the shimmy damper. The T-39 steering system is controlled by rudder pedal movement. If rudder pedal steering is required on the RTA, a dynamic analysis of the system will have to be made to determine interface requirements and insure stability of the gear and steering system. When possible, up/down locks, door latches, retraction/extension actuators, gear position indicators, gear up warning system and door actuation linkages, will be of components common to the T-39 or A-4 gear. A detailed kinematic analysis of the landing gear retraction/extension system will be made during detail design to determine actuator sizes and structural clearances.

1.7 Systems

1.7.1 Accessory Power

The primary power source for the airplane accessories is the accessory drive gear box located in front of, and integrated with the transmission drop box (see Figure 1.5-3). The accessory drive gear box will drive two 20 KVA integrated drive generators, two 30 GPM hydraulic pumps, two gear box lubricator pumps and a cooling fan. Each hydraulic pump will supply a separate hydraulic system to provide redundant actuation for the flight controls. The hydraulic system is shown schematically in Figure 1.7-1.

The electrical power supplied by two 20 KVA integrated drive generators provides two separate power systems. An emergency power unit will be installed to provide emergency electrical and hydraulic power for conventional landings in the event of a gear box failure.

1.7.2 Starting System

The ground starting system will provide for independent engine starting. In addition, the airplane weight allowances are sufficient to support the installation of a reliable inflight start system.

1.7.3 Environmental Control

Crew and equipment environmental cooling will be provided from the existing air conditioning pack and/or ram air. Aircraft pressurization will not be provided.

1.7.4 Oil Cooling

Preliminary analysis indicates that power train gear boxes and clutch oil cooling requirements can be met using two existing air coolers and a blower (Boeing HLH unit) which is the baseline oil cooling system. A description of this system is found in Section 2 of this report.

1.7.5 Fuel System

The basic T-39 fuel tank and system will be modified to accommodate a

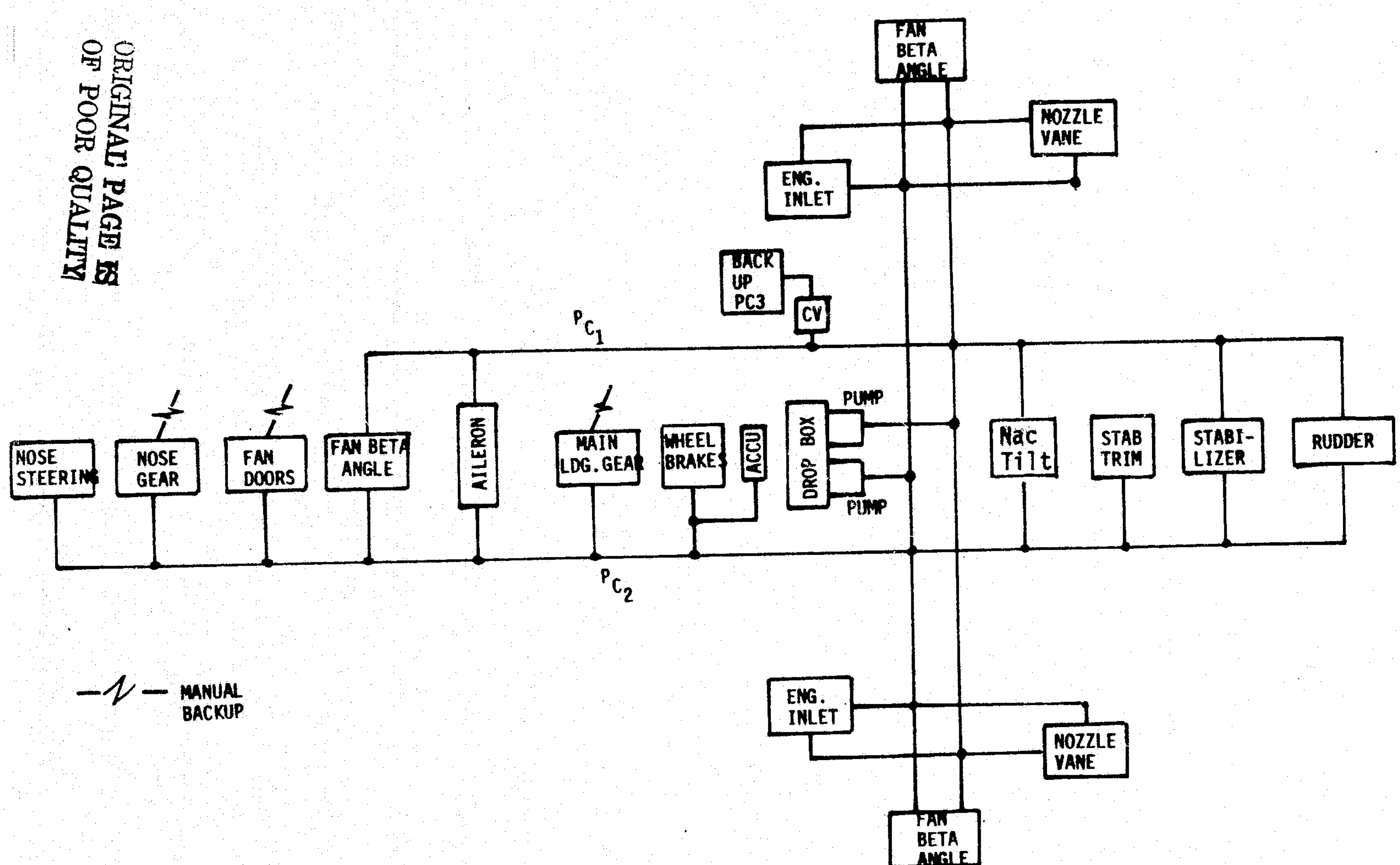


FIGURE 1.7-1 HYDRAULIC SYSTEM SCHEMATIC

three engine configuration. The existing T-39 fuselage tank will be removed, and fuel for the technology demonstrator will be carried in the existing internal wing tanks.

1.7.6 Oxygen System

A five liter liquid oxygen system will be installed to provide sufficient oxygen for both crew members for a two hour mission.

1.8 EMERGENCY ESCAPE SYSTEM

The crew escape system for the RTA provides through-the-canopy, simultaneous ejection of the two seats. This system will be used throughout the operating range of the aircraft.

1.8.1 Requirements

The operating conditions are as follows:

Velocity range	0-154 m/s (0-300 knots)
Altitude range	0-3048m (0-10,000 ft.)
Maximum pitch rate	0.6 rad/sec
Maximum roll rate	1.0 rad/sec
Maximum inflight load factors	2-5g's
Maximum crash load factors	40g's

System performance requirements are listed below in Table 1.8-1.

TABLE 1.8-1

ESCAPE SYSTEM PERFORMANCE

SINK RATE (fpm)	ALTITUDE (ft)	VELOCITY (knots)	PITCH ANGLE (deg)	ROLL ANGLE (deg)
0	0	0	0	0
2500	50	100	0	0
5000	100	100	0	0
2000	100	100	0	90
5000	200	100	0	90
0	200	130	0	180
5000	120	130	20	0

The installation requirements are dictated by the general configuration as shown in Figure 1.5-1. The side-by-side installation of the pilot and co-pilot provides the following installation dimensions:

- o Buttock line of seat centerline $\pm$BL15
- o Design eye point WL45

Other requirements and conditions are as follows:

- o The seat must clear the tail which is 330 inches aft of the cockpit and 90 inches above the canopy.
- o The drogue must clear the engine inlet flow field with the engines in the vertical and the horizontal position.
- o The canopy is made of 1/4" cast acrylic which must be fractured by breakers on the ejection seat.
- o Normal aircraft operations will be flown below 10,000 feet; however, oxygen will be provided in the event that flight at higher altitudes is required.
- o The crew members will wear fire-resistant suits at all times.

The minimum time to clear the aircraft following initiation is critical. A 0.5 second delay should be met. The time between escape system initiation and recovery under a full canopy should also be minimized to under 3 seconds. These requirements will improve the low altitude recovery capability of the escape system.

1.8.2 Baseline Description

The crew escape system selected for the RTA provides through-the-canopy ejection using two Stencel Aero Engineering Company SIIIs-3F16 ejection seats with interseat ejection sequencing. The SIIIs-3 has also been selected for use on the US Navy AV-8A Harrier VTOL aircraft due to its fast acting, low altitude recovery capability. Using this system, the AV-8A has undergone an operational ejection under emergency conditions at an altitude of (80 ft) with the airplane at a 70^{0} roll angle, forward velocity of 110 knots, and a sink rate of 1500 ft./min. The crew was **safely** recovered from this incident. This same capability will be available on the RTA due to minimal design changes on the SIIIS-3F16. The major difference between the Harrier and that proposed for the RTA is the shape of the seat bucket. The escape performance of this ejection seat should not change since the same basic sequencing subsystem available on the Harrier seat is also available on the SIIIS-3F16. Due to this high degree of commonality only a minimum amount of testing will be required to demonstrate compatibility with the RTA.

Preliminary evaluation indicates that the SIIIS-3F16 will meet the performance criteria outlined by Boeing as well as those criteria specified in the US Navy ejection seat specification Mil-S-18471D. Of particular concern are the low altitude terrain clearance capabilities. Figures 1.8-1 and 1.8-2 illustrate the capability of the SIIIS to safely recover crew members under adverse dive and roll angles. The system will provide safe escape at altitudes from sea level to 10,000 feet and for velocities up to 300 knots. The seat structure is designed to withstand crash loads of up to 40g's, and accepting inflight load factors of 5.0g's. Based on test results of the Harrier system, the ejection seats will clear the canopy in 0.17 seconds following initiation of the escape procedure, and the pilot will be safely recovered under a full parachute in less than 2 seconds. This quick parachute deployment is provided due to the use of a ballistic parachute spreading device. The seat is designed to be used by Navy aviators between the 10th and 90th percentile.

To preclude interference between the ejectees after simultaneous initiation, the trajectories of the individual seats diverge. Divergence is produced by staggering the ignition of the dual seat back rockets to provide 14 ft/sec lateral divergence for each seat. The trajectory of the ejection seat and drogue chute under various flight conditions was simulated by Stencel Aero Engineering Company and are shown in Figure 1.8-3. This illustrates the ability of the seat to provide tail clearance at the maximum operating speed of 300 KEAS. This also indicates that the drogue chute will clear the nacelles, but further investigation of this will be required.

Figure 1.8-4 shows the ability of the SIIIS-3F16 to meet the specified flight conditions. Acceleration limits of 10-12g's maximum are met except for a very short 17g spike. A better measure of the injury potential during ejection seat operation is given by the dynamic response index (DRI) as described in Mil-S-9479B. The DRI is representative of the maximum dynamic compression of the vertebral column of the human body. The DRI considers the human body to be a lumped parameter consisting of mass, spring and damper. Figures 1.8-5 through 1.8-8 show acceleration data which has been reduced according to the DRI technique. The figures illustrate the low injury potential of the SIIIS-3 for limiting operating conditions and for a variety of sized air crewmen.

TERRAIN CLEARANCE AS A FUNCTION OF DIVE ANGLE

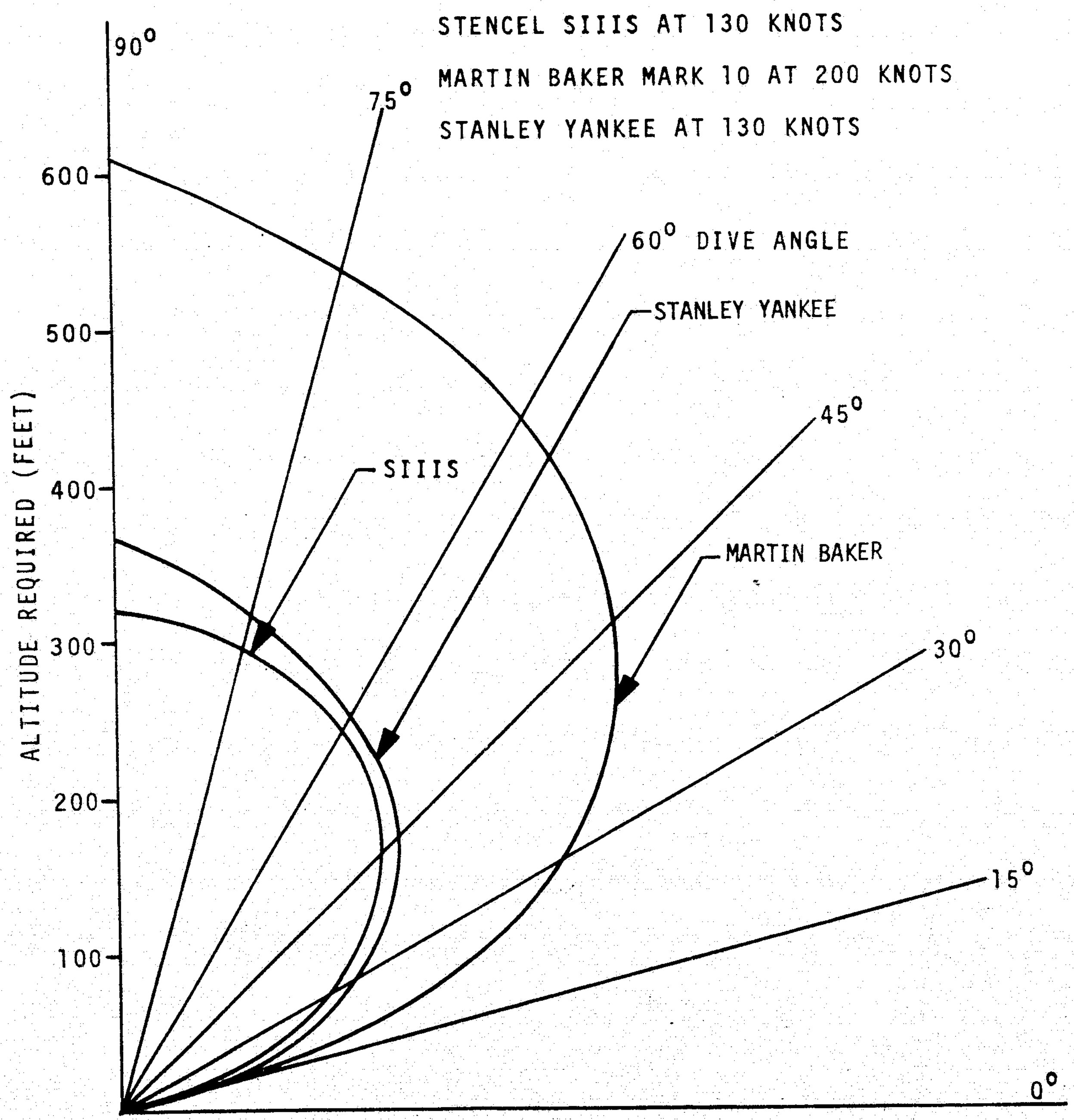
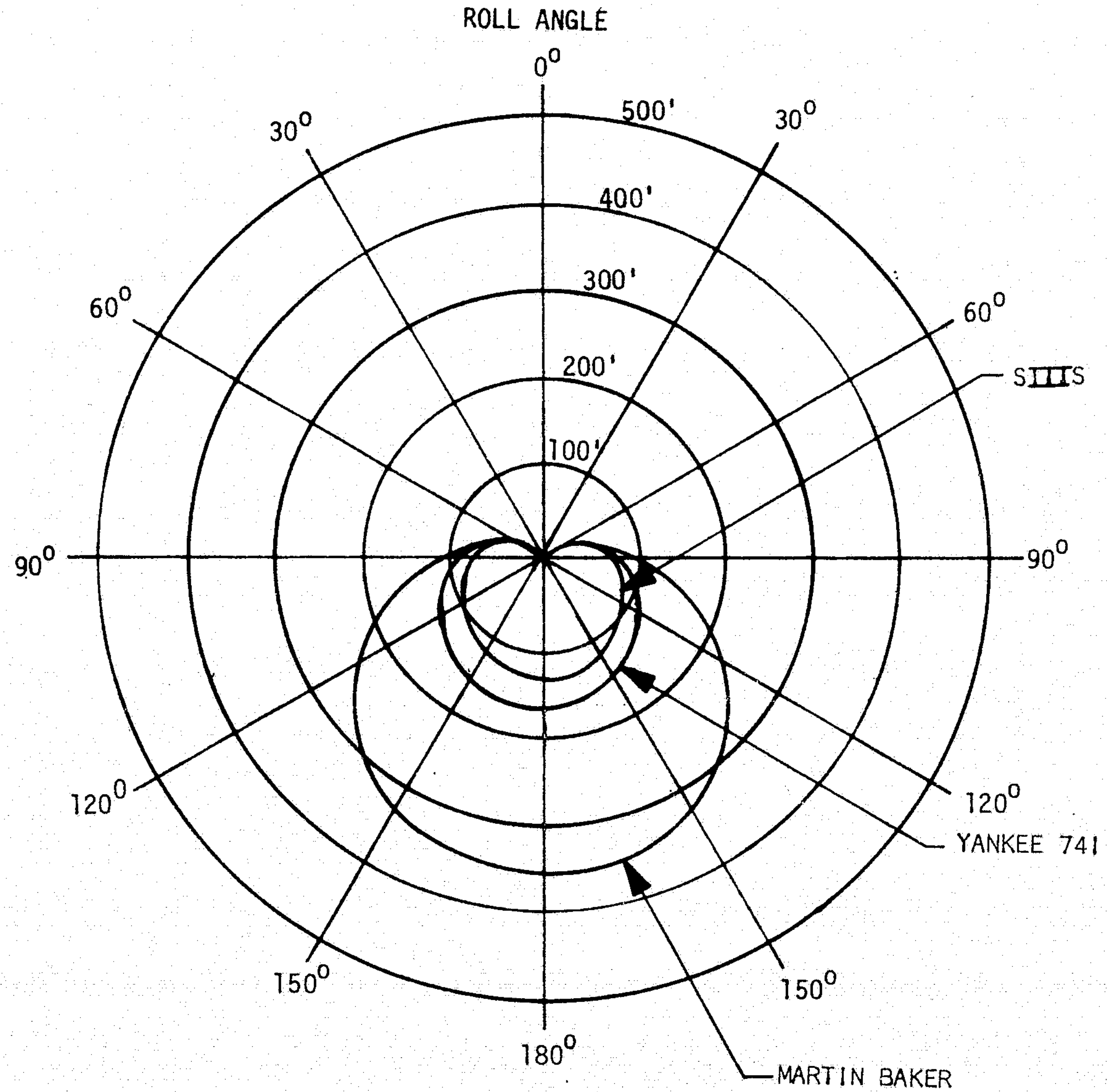


FIGURE 1.8-1

TERRAIN CLEARANCE AS A FUNCTION OF ROLL ANGLE



STENCEL SIIIS AT 130 KNOTS

MARTIN BAKER MARK 10 AT 130 KNOTS

STANLEY YANKEE AT 130 KNOTS

FIGURE 1.8-2

TAIL CLEARANCE

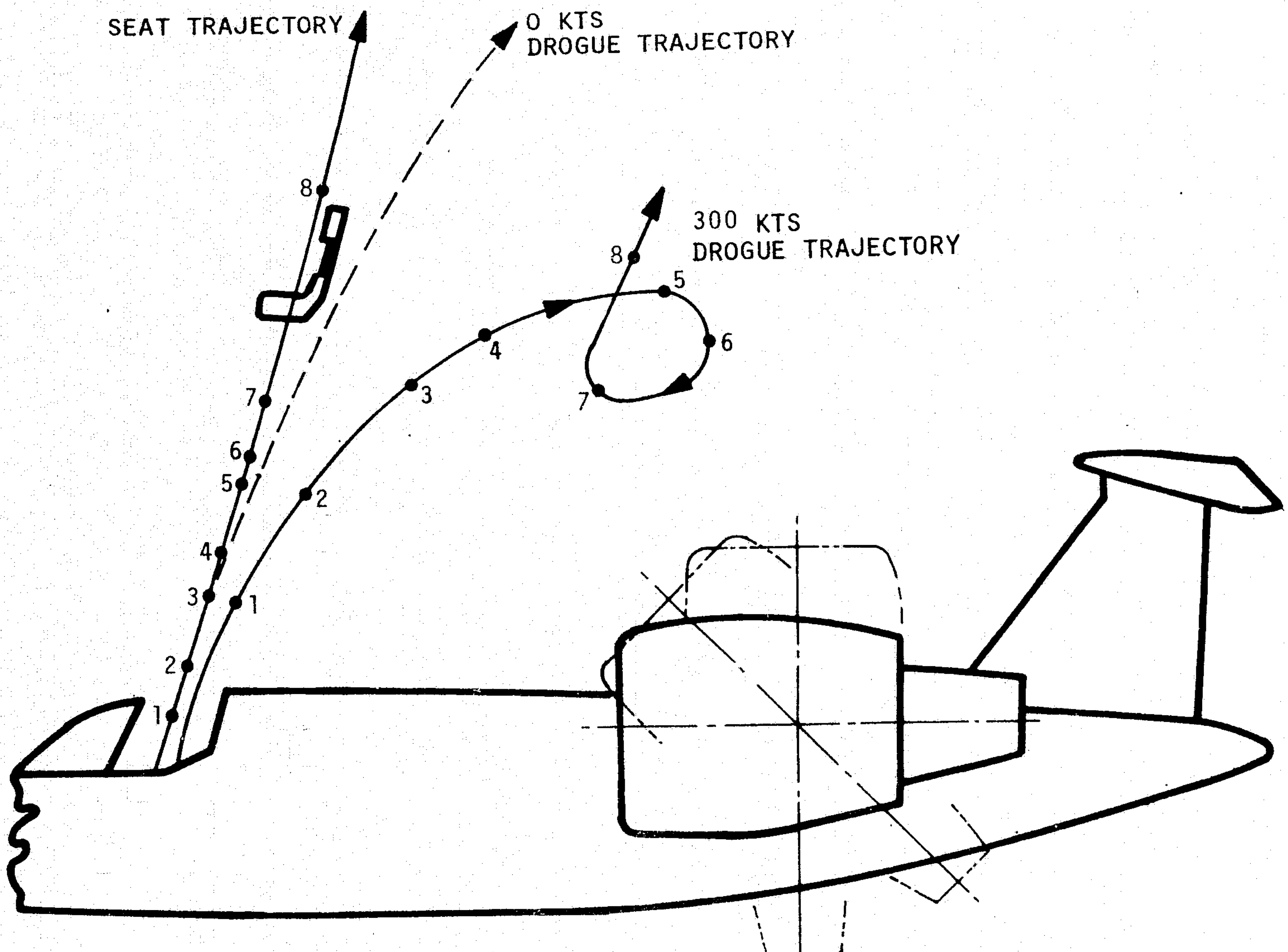


FIGURE 1.8-3

T-39 VSTOL RECOVERY HEIGHTS

TO FIRST FULL INFLATION

SINK RATE FPM	VELOCITY KEAS	PITCH ANGLE DEG	ROLL ANGLE DEG	ALTITUDE REQUIRED	
				SPECIFIED	SIIIS - 3F16 1/
0	10	0	0	0	0
2500	100	0	0	50	12
5000	100	0	0	100	32
2000	100	0	90	100	63
5000	100	0	90	200	118
0	130	0	180	200	137
5000	130	20	0	120	38

1/ ASSUMES WORST CASE PENCENTILE (10% OR 90%) AND ADVERSE CG.

FIGURE 1.8-4

DYNAMIC RESPONSE INDEX

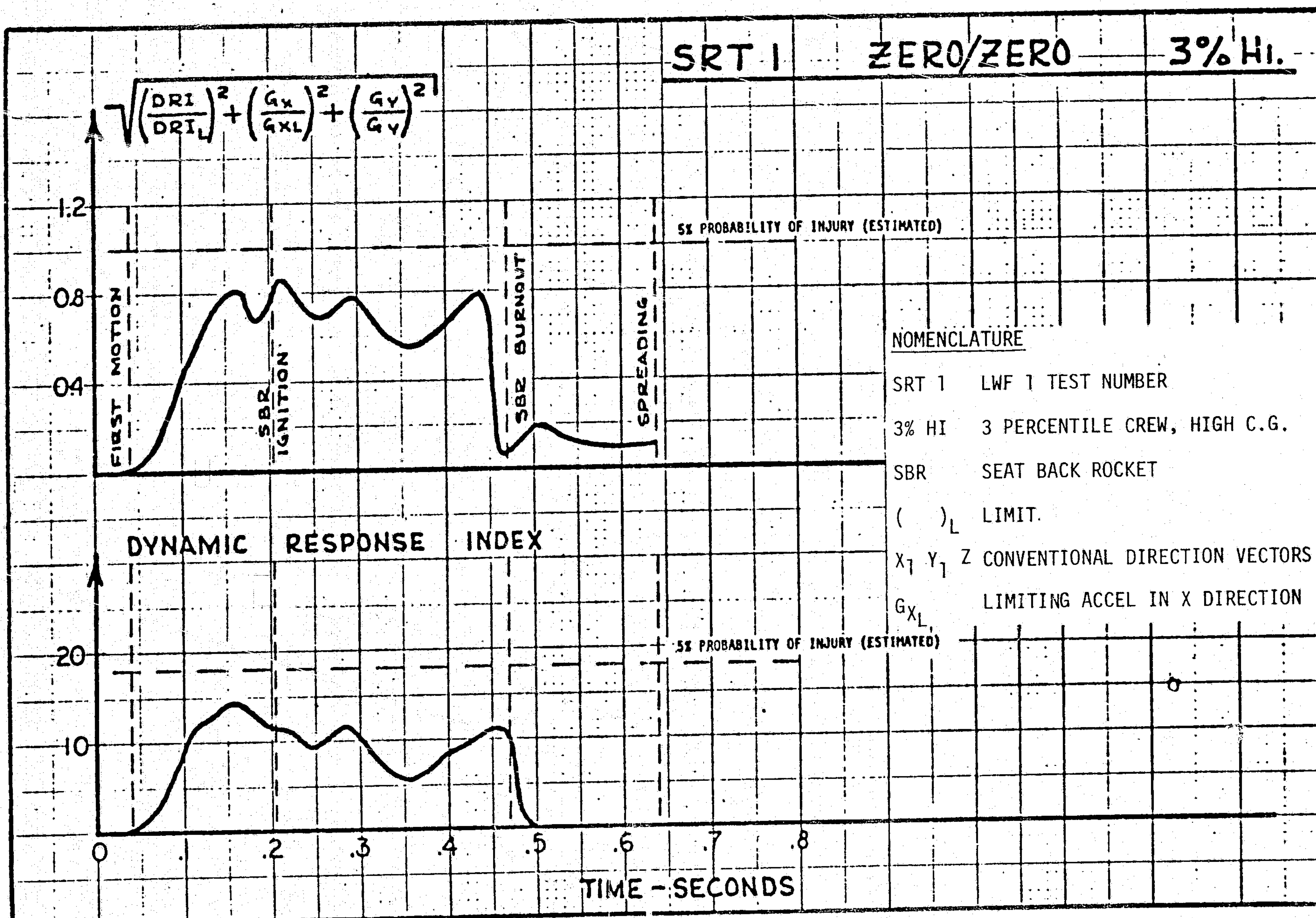


FIGURE 1.8-5

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DYNAMIC RESPONSE INDEX

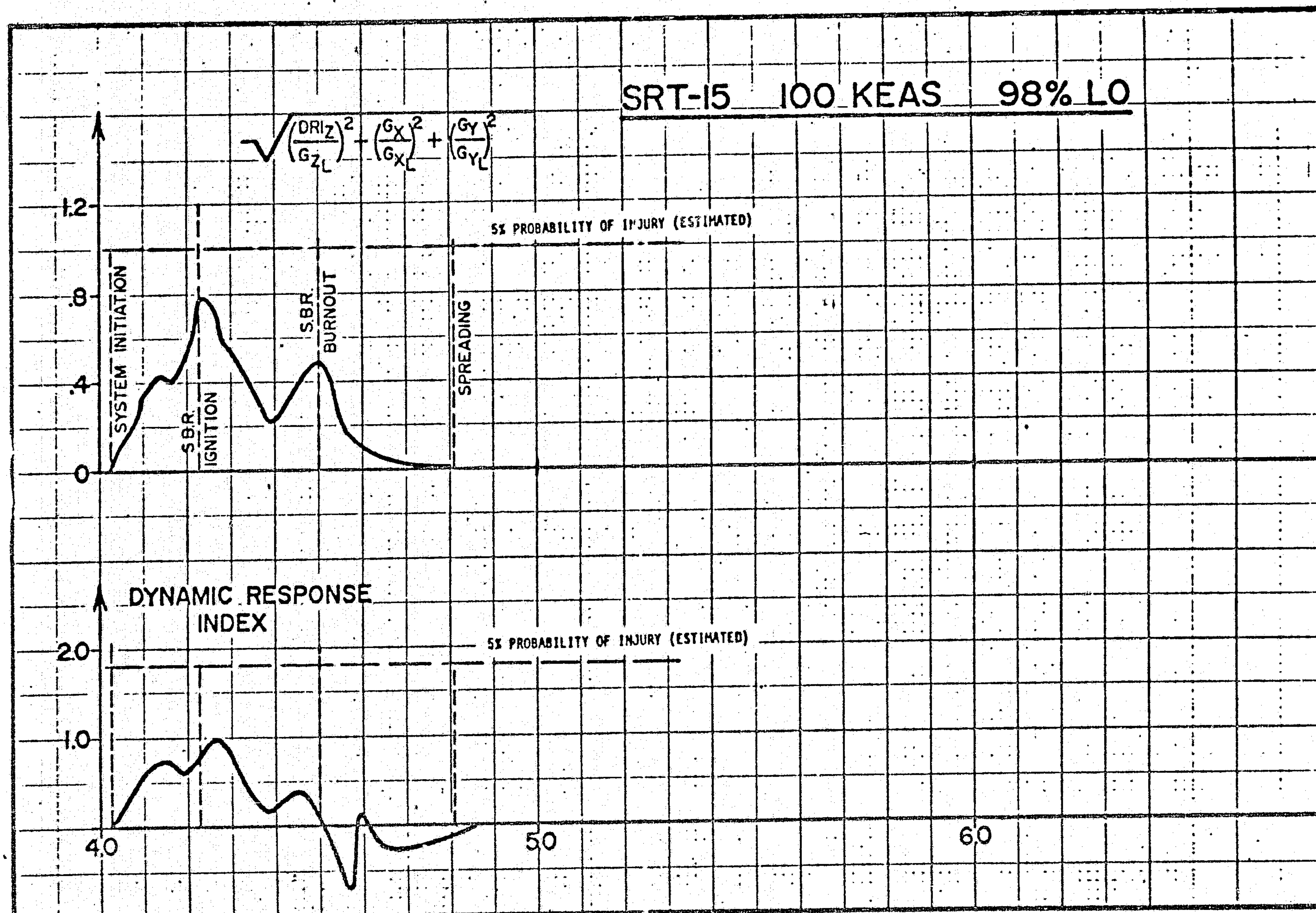

FIGURE 1.8-6

DYNAMIC RESPONSE INDEX

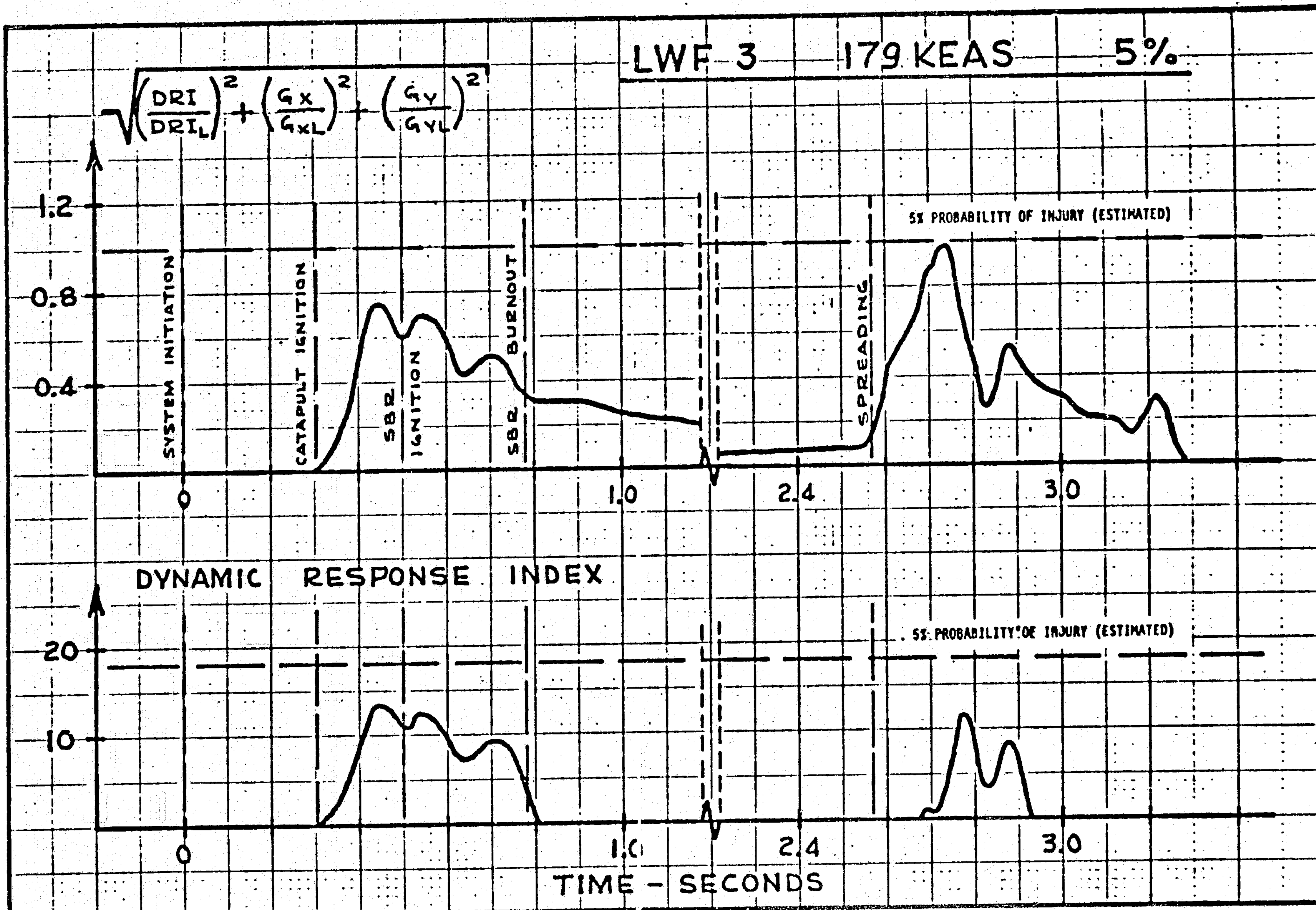

FIGURE 1.8-7

DYNAMIC RESPONSE INDEX

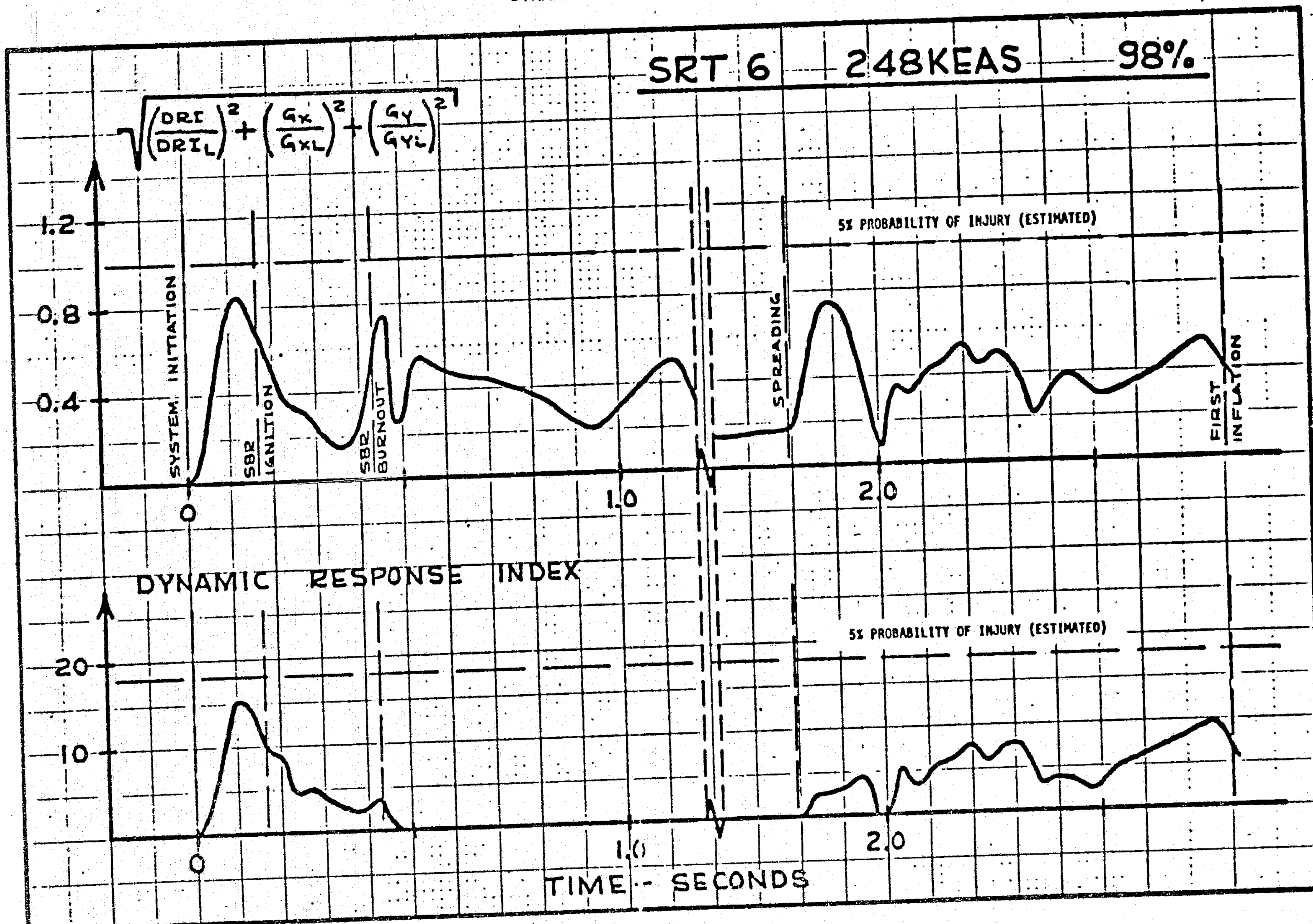

FIGURE 1.8-8
58

The seat will be installed in the RTA following airframe modifications to allow for a clear ejection path. The seat is installed with a 17^{o} seat back and rail angle. The seat is attached to the airframe through eight slippers. The location, size and axes of the slippers is illustrated in Figure 1.8-9. Also listed on the same figure are predicted dynamic loads and moments produced on the slippers during six representative emergency conditions. The seat will be adapted with canopy breakers as developed for use on the Alpha Jet. These canopy breakers have demonstrated excellent protection of the ejectee when used to passively break a biaxially stretched acrylic canopy 0.33 inches thick. The use of canopy breakers and "through-the-canopy" ejection relieves the necessity for providing a canopy removal actuator. This also permits the fastest acting escape system presently available to proceed without any programmed time delays. The base of the seat must be slightly elevated above the existing floor to maintain the design eye point corresponding to that currently available on the T-39. The weight of various major sub-assemblies are:

SUB-ASSEMBLY		WEIGHT (lbs.)
Catapult Components		19.04
Seat Components		14.32
Parachute Components		23.76
Bucket Components		24.40
Bucket-Mounted Components		50.43
Interface-Mounted Components		5.15
Miscellaneous Hardware		4.06
	Total	141.16 lbs.

1.8.3 Trade Studies - Crew Escape System

VTOL aircraft present significant crew escape problems due to low altitude operation, low speed operation, high sink rates and possible fast attitude changes following catastrophic failures.

Trade studies were conducted in two phases. Initially, a survey of vendor literature was conducted to ascertain which systems were functionally

SIIIS SEAT SLIPPERS

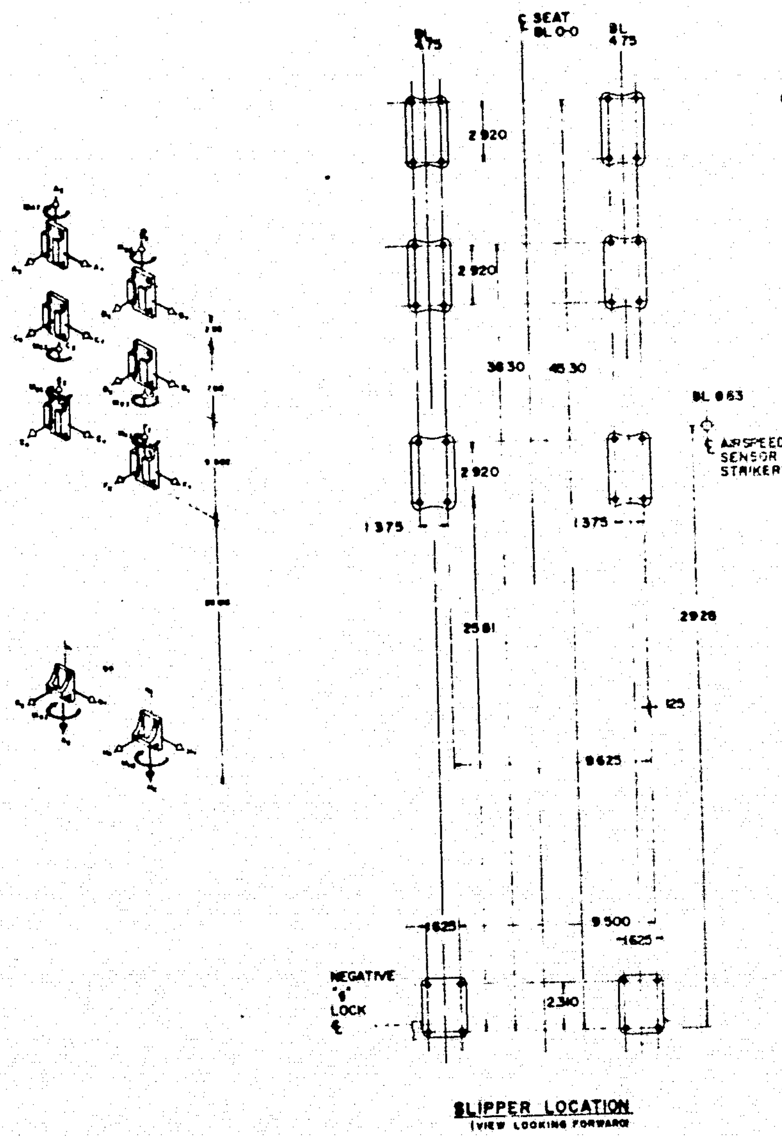

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FIGURE 1.8-9

capable of meeting the Boeing requirements. Following this preliminary screening, the vendors were selected to provide budgetary cost data, detail system description and performance data. The baseline system was selected based on this information.

1.8.3.1 Preliminary Systems Survey. Although detail description of all systems were not available, enough information on the selected systems was available with which to make objective decisions.

The systems which were surveyed are compared on the basis of weight, capability of meeting Boeing specified performance criteria, integration into existing aircraft, test requirements, performance record on similar aircraft, and production availability. The detail comparison of this information is listed in Table 1.8-2 along with subjective comments. The terrain clearance as a function of roll angle and dive angle are compared in Figures 1.8-1 and 1.8-2. The sources of information for the performance data is the same as that noted in Table 1.8-1 unless otherwise indicated. The results of this comparison indicated that two candidate systems provided the operational characteristics and performance required for integration within the RTA. The two systems are the Stanley Yankee 741 and the Stencel S111S-3. Detail analysis was conducted on both of these systems.

1.8.3.2 Detail Analysis. Requirements and configuration sketches were forwarded to both Stanley Aviation and Stencel Aero Engineering Company for system proposals and budgetary costing. Stencel Aviation proposed the SIIIS-3 figure as described in Section 1.8.2. Stanley Aviation proposed the Yankee 741.

The Yankee Escape System uses a tractor rocket to pull the man from the cockpit rather than the pushing rocket of an ejection seat. The escape system consists of the following parts:

1. Model 741 Parachute Assembly
2. Model 7210 Seat Back
3. Model 712 Rocket Catapult
4. Rail Assembly
5. Personnel Harness

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ESCAPE SYSTEM / PARAMETER	MARTIN BAKER MARK 10	STANLEY YANKEE 734	STANLEY YANKEE 741	NORTH AMERICAN LW3B	STENCEL SIIIS-3	MCDONNEL DOUGLAS ACES-2	WEBER T-37	SAAB 37 VIGGEN
WEIGHT Pounds	188		130	134	141	138	130	228
DIMENSIONS height	N.A.	D.N.A.	(56")	(50.5")	(48")	(51.75")	(45.5")	1.314 m
depth	N.A.	D.N.A.	(28")	(22.5")	(28")	(30.37")	(26")	.866 m
width	N.A.	D.N.A.	(20")	(16.5")	(19.5")	(20.0")	(19.5")	N.A.
Seat back angle	N.A.	D.N.A.	13^o	15^o	17^o	17^o	17^o	17^o
CREW SIZE RANGE	5% - 95%	5% - 95%	5% - 95%	5% - 95%	5% - 95%	5% - 95%	5% - 95%	5% - 95%
ACCELERATION maximum	14-16g's	8-10 g's	8-10 g's	12-15 g's	10-12 g	10-15 g	N. A.	20 - 25 g
rate of onset	180-210 g/sec	N.A.	N. A.	150-180 g/sec	N.A.	110g/sec	N. A.	1000 g/sec
OPERATION TIMES clear cockpit	N.A.	0.3 sec	0.3 sec	0.56 sec	0.17 sec	0.17 sec	N. A.	N. A.
full parachute	N.A.	2.0 - 2.5 sec	2.0 - 2.5 sec	2.0 - 4.0 sec	less than 2.0 sec	1.9 sec	N. A.	N. A.
CANOPY BREAKERS	yes	yes	yes	yes	yes	yes	no	N. A.
INITIATION MECHANISM	Central firing handle	N.A.	Dual side mounted controls	Central "D" ring	Central "D" ring	Dual side mounted controls	Dual side mounted controls	Dual side mounted controls
CREW MEMBER DIVERGENCE PROVISIONS	Asymetric thrust	launch angle divergence	launch angle divergence	Asymetric weight	Stagger seat back rocket firing	N. A.	N. A.	N. A.
AIRCRAFT QUALIFICATION STATUS	RAF Harrier	N.A.	N.A.	XV-5A XV-15	U.S. Navy AV-8A Harrier	N. A.	Cessna T-37	SAAB 37 Viggen
OPERATIONAL MODES	0-450 KEAS	0-230 only	0-230 KEAS 230-450 KEAS	0-200 KEAS 200-450 KEAS	0-225 KEAS 225-450 KEAS	0-250 KEAS 250-650 ?KEAS 650 - ++ ? KEAS	120-400 KEAS	40-600 KEAS
PRIMARY SOURCE OF ABOVE INFORMATION	"Martin Baker Mark 10 Ejection Seats" Martin Baker Co. LTD	Versatile Yankee Stanley Aviation	"Versatile Yankee" Stanley Aviation	"The LW3B Escape Sys" North American Inc.	"The SIIIS Family" Stencel A.E. Co.	Report MDC-J4193, "ACES 2" McDonnell Douglas	Compilation of Data on Crew Emergency Escape Systems Boeing	"SAAB 37 Viggen Rocket Seat" SAAB-Scania
COMMENTS	Too heavy Performance Unsatisfactory	lightweight Does not meet speed reqm'ts.	Lightweight	Unable to ascertain production status or availability	Fastest available system	Both the Yankee & SIIIS have better low altitude recovery	No load speed operation	Presently not available for import

N.A. not available

[illegible] not applicable

ESCAPE SYSTEM SURVEY

Table 1.8-2

The parachute assembly contains the recovery parachute, the seat pan, back frame, headrest, drogue parachute and restrain harness. This assembly is extracted with the man. The rails provide structural interface between the seat and the aircraft. The seat is the interface between the rails and the parachute. The ejection is initiated by dual side mounted controls.

The Yankee and SIIIS-3 systems were compared under various flight attitudes and velocities (see Figure 1.8-4). The SIIIS system surpasses the terrain clearance of the Yankee under all conditions except the inverted 130 KEAS condition in which the performance of the Yankee is only slightly better. This performance capability is not hindered by the more sturdy construction of the SIIIS-3 which provides lateral support from the seat side panels. The SIIIS-3 system has been proven to perform superbly within VTOL type aircraft (AV-8A Harrier). The Yankee system has been used on helicopters in a modified form called the "AWAY" system but has not been proven for through-the-canopy VTOL escape. The Yankee system provides a weight saving; however, based on initial evaluation of factors, such as safety, ruggedness and proven compatibility with VTOL aircraft, the SIIIS-3 escape system has been selected.

1.9 AVIONICS

Three different operating environment requirements will be encountered by the aircraft in flight. The aircraft will be flown in the Seattle area for checks of the propulsion system and controls and to verify flight characteristics. The aircraft will then be ferried from the Seattle area to the customer's facility. It is considered that this flight will be made VFR. The aircraft will then be operated for test and evaluation from the customer's facility. These test flights, as well as the Seattle-area flights, will be conducted VFR. These operations will be primarily along the low-altitude racetrack defined by the S.O.W.

The avionics complement for the aircraft has been selected with primary consideration for the safety of the aircraft and other users of the airspace, followed by considerations of equipment weight and program cost. The Federal Aviation Regulations were examined for guidance and applicability.

The baseline is a T-39A which will be supplied to Boeing with normal avionics. To avoid modification of equipment racks or wiring changes, the normal equipment has been evaluated for retention in each case. Equipment was removed when possible to avoid excessive total demonstrator aircraft weight. The avionics complement of the basic T-39A, the recommended test complement, and the recommended ferry complement, are shown in Table 1.9-1.

The Intercommunication System is retained since it provides the audio connection of the pilot and copilot to the communications systems, the Tacan identification audio, and the ground crew intercom station.

The quantity of VHF radio systems has been increased from 1 to 2. This has been done because of the importance of communications during test operations. VHF was assumed to be the primary communications mode at the demonstration site. Addition of this capability will require installation of an equipment mount, an antenna pair (upper and lower) and antenna switch and selector, and a control unit.

ITEM	T-39A BASIC	TEST	FERRY
Intercommunication System AIC-10	1	1*	1*
UHF Command Radio System AN/ARC-34A	1	2	2
VHF Command Radio System Collins VHF 101	1	0	1
Tacan System AN/ARN21C	1	0**	1**
Glide Slope Receiving System Collins 51V3	1	0	0
VOR/Localizer Receiving System Collins 51X-2	1	0	1
Collins 344B	1	0	1
Marker Beacon Receiving System Collins 51Z-2	1	0	0
IFF-SIF Systems AN/APX-46V	1	0	0
Automatic Direction Finder System AN/ARN-59	1	0	0
AIMS System AN/APX-72	1	0	1
Weather Radar System AN/APS-113	1	0	0
Gyro Compass	1	1	1

* Cabin Loudspeaker and cabin intercom control are removed during modification

** Cabin Tacan indicator is removed with navigators position

TABLE 1.9-1 AVIONICS FOR DEMONSTRATOR AIRCRAFT

The VHF radio is retained for the ferry operation to facilitate emergency landing at a civil airport. This is achieved with little impact, since the racking, antenna, control and wiring are part of the basic T-39A. During test operations, the transmitter and receiver will be removed from the equipment rack to avoid unnecessary operating time and weight.

The Tacan system is used in the aircraft during the ferry operation to ease the cockpit load by permitting flight on Tacan radials rather than compass navigation. The receiver-transmitter of this system is removed from the equipment rack during test operations.

No IFR flights are currently planned during test or ferry operations, and, therefore, the glide-scope receiver and marker beacon receiver are removed from the aircraft. The VOR/Localizer system is retained to provide backup for Tacan during the ferry flight. The racked receiver and instrumentation unit are removed during test operations.

The IFF-SIF system is removed completely from the demonstrator aircraft since no need can be shown for this capability during the test and ferry flights. The Automatic Direction Finder system is removed from the aircraft because the Tacan and VOR equipments provide better navigation aids during the ferry flight.

The AIMS system is retained for the ferry flight. Retention of this capability permits operation in controlled airspace above 12,500 feet without an ATC-authorized deviation from the requirements of Section 91-24 of Part 91 of the Federal Aviation Regulations, General Operating and Flight Rules. Since the installation and equipment is provided in the basic T-39A, there is no program impact from its retention.

The Weather Radar system is removed from the aircraft. The plan for VFR operations only eliminates the need for this capability, and removes the requirement for providing the installation ahead of the forward lift fan.

The gyrocompass system of the T-39A is retained to provide the compass capability. Similarly, the T-39A backup magnetic compass is retained to backup the gyrocompass system.

Support of the avionics is recommended by stocking spare LRU's at the test site with cycling of replaced units through normal Government channels. This would include both AN equipments and commercial equipment (Collins, Wilcox, or Eclipse-Pioneer). This approach is recommended to reduce interruptions of the test program. However, glide-slope and marker beacon antennas and antenna cables will be retained should a requirement for IFR develop.

1.10 FLIGHT CONTROL SYSTEM

1.10.1 Introduction

A functional diagram of the flight control system (FCS) is shown in Figure 1.10-1. Principal features of the FCS are a primary mechanical control system for CTOL flight, and a primary electrical control system for V/STOL flight. Automatic flight control signals, for stability and control augmentation and for automatic flight path control, are transmitted electrically to the appropriate power actuators.

Before giving a detailed description of the FCS, the design criteria will be summarized, so that a better understanding of certain control system features will result.

1.10.2 Design Criteria

The following criteria were used when developing and modifying the T-39 control system to perform the lift-fan technology demonstration task.

A) The flight control system must be at least single-fail-operational; and no single failure will be permitted to jeopardize flight safety.

B) Less than full operational capability will be acceptable after first failure of a power actuator.

C) Power actuator jams need not be considered, since redundant actuators can overcome most potential jams.

D) Redundant channels will have independent power supplies, hydraulic supplies, and failure detection.

E) Control surfaces must be irreversible, to prevent inadvertent back-driving of the V/STOL controls during transition and hover.

F) The primary and the automatic control systems shall be separated to the maximum extent practicable.

G) Stability and control augmentation commands shall be implemented in series and with limited authority.

H) Automatic flight path control commands shall be implemented into the flight control system in parallel and with limited authority.

I) Automatic flight control system authority limits shall be selected so that a hardover command can be overridden by normal pilot action using the primary controls.

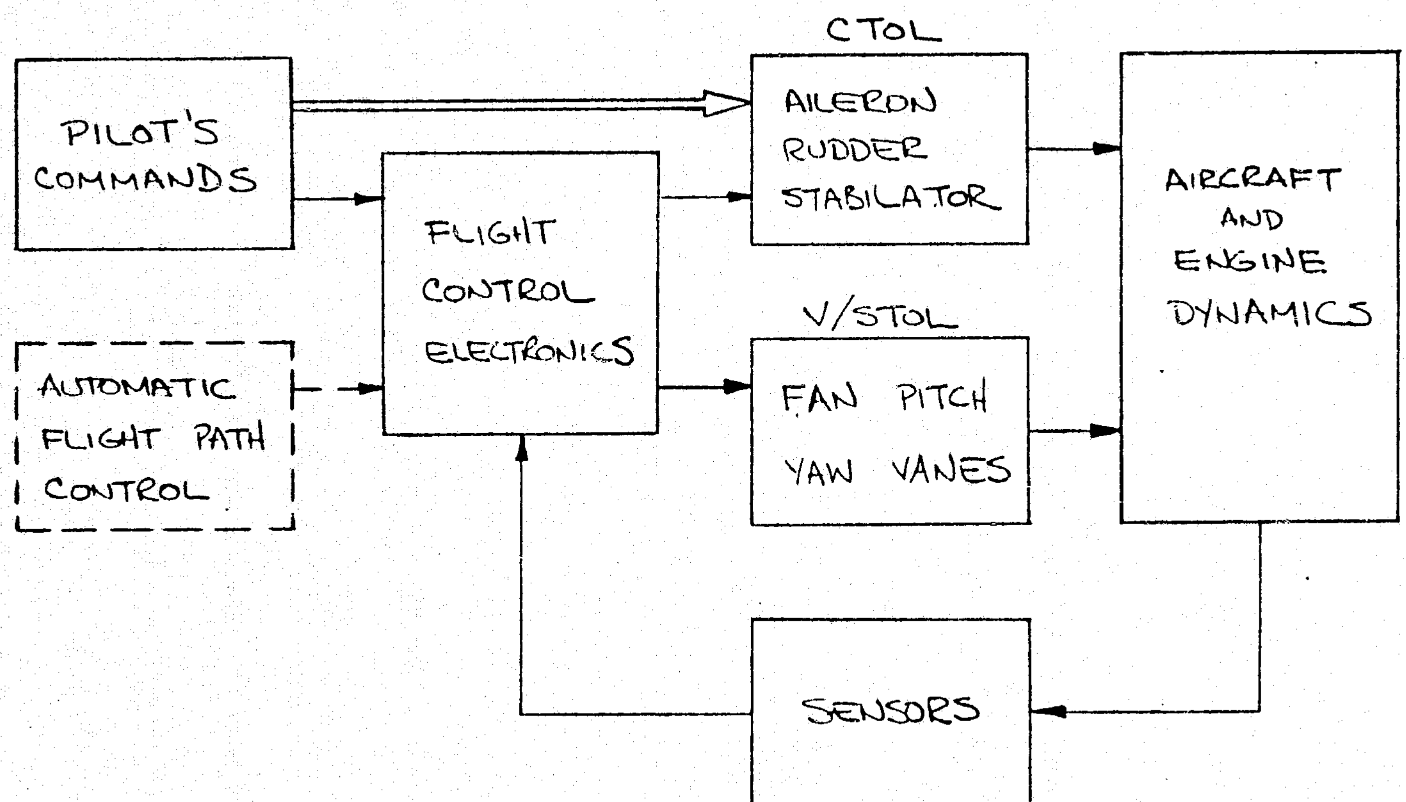

FIGURE 1.10-1 FLIGHT CONTROL SYSTEM SCHEMATIC DIAGRAM

J) Existing surfaces, cables, rods, and control parts will be retained wherevere possible.

K) Flight proven components, actuators and electronic systems shall be adapted for all possible applications.

L) Electronic packaging and construction shall consider system modifications and equipment growth.

M) Conventional cockpit controls shall be used.

In addition to the criteria listed above, fly-by-wire signaling was chosen for the V/STOL controls. Two major considerations led to fly-by-wire controls. First, the fan blade and yaw vane actuators in the engine pods were outboard of the nacelle pivot. Mechanical signaling of these actuators, while not impossible, seemed a more difficult design problem than electrical signaling. Second, it was anticipated that relatively complicated gain-charging would be required for the V/STOL controls during transition. Again, it is conceivable to gain-change a mechanical control linkage, but an electrical signaling system would be simpler to design and fabricate.

Finally, preliminary analysis of the dynamic stability of the demonstrator airplane shows that NASA guideline Level 2 frequency and damping requirements are not met with augmentation off. This implies that fail-operational stability augmentation is required.

1.10.3 Flight Control System Description

1.10.3.1 Roll Control. During conventional flight, control will be provided by the ailerons. During V/STOL transition and hover flight control will be provided mainly by differential thrust through fan blade angle. Ailerons will remain active and some aerodynamic control will be available. Yaw vane deflection will be employed to counter cross-coupling between roll and yaw (see Section 1.10.3.3 for yaw control description).

Manual flight control of the aileron surfaces is accomplished through a mechanical control system to transmit pilot commands, in series with electrical augmentation commands to an aileron hydraulic power servo actuator. Differential lift fan control is accomplished through a fly-by-wire system to provide roll control via differential fan blade angle commands. AFCS commands for flight path control are transmitted electrically, and in parallel, to the dual-input aileron servoactuator.

Lateral trim commands are introduced through the feel, centering, and trim unit.

Modification of T-39 lateral controls will involve the following:

- o Installation of a dual power, dual input (mechanical and electrical) servoactuator in the fuselage connected to the existing cable quadrants. Mechanical controls to ailerons in the wings will remain unchanged.
- o Installation of series secondary electric command servoactuator in the fuselage (dual for fail-op).
- o Installation of a feel, centering, and trim unit in pilot's mechanical control system.
- o Installation of aileron and control stick sensors.
- o Installation of fan blade actuation controls (see 1.10.3.4, hover controls).

A schematic diagram of the roll control system is shown in Figure 1.10-2.

1.10.3.2 Pitch Control. Pitch control during conventional flight will be provided by an all-flying tail stabilator. The T-39 elevator and horizontal stabilizers will be reworked to a T-tail configuration. During V/STOL transition and hover flight, pitch control will be provided by differential thrust between forward and cruise fans. The stabilator will remain active during transition and some aerodynamic control will be available. Manual flight control of the stabilator is accomplished through a mechanical control system in series with electrical augmentation commands to the hydraulic power servoactuator. Differential lift fan control is accomplished through a fly-by-wire system to provide pitch control via differential fan blade angle commands.

AFCS commands are transmitted electrically, and in parallel, to the dual-input aft servoactuator. Pitch trim is introduced in series due to the large trim excursions during V/STOL transition.

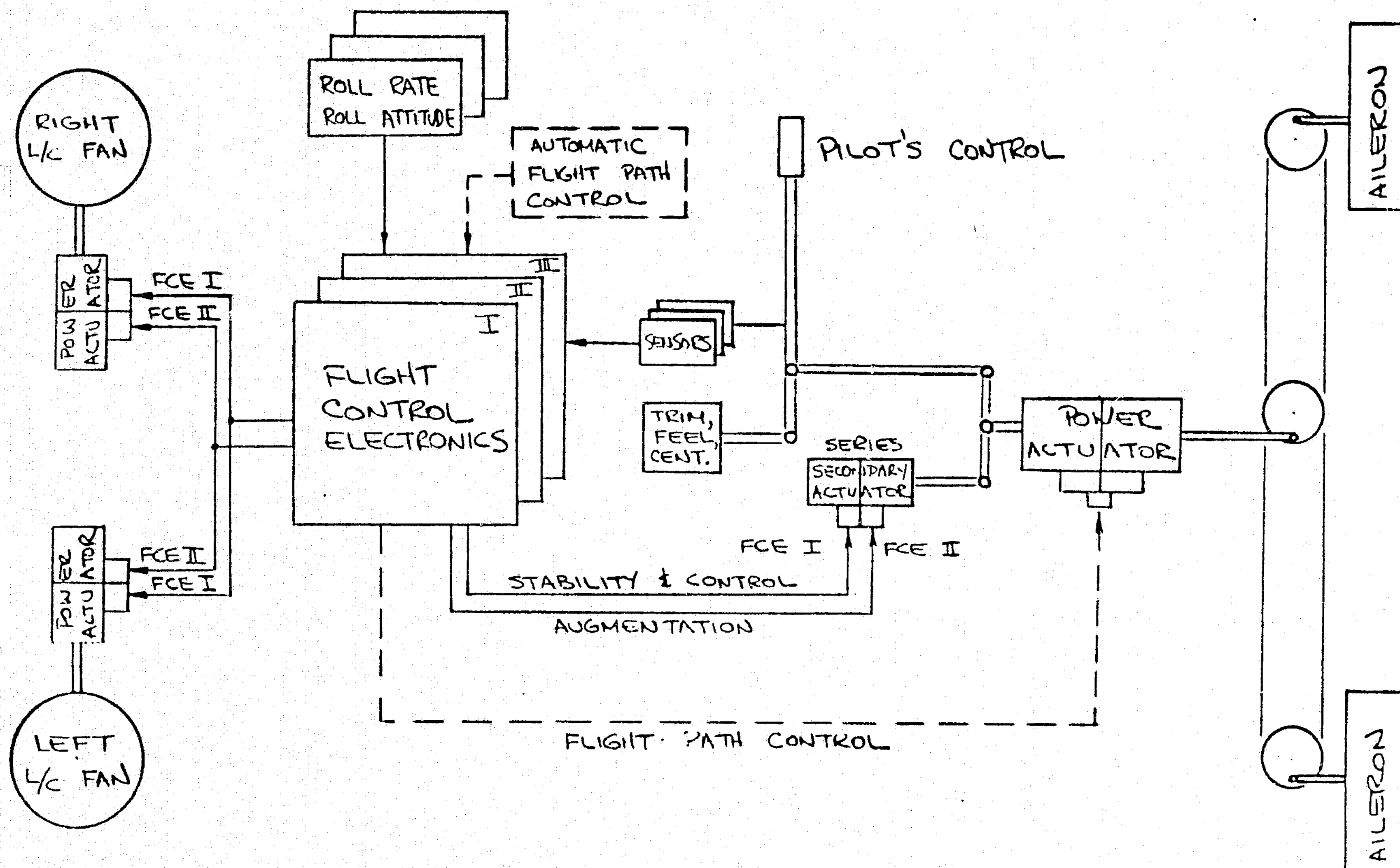

FIGURE 1.10-2 ROLL CONTROL SCHEMATIC DIAGRAM

Pitch control modification to the T-39 will involve the following:

- o Installation of a dual-power, dual-input (mechanical and electrical) servoactuator onto the horizontal stabilizer.
- o Installation of series secondary electric command servoactuator (dual for fail-op).
- o Installation of feel, centering and trim unit in pilot's mechanical control system.
- o Installation of surface and control stick sensors.
- o Installation of fan blade actuation controls.

A schematic diagram of the pitch control system is shown in Figure 1.10-3.

1.10.3.3 Yaw Control. Yaw control during conventional flight will be provided by the rudder. During V/STOL transition and hover flight, yaw control will be provided by yaw vane deflection. During transition, the rudder will remain active, and some aerodynamic control will be available.

Manual flight control of the rudder is accomplished through a mechanical control system in series with electrical augmentation commands to the rudder hydraulic power servoactuator. Yaw vane deflection commands are transmitted through a fly-by-wire system to provide yaw control.

Rudder trim is introduced in through the feel, centering, and trim unit.

No AFCS commands are transmitted to the rudder control system.

Yaw control modification to the T-39 will involve the following:

- o Installation of a dual-power, dual-input (mechanical and electrical) servoactuator onto the rudder surface.
- o Installation of feel, centering, and trim unit in pilot's mechanical control system.
- o Installation of surface and pedal sensors.
- o Installation of yaw vane actuation controls.

A schematic diagram of the yaw control system is shown in Figure 1.10-4.

1.10.3.4 Height Control. In addition to the pitch, roll, and yaw control discussed in the previous paragraphs, height control will be provided during VTOL flight and V/STOL transition by controlling fan blade

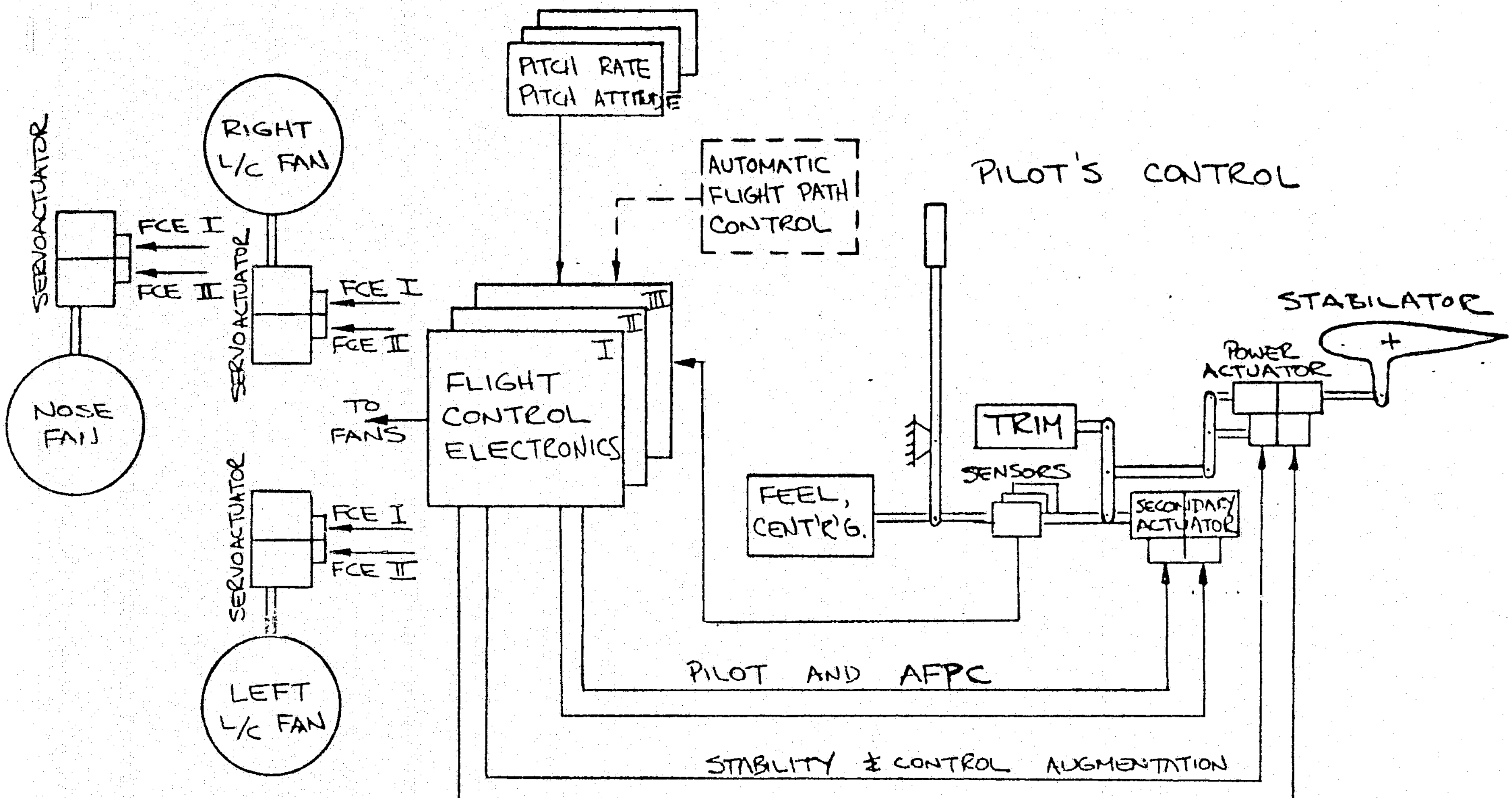

FIGURE 1.10-3 PITCH CONTROL SCHEMATIC DIAGRAM

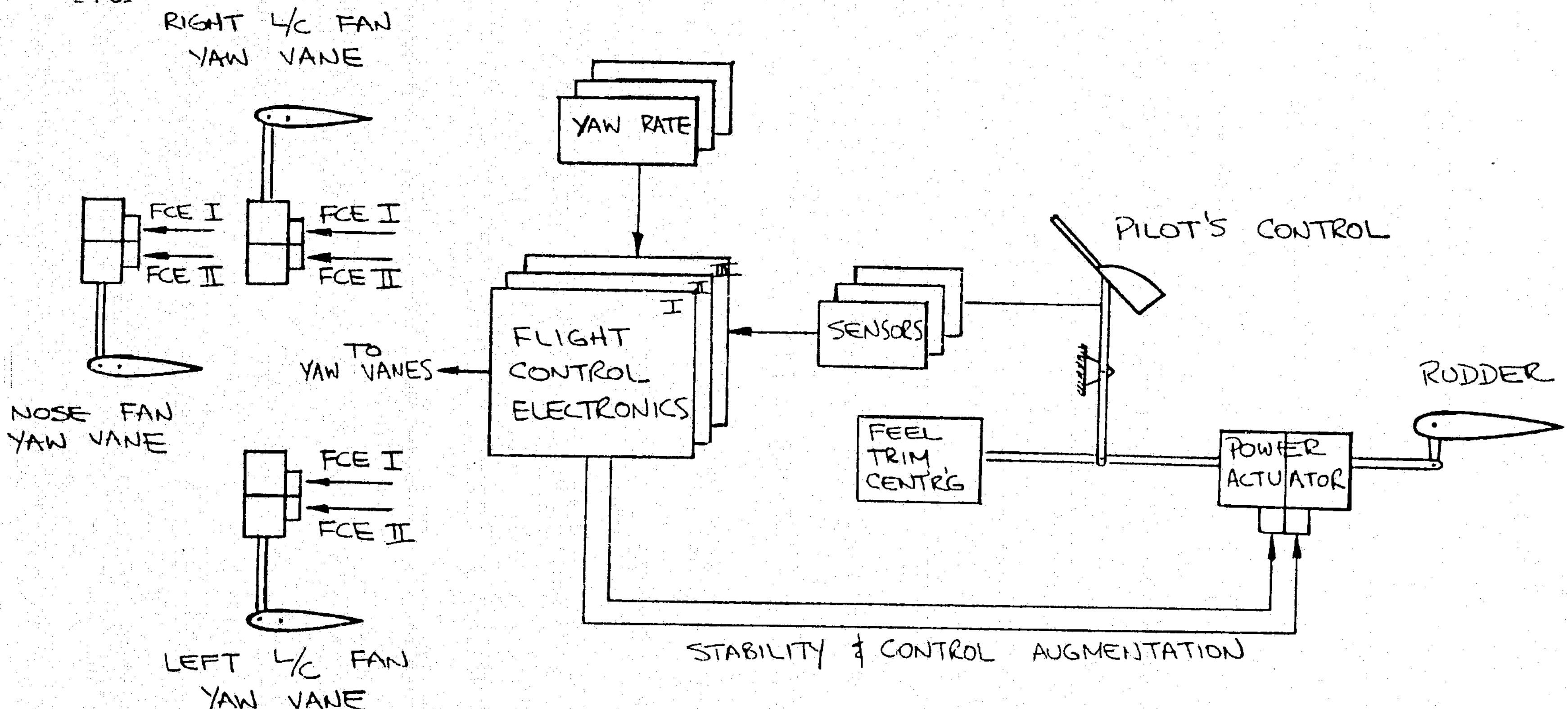

FIGURE 1.10-4 YAW CONTROL SCHEMATIC DIAGRAM

angle of all three fans simultaneously.

Manual and automatic height control is accomplished through a fly-by-wire system. Pilot commands are introduced through a new height control (collective pitch) lever.

Hover controls addition to the T-39 will involve the following:

- o Installation of height control lever (collective pitch) and sensors in the cockpit.

A schematic diagram of the height control is shown in Figure 1.10-5.

1.10.3.5 Nacelle Tilt Angle Control. During transition from conventional to V/STOL flight or vice versa, it will be necessary for the pilot to command nacelle tilt angle changes. For the present, it is assumed that a thumb switch on the height control lever will be used for this purpose. Since details of this control are not known, no sketch is presented.

1.10.3.6 Avionics.

A) Computing and Interfacing

The digital computer will be similar to that used on the HLH prototype or NASA 515 (terminally-configured 737) program. The computer will have the following characteristics:

- o Triplex redundancy.
- o Failure protection to prevent AFCS failures from adversely affecting flight safety.
- o Sensor inputs will be consolidated through signal select processing.
- o Incremental serial digital computers with cross-channel synchronization.
- o Failure monitoring with automatic correction.
- o Off-line built-in test equipment (BITE).

B) Flight Deck Controls and Displays

The flight deck will be the basic T-39 with some new instrumentation and with a collective pitch (height control) lever.

New instrumentation and displays will consist of a low speed airspeed indicator, angle of attack and sideslip, radar altimeter and altitude rate. In addition, new sensors required but probably not displayed

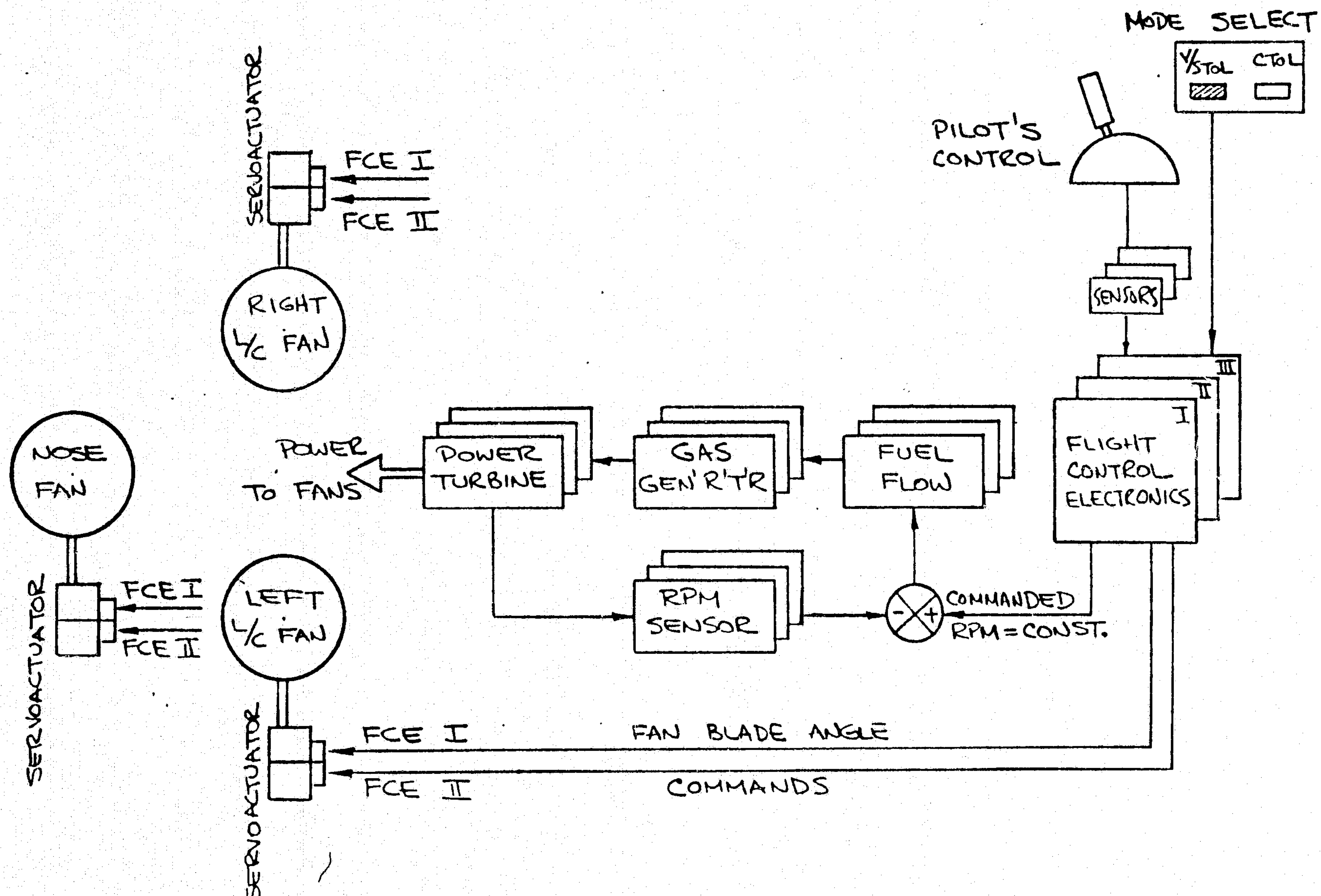

FIGURE 1.10-5 HEIGHT CONTROL SCHEMATIC DIAGRAM

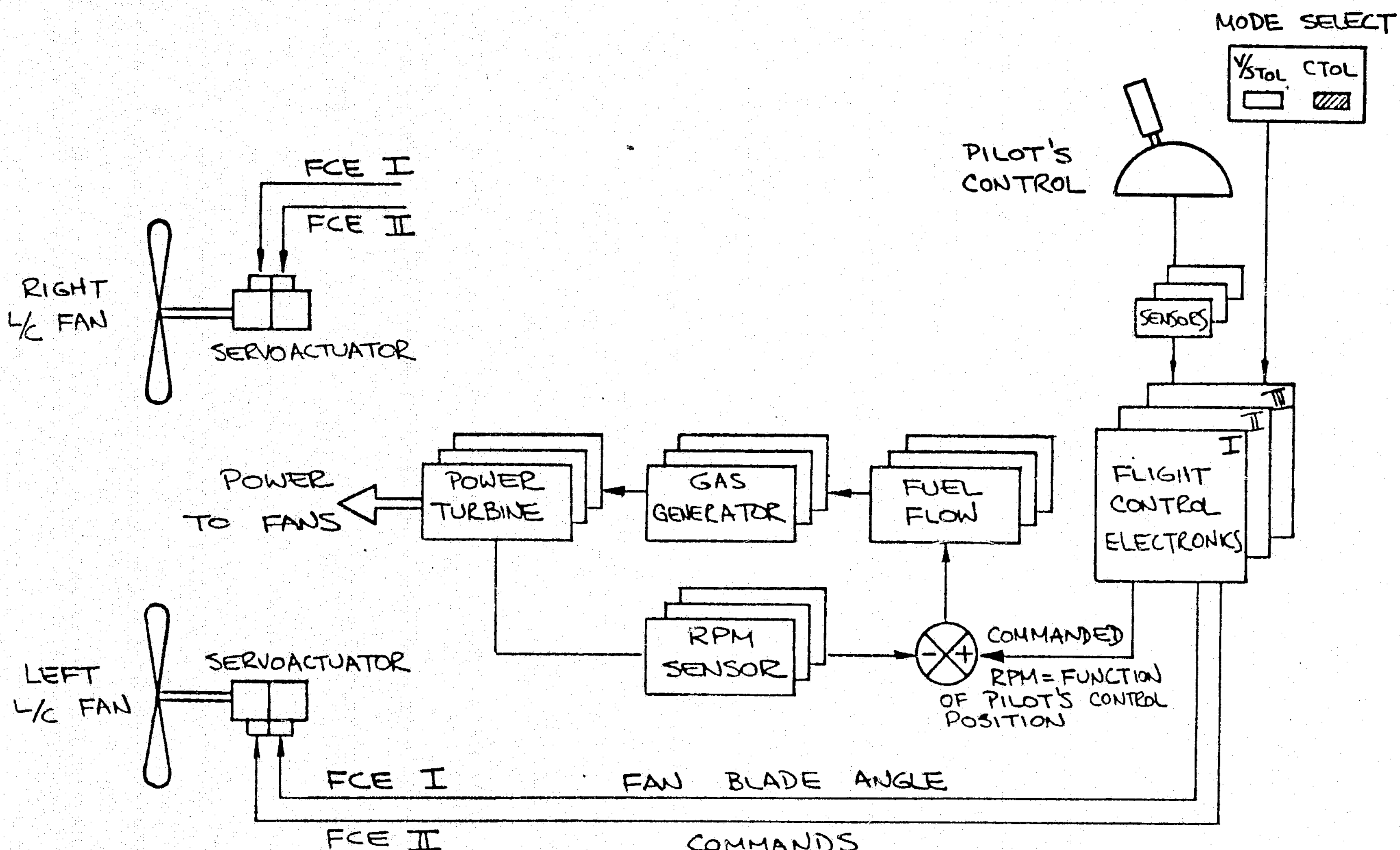

FIG. 1.10-5A UP-AND-AWAY THRUST CONTROL SCHEMATIC DIAGRAM

are vertical acceleration, additional vertical gyro, and a triplex three-axis rate sensor.

A schematic diagram of the flight control avionics is shown in Figure 1.10-6.

1.10.4 Design Control Power

Each fan has a blade angle stop that restricts the maximum thrust obtainable. At design rpm, on a standard day, the maximum thrust per fan is 11,100 pounds. When all engines are operating, all three fans can deliver 11,100 pounds simultaneously. When a pod engine has failed, the maximum total thrust available from three fans is 26,281 pounds. Any single fan can still deliver 11,100 pounds, but the total is reduced due to a power limit from two engines. A pod engine failure is slightly more critical than failure of the engine in the body. This is because the pod engines operate in the fan slipstream, which creates a supercharging effect.

At a design weight of 26,300 pounds, the airplane moments of inertia are:

Roll, $I_{XX} = 19{,}565$ slug-ft^2

Pitch, $I_{YY} = 91{,}101$ slug-ft^2

Yaw, $I_{ZZ} = 100{,}834$ slug-ft^2

With the c.g. at the nominal location (30% $\bar{c}$), the moment arms of the various fans are shown in the figure that follows.

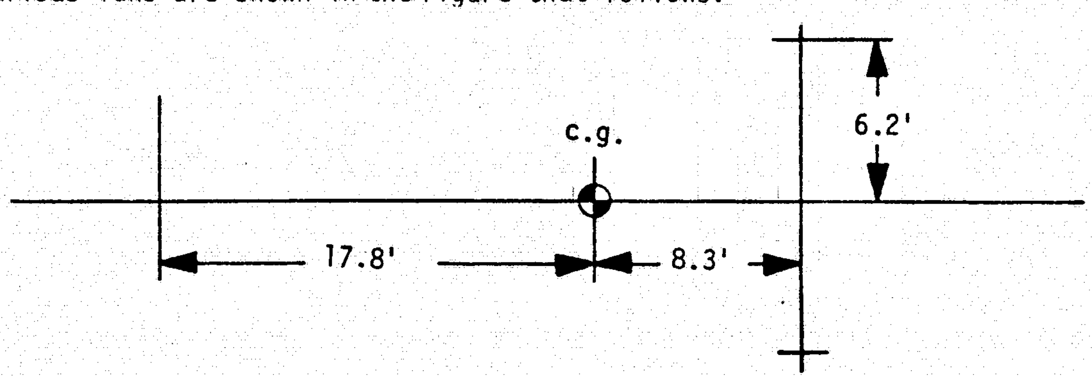

FIGURE 1.10-7

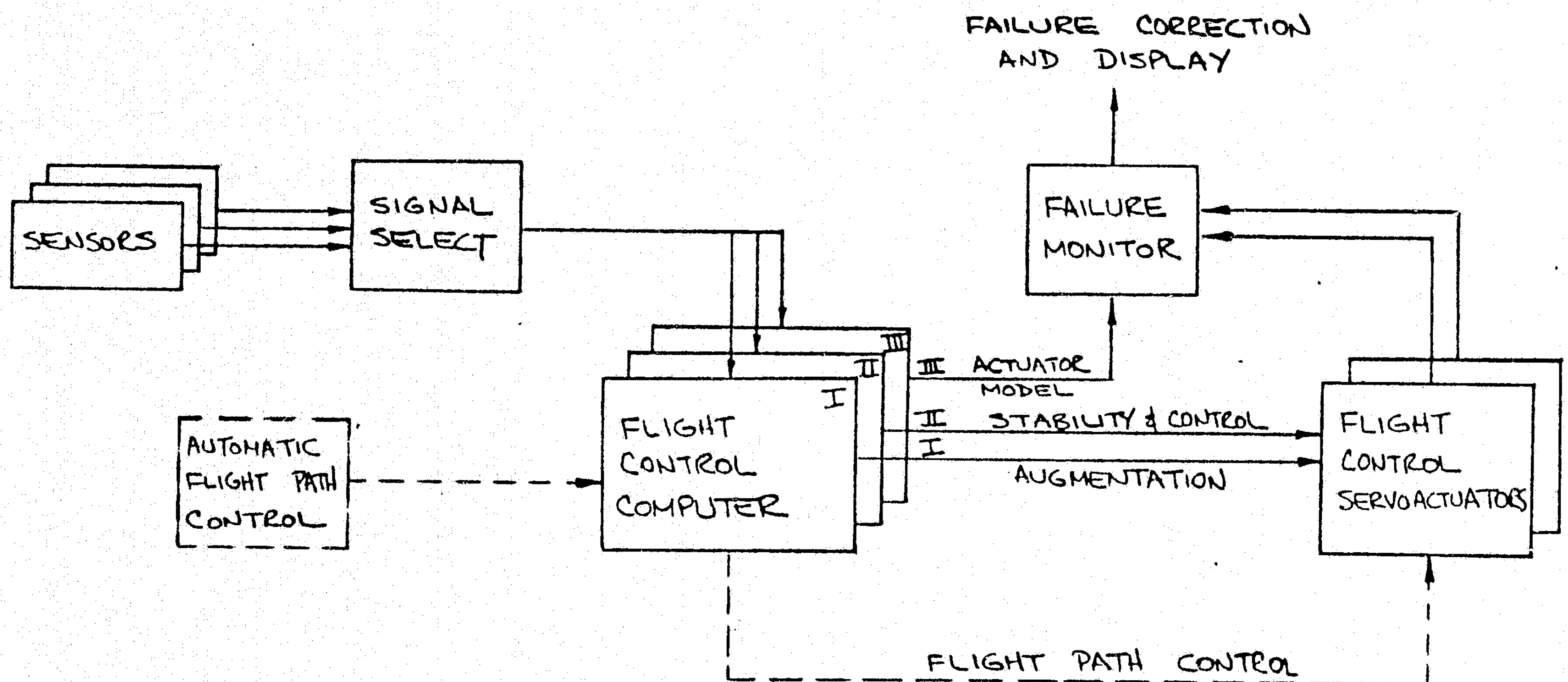

FIGURE 1.10-6 FLIGHT CONTROL SYSTEM REDUNDANCY LEVELS

1.10.4.1 All Engines Operating

Nominal thrust on the three fans is:

L/C = 8975 pounds
Nose = 8350 pounds

In pitch, for airplane nose up, the nose fan will go to 11,100 pounds and each cruise fan to 7,600 pounds. Pitch acceleration equals .77 rad/ sec^2.

In roll, one cruise fan will go to 11,100 pounds and the other to 6,850 pounds. Roll acceleration equals 1.34 rad/sec^2.

In yaw, the guideline value of .3 rad/sec^2 is met. No change in drive system gearbox loading is imposed for yaw control.

For height control, the cruise fans will go to 11,100 pounds each and the nose fan to 10,330 pounds. A thrust to weight ratio of 1.19 can be developed at a design weight of 27,400 lbs.

1.10.4.2 Most Critical Gas Generator Inoperative

As above, nominal thrust on the fans is:

L/C = 8975 pounds
Nose = 8350 pounds

In pitch, the nose fan will again go to 11,100 pounds, each cruise fan to 7,600 pounds, so the pitch acceleration is .77 rad/sec^2.

In roll, engine-out is the same as all engines operating, total thrust is 26,300 and $\ddot{\phi} = 1.34\ rad/sec^2$.

In yaw, since yawing moment is produced with practically no change in vertical thrust, the engine-out yaw acceleration will be essentially the same as the all-engine value.

For height control, a maximum of 26,300 pounds total thrust is available which produces a thrust-to-weight ratio of 1.03.

1.11 PROPULSION

The propulsion system for the Technology Demonstrator Airplane consists of three variable pitch fans driven by turboshaft engines through an interconnected mechanical drive system. It is required to demonstrate and test V/STOL thrust capability at low cost within the time frame of the program. This requires engine components which are either in production or in advanced stages of design and development. A survey was conducted of existing turboshaft engines either in production or in test. The results of this search are shown in Table 1.11-1.

For the modified T-39 demonstrator concept, the Detroit Diesel Allison (DDA) XT701 engine was a logical candidate because of its power output level and availability. Alternate propulsion systems considered for the T-39 are given as follows: (1) Model 1041-135-2R using two XT701 engines, (2) Model 1041-135-2A using three XT701 engines as shown in Figure 1-2-1, and (3) Model 1041-13-2B using two XT701 engines and one DDA T-56-14A engine. Other candidate engines, engine combinations and airframes are considered in studies shown in Task III. The overall studies indicate that Model 1041-135-2A with three XT701 engines should be the baseline configuration because of the thrust performance margin and installation compatibility.

The baseline propulsion system consists of two lift-cruise fan/engines mounted in vectorable nacelles and a fuselage-mounted engine and lift fan all interconnected through a mechanical drive system as shown in Figure 1.5-1. The lift fan provides vertical thrust only for the V/STOL modes of operation and is disengaged for conventional flight. However, the installation of the fuselage-mounted center engine is such that it will be capable of continuous operation through both low speed and high speed flight. In this way, the center engine avoids needing to be restarted in flight and all engines may be operated from a common power control resulting in an overall simpler and more reliable system. The inlet is designed to provide good recovery at low airplane speeds when high thrust levels are needed and sufficiently low distortion flow at high airplane speeds to allow stable engine operation. The exhaust nozzle is designed to minimize both the back pressure effect on the center engine and engine flow interactions with the fuselage.

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TABLE 1.11-1
TURBOSHAFT ENGINE CANDIDATES

			SLS PERFORMANCE, $90^{O}F$			
ENGINE MANUFACTURER	TYPE	USE	TAKEOFF POWER (SHP)	CONTINGENCY POWER/TIME (SHP/(MIN)	BARE ENGINE WEIGHT (LBS)	DEVELOPMENT STATUS
AVCO-LYCOMING	T55-L-11C	CHINOOK HELICOPTER	3500	4400/30 (a)	735	IN PRODUCTION
GENERAL ELECTRIC	T64-GE-415	SIKORSKY HELICOPTER	3850	4380/2.5 (a)	730	IN PRODUCTION
DETROIT DIESEL ALLISON	T56-A-14	LOCKHEED C130	4270	-- / --	1335	IN PRODUCTION
	XT701-AD-700	HEAVY LIFT HELICOPTER	7540	8780/2.5 (b)	1179	PASSED PPFRT AND SDT
ROLLS-ROYCE	RB410 D-02	DEMONSTRATOR ENGINE WITH DOWTY-ROTOL FAN	8000 TO 10,000	--	--	DEVELOPMENT TEST M45H DERIVATIVE
PRATT & WHITNEY	JFTD-12	SIKORSKY HELICOPTER	4250	--	920	IN PRODUCTION

(a) DEMONSTRATED IN TESTING

(b) DESIGN REQUIREMENT

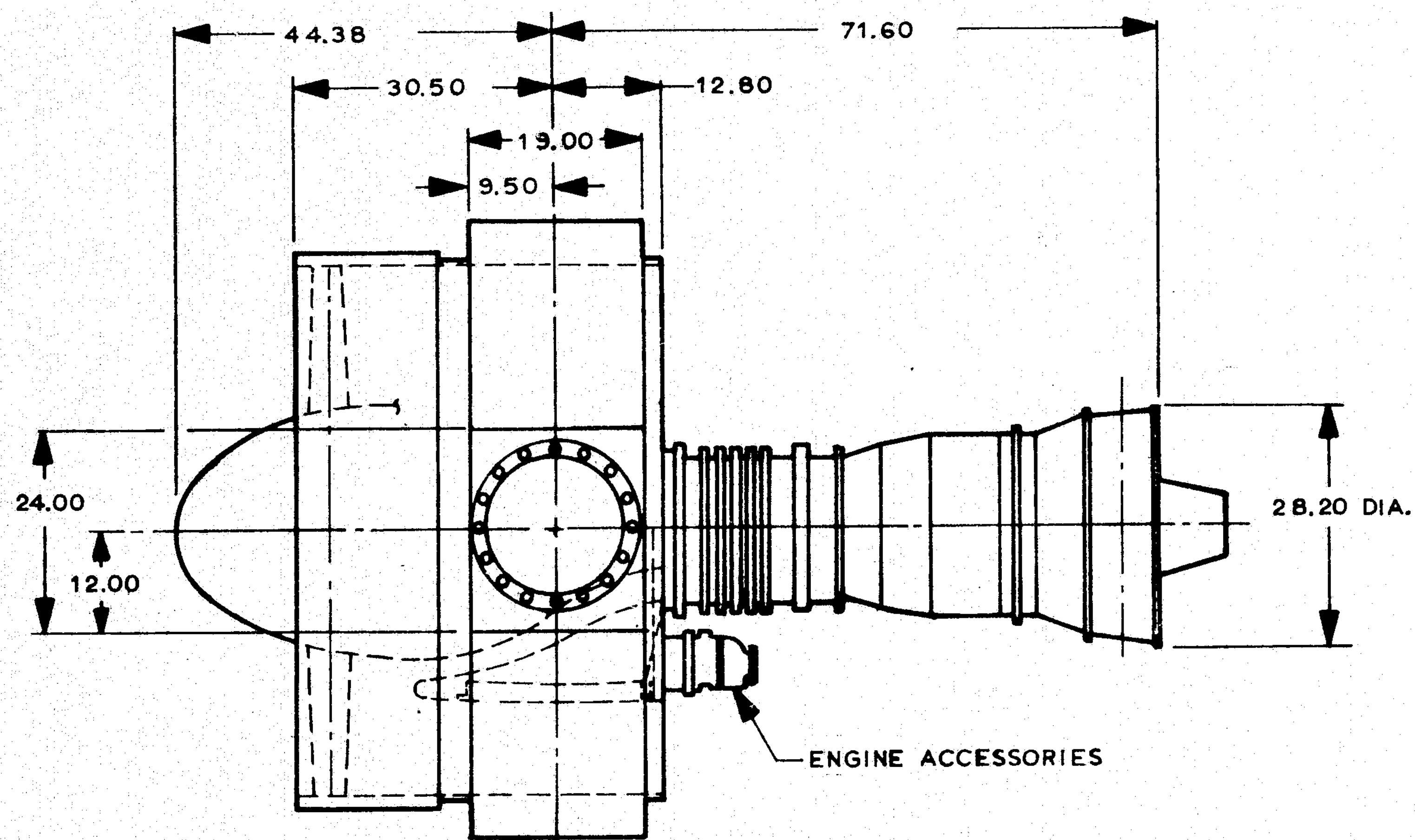
Each of the three fans on the airplane are 1.575 m (62 inch) fan face diameter variable pitch fans based on Hamilton-Standard designs, and experience with their 1.397 m (55 inch) diameter demonstrator Q-fan. The variable pitch blades provide fast thrust change for control response requirements. The fan manufacturer will also provide the front lift fan bevel gear set which provides the proper speed ratio between the drive system and the front fan.

The integrated lift/cruise fan, engine, and gearbox installation, Detroit Diesel Allison Model PD370-25E, envelope is shown in Figure 1.11-1. The XT701 engine is a DDA turboshaft engine developed for the Heavy Lift Helicopter Program and has successfully completed the Prototype Preliminary Flight Rating Test (PPFRT). Each engine has an overrunning clutch which will automatically disengage the engine from the fan system if the engine fails in which case the two remaining engines will provide the power. The engine manufacturer will also provide the gear reduction system between the engine and fans for the lift-cruise system.

The inlet for the rotating lift-cruise nacelles must satisfy the requirements of good recovery and low drag at cruise and low fan-face distortion levels when operating at the high in-flow angles during transitional modes of operation. Based on results of 1/4 scale model inlet wind tunnel tests, a fixed geometry, asymetric inlet design was selected. This inlet achieves a good compromise between the conflicting requirements while minimizing complexity and weight. A large scale inlet is currently being tested in the NASA Ames 40 x 80 wind tunnel with the Hamilton Standard 1.397 m (55 inch) diameter Q-fan and a T-55 Lycoming engine.

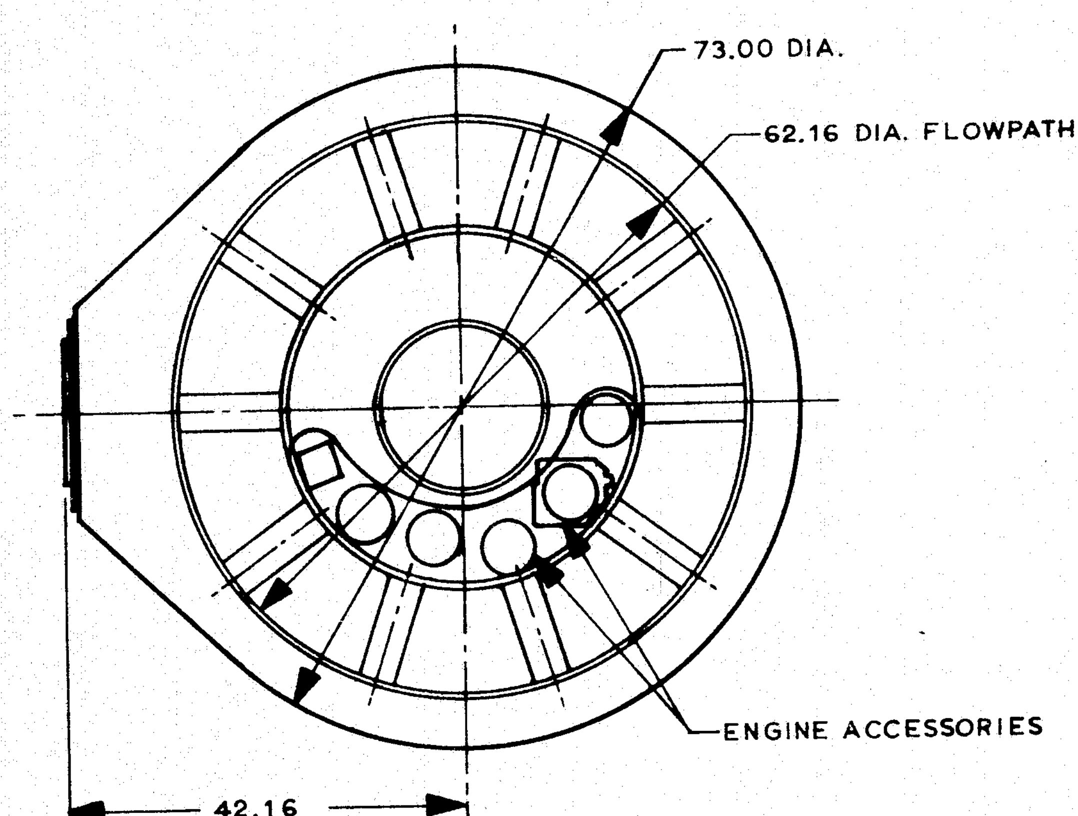
The fan nozzle is a two-position nozzle to provide area match at high power and at loiter. The engine core nozzle is a fixed area nozzle. Yaw vanes are mounted in the fan exhaust stream to provide yaw control during V/STOL modes of operation.

The propulsion system thrust requirement is sized by the VTOL mode with one engine inoperative for a 90^{0}F day at sea level altitude. For this condition, reference (RFP) requires that a capability for a thrust to weight ratio of 1.03 be achieved in the VTOL mode without altitude control at an airplane weight that shall include 1136 kg (2500 lbs) payload and fuel for two VTOL mission circuits.



INSTALLATION ENVELOPE
ALLISON T-701-PD370-25E

FIGURE 1.11-1



REAR VIEW INSTALLATION ENVELOPE
ALLISON T-701-PD370-25E

FIGURE 1.11-1 (continued)

TABLE 1.11-2

GROSS THRUST, ONE ENGINE INOPERATIVE, SLS, $90^{o}F$

CONFIGURATION	ENGINE RATINGS INTERMEDIATE (T.O.)	INTERMEDIATE PLUS WATER	CONTINGENCY PLUS WATER
1041-135-2A	26280	27870	30740
1041-135-2R	17530	18500	20220
1041-135-2B	22550	24120	25750

Studies showed that for this condition, a good match between thrust, power and weight for Model 1041-135-2A resulted in a fan diameter of (62 inches). This diameter is an upper limit with minimal or no risk. A smaller diameter could be selected commensurate with increased risk; the advantages of minimum diameter stems from both reduced nacelle size and increased fan pressure ratio. The smaller size eases installation and reduces aerodynamic/propulsion interference and the increased pressure ratio leads to higher cruise speed capability.

The installed thrusts using the selected fan are shown in Table 1.11-2 for the one engine inoperative condition at 32°C (90°F) and sea level static. Comparisons are shown for the various model configuration with and without water augmentation.

1.12 MASS PROPERTIES

Weight and balance statements, dimensional and structural data, and a summary of the weight additions and deletions required to modify the basic T-39A are included in this section. Appendix A contains a detailed listing of proposed T-39A airplane revisions and their associated weights and centers of gravity.

The basic T-39A, as reported in Reference 1, was first modified to include changes required to produce weights and balance for a configuration with (2) XT701 engines and (3) 62" diameter fans (Model 1041-135-2R), an updated version of the demonstrator airplane originally reported in Reference 2. A third XT701 engine and associated changes were added to the -2R airplane and mass properties data were also produced for the Model 1041-135-2A. An additional configuration (Model 1041-135-2B), which replaces the body-contained XT701 on the -2A with a T-56A-14 engine, is also presented. Although data are shown for the basic T-39A and three modifications thereof; the Model 1041-135-2A is considered the most promising configuration at this time so additional mass properties data are included in this report for the -2A.

Weights and/or weight increments were calculated using Boeing stress sizing information, vendor's quotations, actual T-39A parts, preliminary design detail analysis results (drive system components), historical weight growth data and parametric/statistical weight prediction methods.

A summary description of the tables showing the mass properties information follows:

Table 1.12-1

Summary group weight statements for the basic T-39A, Models 1041-135-2R, -2A and -2B showing group weight increments resulting from implementing the proposed changes. **TABLE 1.12-1A Presents inertial data for Model 1041-135-2A.**

Table 1.12-2 through 1.12-5

Weight and horizontal balance statements for the basic and modified configurations. Vertical centers of gravity for the basic T-39A and the Model 1041-135-2A are included on Tables 1.12-3 and 1.12-4 respectively.

Table 1.12-6

"Dimensional and Structural Data", page 5 of MIL-STD-1374, Part I for the Model 1041-135-2A.

Figure 1.12-1

C.G. Travel for Model 1041-135-2A.

Table 1.12-7

Pitch, roll and yaw inertias, products of inertia and principal axis slopes are shown for various weight conditions of Model 1041-135-2A.

Listed below is a summary of the design weights for the T-39 RTA, Model 1041-135-2A.

Maximum Design Weight	30190	Design STOL Landing Weight	27650
Flight Design Weight	26410	Design CTOL Landing Weight	26536
Design Vertical Landing Weight	25500	Maximum CTOL Landing Weight	29840
Maximum Vertical Landing Weight	29840		

The contingency weight center of gravity calculated at body station 223.6 (2% MAC) by removing major unmodified items from the basic -2A operating weight and adding the flight test equipment estimated weight and center of gravity as follows:

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Item	Weight (LBS/APL)	Horizontal Body Station C.G. (in.)
Basic -2A O.W. less contingency Weight	19550	278.5
Plus Flt. Test Equipment	+2500	59.6
Minus Wing	-1780	263.7
Minus Engines	-3410	371.3
Minus Unmodified Main Gear	-620	284.0
Minus Unmodified Nose Gear	-170	80.0
Minus Unmodified Empennage	-290	487.0
	(15780)	(223.6)
		2% M.A.C.

The computation or selection of contingency weight center of gravity is controversial. The c.g. location could result further aft with a different philosophy from that shown above. An extreme location of approximately three feet aft of 2% MAC or at 40% MAC was exercised for its effects on airplane balance. Figure 1.12-1 shows the center of gravity travel for Model 1041-135-2A with the chosen contingency weight C.G. (2% MAC) and the extreme aft location (40% MAC). The forward and aft aerodynamic c.g. limits are not exceeded in either case.

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ORIGINAL PAGE IS OF POOR QUALITY	BASIC* T-39A WTS. (LBS)	MODEL 1041-135 ΔWT. ⟷ (LBS)	-2R WTS. (LBS)	ΔWT. ⟷ (LBS)	-2A WTS. (LBS)	ΔWT. ⟷ (LBS)	-2B WTS. (LBS)
WING	1661	+189	1850	-70	1780		1780
HORIZONTAL TAIL	220	+130	350		350		350
VERTICAL TAIL	91	+99	190		190		190
BODY	1780	+490	2270	+50	2320		2320
MAIN GEAR	467	+213	680		680		680
NOSE GEAR	147	+63	210		210		210
LAUNCH AND RECOVERY GEAR	—		—		—		—
ENG. SECTION OR NACELLE	372	+1898	2270	+80	2350		2350
WING SKID	15	-15	—		—		—
STRUCTURE	(4753)	(+3067)	(7820)	(+60)	(7880)	(0)	(7880)
ENGINE	980	+1290	2270	+1140	3410	+200	3610
THRUST REVERSER	—		—		—		—
ENGINE ACCESSORIES	17	-17	—	—	—		—
ENGINE CONTROLS	30		30	+15	45		45
STARTING SYSTEM	22	-2	20	+15	35		35
FUEL SYSTEM	195	-15	180	+50	230		230
FANS	—	+960	960		960		960
DRIVE SYSTEM	—	+2070	2070	+550	2620		2620
EXHAUST & DEFLECTORS	28	+952	980	+80	1060	-10	1050
H_2O INJECTION SYSTEM	—	+150	150	-150	—		—
PROPULSION	(1272)	(+5388)	(6660)	(+1700)	(8360)	(+190)	(8550)
FLIGHT CONTROLS	326	+204	530		530		530
AUXILIARY POWER PLANT	—		—		—		—
INSTRUMENTS	166	+74	240	+30	270		270
HYDRAULIC & PNEUMATIC	145	+75	220		220		220
ELECTRICAL	924	-464	460		460		460
AVIONICS	464	-204	260		260		260
ARMAMENT	—		—		—		—
FURNISHINGS & EQUIPMENT	877	-397	480	+30	510		510
AIR CONDITIONING	279	+51	330	+30	360		360
ANTI-ICING	54	-4	50	+10	60		60
LOAD & HANDLING	3	+7	10		10		10
FIXED EQUIPMENT	(3238)	(-658)	(2580)	(+100)	(2680)	(0)	(2680)
WEIGHT EMPTY	9263	+7797	17060	+1860	18920	+190	19110.
CREW (2)	340	+20	360		360		360
UNUSABLE FUEL	170	-30	140		140		140
OIL & TRAPPED OIL	24	+86	110	+20	130		130
EXTERNAL TANKS	—		—		—		—
GUN INSTALLATIONS	—		—		—		—
WEAPON INSTALLATIONS	—		—		—		—
CREW EQUIPMENT	—		—		—		—
NON-EXP. USEFUL LOAD	(534)	(+76)	(610)	(+20)	(630)		(630)
CONTINGENCY WT.	—	+1860	1860	+190	2050	+20	2070
OPERATING WEIGHT	9797	+9733	19530	+2070	21600	+210	21810
FUEL - INTERNAL (WING)	5818	+272	6090		6090		6090
FUEL - EXTERNAL	—		—		—		—
H_2O + ALCOHOL	—	+240	240	-240	—		—
FLIGHT TEST EQUIP.	—	+2500	2500		2500		2500
PASSENGERS (4)	680	-680	—		—		—
BAGGAGE (INCL. PARACH.)	420	-420	—		—		—
GROSS WEIGHT (FULL WING)	16715	+11645	28360	+1830	30190	+210	30400

* AS DESCRIBED IN REFERENCE 1

Table 1.12-1 SUMMARY GROUP WEIGHT STATEMENT

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	BASIC T-39A *		
	WEIGHT (LBS)	CENTER OF GRAVITY HORIZONTAL BODY STATION (IN.)	CENTER OF GRAVITY VERTICAL WATER LINE (IN.)
WING	1661	265.9	72.5
HORIZONTAL TAIL	220	472.6	121.0
VERTICAL TAIL	91	471.6	153.7
BODY	1780	217.3	93.9
MAIN GEAR (GR. DOWN)	467	254.0	58.0
NOSE GEAR (GR. DOWN)	147	98.0	56.0
LAUNCH AND RECOVERY GEAR	—	—	
ENG. SECTION OR NACELLE	372	334.0	101.2
WING SKID	15	223.0	59.0
STRUCTURE	(4753)	(260.0)	(84.6)
ENGINE (2) J60-P-3A	980	336.0	100.0
THRUST REVERSER	—	—	—
ENGINE ACCESSORIES	17	330.2	90.2
ENGINE CONTROLS	30	194.0	96.0
STARTING SYSTEM	22	269.0	118.0
FUEL SYSTEM	195	292.0	87.0
EXHAUST	28	384.0	100.0
PROPULSION	(1272)	(325.7)	(98.1)
FLIGHT CONTROLS	326	236.1	88.8
AUXILIARY POWER PLANT	—	—	—
INSTRUMENTS	166	157.9	98.9
HYDRAULIC & PNEUMATIC	145	341.3	86.5
ELECTRICAL	924	262.4	103.2
AVIONICS	464	97.5	94.5
ARMAMENT	—	—	—
FURNISHINGS & EQUIPMENT	877	202.3	100.6
AIR CONDITIONING	279	257.2	96.9
ANTI-ICING	54	237.0	105.0
LOAD & HANDLING	3	210.3	108.3
FIXED EQUIPMENT	(3238)	(219.2)	(98.3)
WEIGHT EMPTY	9263	254.8	91.2
CREW (2)	340	115.0	101.0
UNUSABLE FUEL	170	248.4	63.1
OIL & TRAPPED OIL	24	324.0	110.0
EXTERNAL TANKS	—	—	—
GUN INSTALLATIONS	—	—	—
WEAPON INSTALLATIONS	—	—	—
CREW EQUIPMENT	—	—	—
NON-EXP. USEFUL LOAD	(534)	(166.9)	(89.3)
OPERATING WEIGHT	9797	250.0	91.1
FUEL - INTERNAL (WING)	5818	256.0	72.0
FUEL - EXTERNAL			
PASSENGERS (4)	680	253.0	102.0
BAGGAGE (INCL PARA.)	420	204.7	102.5
GROSS WEIGHT (PRI. MISSION)	16715	251.1	85.2

LEMAC @ BSTA 225.11
MAC LENGTH = 100.6 IN.
WL 100 IS 28.46 IN. ABOVE WING CHORD PLANE @ SIDE OF BODY
* AS DESCRIBED IN REFERENCE 1

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TABLE 1.12-2 WEIGHT AND BALANCE DATA T-39A

	MODEL 1041-135-2R WEIGHT (LBS)	HORIZONTAL BODY STATION C.G. (IN.)	
WING	1850	266.6	
HORIZONTAL TAIL	350	497.0	
VERTICAL TAIL	190	467.9	
BODY	2270	206.1	
MAIN GEAR (GR. DOWN)	680	284.0	
NOSE GEAR (GR. DOWN)	210	80.0	
LAUNCH AND RECOVERY GEAR	—	—	
ENG. SECTION OR NACELLE	2270	290.2	
WING SKID	—	—	
STRUCTURE	(7820)	(267.6)	
ENGINE (2) XT701-AD-700	2270	391.0	
THRUST REVERSER	—	—	
ENGINE ACCESSORIES	—	—	
ENGINE CONTROLS	30	218.0	
STARTING SYSTEM	20	368.0	
FUEL SYSTEM	180	267.1	
FANS (3) 62" HAM. STD.	960	233.7	
DRIVE SYSTEM	2070	288.3	
EXHAUST & DEFLECTORS	980	293.4	
H2O INJECTION SYSTEM	150	375.0	
PROPULSION	(6660)	(317.5)	
FLIGHT CONTROLS	530	266.6	
AUXILIARY POWER PLANT	—	—	
INSTRUMENTS	240	180.8	
HYDRAULIC & PNEUMATIC	220	355.0	
ELECTRICAL	460	276.0	
AVIONICS	260	97.5	
ARMAMENT	—	—	
FURNISHINGS & EQUIPMENT	480	151.3	
AIR CONDITIONING	330	206.2	
ANTI-ICING	50	240.1	
LOAD & HANDLING	10	210.3	
FIXED EQUIPMENT	(2580)	(220.9)	
WEIGHT EMPTY	17060	280.0	
CREW (2)	360	115.0	
UNUSABLE FUEL	140	227.7	
OIL & TRAPPED OIL	110	299.0	
EXTERNAL TANKS	—	—	
GUN INSTALLATIONS	—	—	
WEAPON INSTALLATIONS	—	—	
CREW EQUIPMENT	—	—	
NON-EXP. USEFUL LOAD	(610)	(174.0)	
CONTINGENCY WT.	1860	223.6	
OPERATING WEIGHT	19530	271.3	
FUEL - INTERNAL (WING)	6090	256.0	
FUEL - EXTERNAL	—		
H2O + ALCOHOL	240	375.0	
FLIGHT TEST EQUIP.	2500	59.6	
GROSS WEIGHT (FULL WING)	28360	250.2	

LEMAC @ BSTA 225.11
MAC LENGTH = 100.6 IN.

Table 1.12-3 WEIGHT AND BALANCE DATA 1041-135-2R

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	MODEL 1041-135-2A		
	WEIGHT (LBS)	CENTER OF GRAVITY	
		HORIZONTAL BODY STATION (IN.)	VERTICAL WATER LINE (IN.)
WING	1780	263.7	72.5
HORIZONTAL TAIL	350	457.0	221.0
VERTICAL TAIL	190	467.9	165.0
BODY	2320	206.1	93.0
MAIN GEAR (GR. DOWN)	680	284.0	35.0
NOSE GEAR (GR. DOWN)	210	80.0	30.0
LAUNCH AND RECOVERY GEAR	—	—	—
ENG. SECTION OR NACELLE	2350	291.7	121.0
WING SKID	—		
STRUCTURE	(7880)	(267.3)	(97.5)
ENGINE (3) XT701-AD-700	3410	371.3	119.0
THRUST REVERSER	—	—	—
ENGINE ACCESSORIES	—	—	—
ENGINE CONTROLS	45	247.0	100.0
STARTING SYSTEM	35	341.0	104.7
FUEL SYSTEM	230	272.1	86.7
FANS (3) 62" HAM. STD.	960	233.7	112.0
DRIVE SYSTEM	2620	280.9	111.5
EXHAUST & DEFLECTORS	1060	300.1	112.2
H_2O INJECTION SYSTEM	—	—	—
PROPULSION	(8360)	(319.6)	(113.9)
FLIGHT CONTROLS	530	266.6	98.5
AUXILIARY POWER PLANT	—	—	—
INSTRUMENTS	270	175.2	99.6
HYDRAULIC & PNEUMATIC	220	355.0	86.5
ELECTRICAL	460	276.0	103.0
AVIONICS	260	97.5	94.0
ARMAMENT	—	—	—
FURNISHINGS & EQUIPMENT	510	160.9	95.7
AIR CONDITIONING	360	214.0	97.0
ANTI-ICING	60	246.9	112.7
LOAD & HANDLING	10	210.3	108.3
FIXED EQUIPMENT	(2680)	(222.0)	(97.6)
WEIGHT EMPTY	18920	281.8	104.3
CREW (2)	360	115.0	101.0
UNUSABLE FUEL	140	227.7	63.0
OIL & TRAPPED OIL	130	300.7	117.4
EXTERNAL TANKS	—	—	—
GUN INSTALLATIONS	—	—	—
WEAPON INSTALLATIONS	—	—	—
CREW EQUIPMENT	—	—	—
NON-EXP.USEFUL LOAD	(630)	(178.4)	(95.9)
CONTINGENCY WT.	2050	223.6	99.8
OPERATING WEIGHT	21600	273.3	104.1
FUEL - INTERNAL (WING)	6090	256.0	72.0
FUEL - EXTERNAL	—		
FLIGHT TEST EQUIP.	2500	59.6	88.4
GROSS WEIGHT (FULL WING)	30190	252.1	96.3

LEMAC @ BSTA 221.45
MAC LENGTH = 105.17 IN.
WL 100 IS 28.46 IN. ABOVE WING CHORD PLANE @ SIDE OF BODY

Table 1.12-4 WEIGHT AND BALANCE DATA 1041-135-2A

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	MODEL 1041-135-2B		
	WEIGHT (LBS)	HORIZONTAL BODY STATION C.G. (IN.)	
WING	1780	263.7	
HORIZONTAL TAIL	350	497.0	
VERTICAL TAIL	190	467.9	
BODY	2320	206.1	
MAIN GEAR (GR. DOWN)	680	284.0	
NOSE GEAR (GR. DOWN)	210	80.0	
LAUNCH AND RECOVERY GEAR	—	—	
ENG. SECTION OR NACELLE	2350	291.7	
WING SKID	—	—	
STRUCTURE	(7880)	(267.3)	
ENGINE (2) XT701 + (1) T56	3610	372.8	
THRUST REVERSER	—	—	
ENGINE ACCESSORIES	—	—	
ENGINE CONTROLS	45	247.0	
STARTING SYSTEM	35	341.0	
FUEL SYSTEM	230	272.1	
FANS (3) 62" HAM. STD.	960	233.7	
DRIVE SYSTEM	2620	280.9	
EXHAUST & DEFLECTORS	1050	300.9	
H_2O INJECTION SYSTEM	—	—	
PROPULSION	(8550)	(316.7)	
FLIGHT CONTROLS	530	266.6	
AUXILIARY POWER PLANT	—	—	
INSTRUMENTS	270	175.2	
HYDRAULIC & PNEUMATIC	220	355.0	
ELECTRICAL	460	276.0	
AVIONICS	260	97.5	
ARMAMENT	—	—	
FURNISHINGS & EQUIPMENT	510	160.9	
AIR CONDITIONING	360	214.0	
ANTI-ICING	60	246.9	
LOAD & HANDLING	10	210.3	
FIXED EQUIPMENT	(2680)	(222.0)	
WEIGHT EMPTY	19110	283.0	
CREW (2)	360	115.0	
UNUSABLE FUEL	140	227.7	
OIL & TRAPPED OIL	130	300.7	
EXTERNAL TANKS	—	—	
GUN INSTALLATIONS	—	—	
WEAPON INSTALLATIONS	—	—	
CREW EQUIPMENT	—	—	
NON-EXP. USEFUL LOAD	(630)	(178.4)	
CONTINGENCY WT.	2070	273.6	
OPERATING WEIGHT	21810	274.3	
FUEL - INTERNAL (WING)	6090	256.0	
FUEL - EXTERNAL			
FLIGHT TEST EQUIP.	2500	59.6	
GROSS WEIGHT (FULL WING)	30400	253.0	

LEMAC @ BSTA 221.45
MAC LENGTH = 105.17 IN.

Table 1.12-5 WEIGHT AND BALANCE DATA 1041-135-2B

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GROUP WEIGHT STATEMENT
DIMENSIONAL AND STRUCTURAL DATA

Name ____________ Page ____________

Date 10/76 Model 1041-135-2A

Report ____________

1	WING, ROTOR & TAIL GROUPS	SPAN OR RADIUS (FT)	SPAN AT **** 25% CHORD	THEO ROOT ** CHORD (IN)	MAX THICK ** ROOT CHORD (IN)	THEO TIP CHORD (IN) **	MAX THICK ** TIP CHORD (IN)
2	WING	39.75	45.67	144.0	15.8	54.0	5.1
3							
4	MAIN ROTOR (Blades/Rotor)						
5	TAIL ROTOR (Blades/Rotor)						
6	HORIZ. TAIL	15.58	17.83	75.0	7.50	24.0	2.4
7	VERT. TAIL	8.17	8.83	85.0	8.50	37.5	3.8
8							
9	AREAS - (Sq. Ft.)	Wing	MAIN ROTOR BLADE AREA	TAIL ROTOR BLADE AREA	Horiz. Tail	Vert. Tail	Dorsal
10	(Theo. for Wing & Rotor, All Others Exposed)	327.9			64.3	41.7	6.2
11		Speed Brks.	Flaps (L.E.)	Flaps (T.E.)	Slats	Spoilers	Ailerons
12	AREAS - (Sq. Ft.)			40.3	36.3		16.4
13	BODY & NACELLE GROUPS	Length (Ft.)	Depth (Ft.)	Width (Ft.)	WETTED AREA (SQ FT)	Vol. (Cu. Ft.)	VOL PRESS (CU FT)
14	FUSELAGE or HULL	48.0	6.42	5.75	678.0		0
15	BOOMS						
16	NACELLES	13.33	6.75	6.75	214.0 EA		
17							
18							
19	ALIGHTING GEAR GROUP	Length - Oleo Ext.		Oleo Travel		Length - Arrest Hook	
20		Axle to ℄ Trunnion		Ext. to Collapsed		Hook Trunnion to Pt.	
21	LOCATION	MAIN	NOSE	MAIN	NOSE		
22	DIMENSIONS (Inches)	65.0	67.0	14.0	14.0		
23							
24	PROPULSION GROUP	S.L.S. THRUST IN LBS./ENG. WITH AFTERBURNER		S.L.S. THRUST IN LBS./ENG WITHOUT AFTERBURNER		MAX. S.L.S. SHAFT HP	SHAFT RPM AT MAX HP
25	ENGINES						
26	MAIN			12800		7540 ▷	11500 ▷
27	AUXILIARY						
	ROTOR DRIVE SYSTEM	Design H.P.	Input R.P.M.	OUTPUT R.P.M. AT ROTOR	INTER ROTOR R.P.M.	NUMBER GEAR BOXES	
		7540 ▷	11500 ▷			5	
30		Protected		Unprotected		Integral	
31	FUEL - INTERNAL *** LOCATION	No. Tanks	Gallons	No. Tanks	Gallons	No. Tanks	Gallons
32	WING					3	895
33	FUSELAGE						
34	- EXTERNAL ***						
35							
36	OIL = ENGINE			3	17.3		
37	ELECTRICAL & LOAD & HANDLING GROUPS	QUAN. MAIN GENERATORS	GENERATOR OUTPUT - D.C.	GENERATOR OUTPUT - A.C.		CARGO FLOOR AREA	
38		2	.	20 KVA			
39		BODY PLUS INT. CONTENT - LBS.	EXTERNAL WT. ON BODY	FUEL IN WINGS (LBS)		DESIGN GROSS WEIGHT	Ult. L.F.
40	STRUCTURAL DATA - CONDITION						
41	FLIGHT - MANEUVER					26410	3.75
42	- GUST					24100	3.92
43	LANDING - DESIGN VL					25500	4.50
44							
45	MAX. GROSS WITH ZERO WING FUEL					24100	3.92
46	CATAPULTING						
47	LIMIT LANDING SINK SPEED (Ft./Sec.)						12.0
48	LIFT ASSUMED FOR LANDING DESIGN CONDITION (% W)						
49	STALL SPD. - LDG. CONFIG. - POWER OFF						
50	PRESSURIZED CABIN - ULT. DESIGN PRESS. DIFFERENTIAL - FLIGHT P.S.I.						
51	ROTOR TIP SPD AT DESIGN LIMIT	R.P.M.	Power	Ft./Sec.	CONTRACTOR DESIGN FACTOR		
52							
53	% DESIGN LOAD	Wing		Rotor		Rotor	
54	DESIGN SPEED AT S.L. (Knots)	Level	300	Dive	330		
55	DESIGN SPD. AT OTHER ALTITUDES		Alt.		Alt.		
56			21000	330 KTS	35000	224 KTS	
57	DCPR WEIGHT (Airframe)						

*Nose to aft tip of fuselage (excluding equipment protuberances)
**Parallel to ℄ at ℄ Aircraft for Wing & Tail. Insert inches from ℄ Rotor for Rotors.
***Total Unable Capacity.
****Insert inches from ℄ Rotor to Blade Attachment for Rotors.

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▷ INTERMEDIATE SLS 90°F

WEIGHT CONDITION	WEIGHT (LBS)	INERTIA (SLUG – FT²)			PRODUCT OF INERTIA (SLUG·FT²)	PRINCIPAL AXIS SLOPE* (DEGREES)
		PITCH	ROLL	YAW		
OPERATING WT.	21600	65280	18092	75130	5081	-5.05
OPERATING WT. PLUS FLIGHT TEST EQUIP.	24100	89179	18541	99046	6693	-4.72
MAX. VTOL WEIGHT (THRUST DETERMINED)	27400	91101	19565	100834	7059	-4.93
MAX. GROSS WEIGHT (FULL WING FUEL)	30190	91536	41960	123546	6653	-4.63

* NEGATIVE ANGLE IS NOSE DOWN

TABLE 1.12-1A INERTIA DATA - MODEL 1041-135-2A

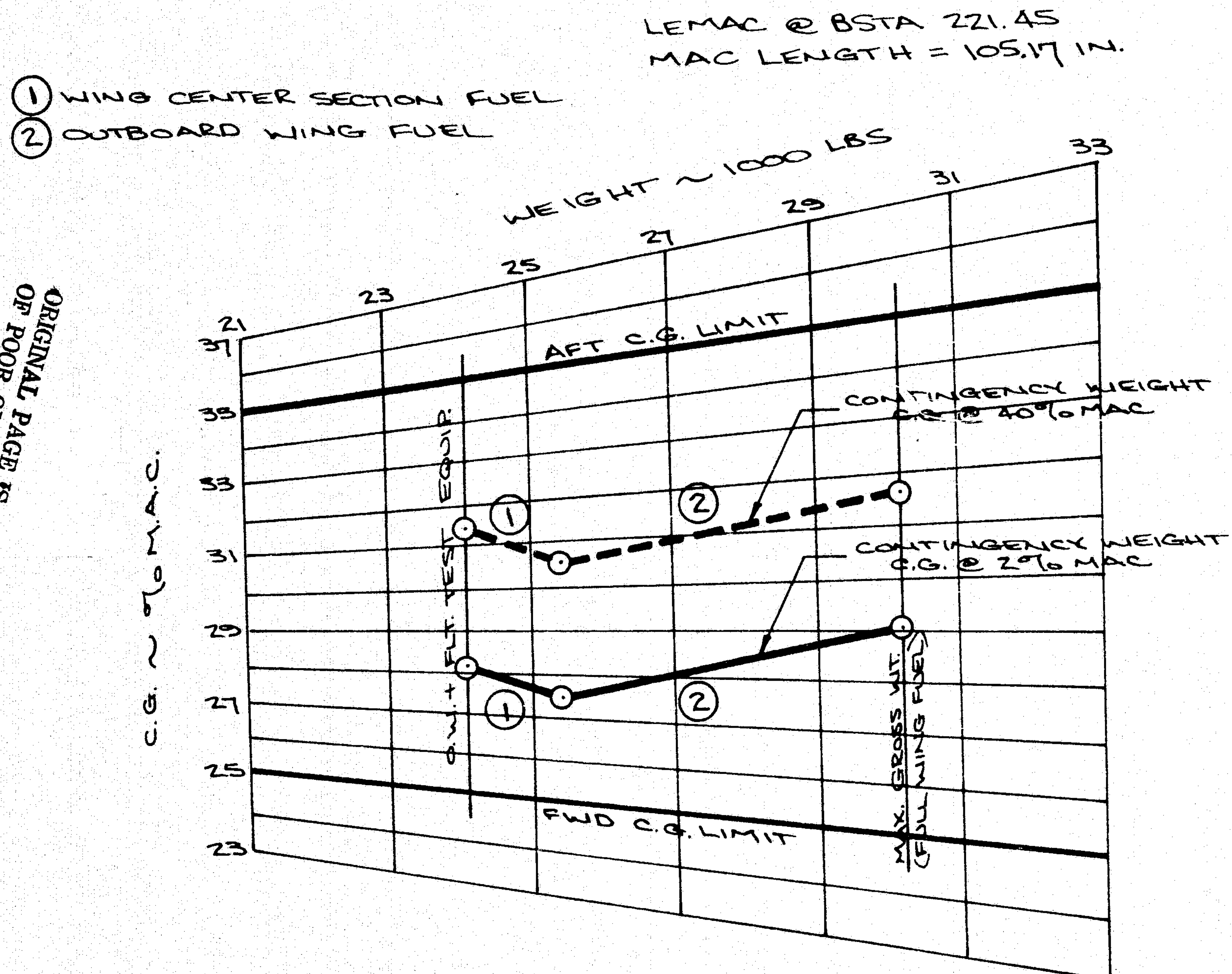

FIGURE 1.12-1 C.G. TRAVEL FOR MODEL 1041-135-2A

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1.13 Low Speed Flight Envelope and Aerodynamic Performance

1.13.1 Low Speed Flight Envelope

The low speed flight envelope for the 1041-135-2A for a VTOL weight of 25300 lbs is presented in Figure 1.13-1. The altitude boundaries at lower Mach numbers represent envelopes of 100 fpm rate-of-climb capability for a 90°F day where engine tilt angle is decreasing with increased Mach number. The envelopes are shown for all engines operating and critical engine out, and show that the 1041-135-2A has vertical capabilities in both cases. The envelopes were determined to an altitude of 25,000 feet which encompasses the NASA guidelines for cruise flight. At higher Mach numbers the low speed flight is limited by a part-flap placard of 225 KEAS. For completeness, the high speed (flaps-up) flight envelope is also shown. In the cruise configuration, the lower speed boundary is 1.2 V_S using a clean wing C_{Lmax} of 0.9. The higher speed boundary is V_{MO} = 300 KEAS to an altitude of 22,000 feet and Mach number limit of M_{MO} = 0.7. The overlap of the low speed and high speed envelopes of Figure 1.13-1 represents the minimum transition region for the 1041-135-2A. In that the T-39 has an aerodynamically deployed leading edge device which produces an estimated C_{Lmax} of 1.3, the transition region actually spans from a Mach number of 0.2B to 0.34 at sea level.

1.13.2 V/STOL Performance

The 1041-135-2A, as indicated above has adequate thrust for vertical flight with critical engine out as well as with all engines operating. The excessive thrust (all engine T/W = 1.14) provides flexibility in demonstrating vertical flight for a range of thrust-to-weight ratios.

An STO analysis indicated that the 1041-135-2A requires only 200 feet of ground roll due to the excessive thrust and large wing. The analysis was conducted using a flaps-down C_{Lmax} of 1.63 and a gross weight of 28630 lbs. This weight includes the fuel required for conducting 11 STO demonstrations in 60 minutes. Various combinations of flap and power settings can be selected to demonstrate a spectrum of configurations as well as determining pilot reactions for ground rolls above and below 400 feet.

Fuel usage for the six minute VTOL demonstration circuit shown in Figure

1.13-2 on a 90°F day and for a gross weight of 25300 lbs. has been estimated at 320 pounds. The circuit is divided into three segments as follows:

(1) The first segment includes the take-off and transition to a loiter configuration (flaps up) at an altitude of 1,000 feet. Approximately

a thrust-to-weight ratio of 1.05 and requires 63 lbs. of fuel.

(2) The second segment is cruising the 4.7 NMI from the end of take-off transition to the beginning of approach. This segment is flown in the loiter configuration at a speed equal to 1.2 V_s (approximately 160 KTS) requiring a time of 4.5 minutes and 133 lbs of fuel.

(3) The remaining time of one minute is spent in approach, hover and vertical landing. During this segment the thrust is equal to the weight and the fuel burned amounts to 124 lbs.

Fuel reserves for 4 minutes of hover with critical engine out has been estimated at 570 lbs. This figure is also based on a gross weight of 25,300 lbs and for a 90°F day.

1.13.3 Ferry Range

The ferry range of the 1041-135-2A has been calculated to be 625 nautical miles. This range is for a takeoff gross weight of 30,540 lbs. with a full fuel load of 6,090 lbs. and a payload of 2,500 lbs. Take-off allowances for warm-up, take-off, and acceleration to a 250 KEAS climb speed were included. The fuel reserves used were 10% of initial fuels plus fuel for 10 minutes of loiter at the end of the mission. The ferry range was calculated for a cruise altitude of 10,000 feet and a cruise Mach number of 0.45 which is near optimum for this altitude. The cruise altitude of 10,000 feet was selected in consideration of the limited oxygen system which will be provided with the demonstrator.

T-39 TECHNOLOGY DEMONSTRATOR OPERATING ENVELOPE

(1041-135-2A)

GROSS WT = 25300 LB.

90°F DAY

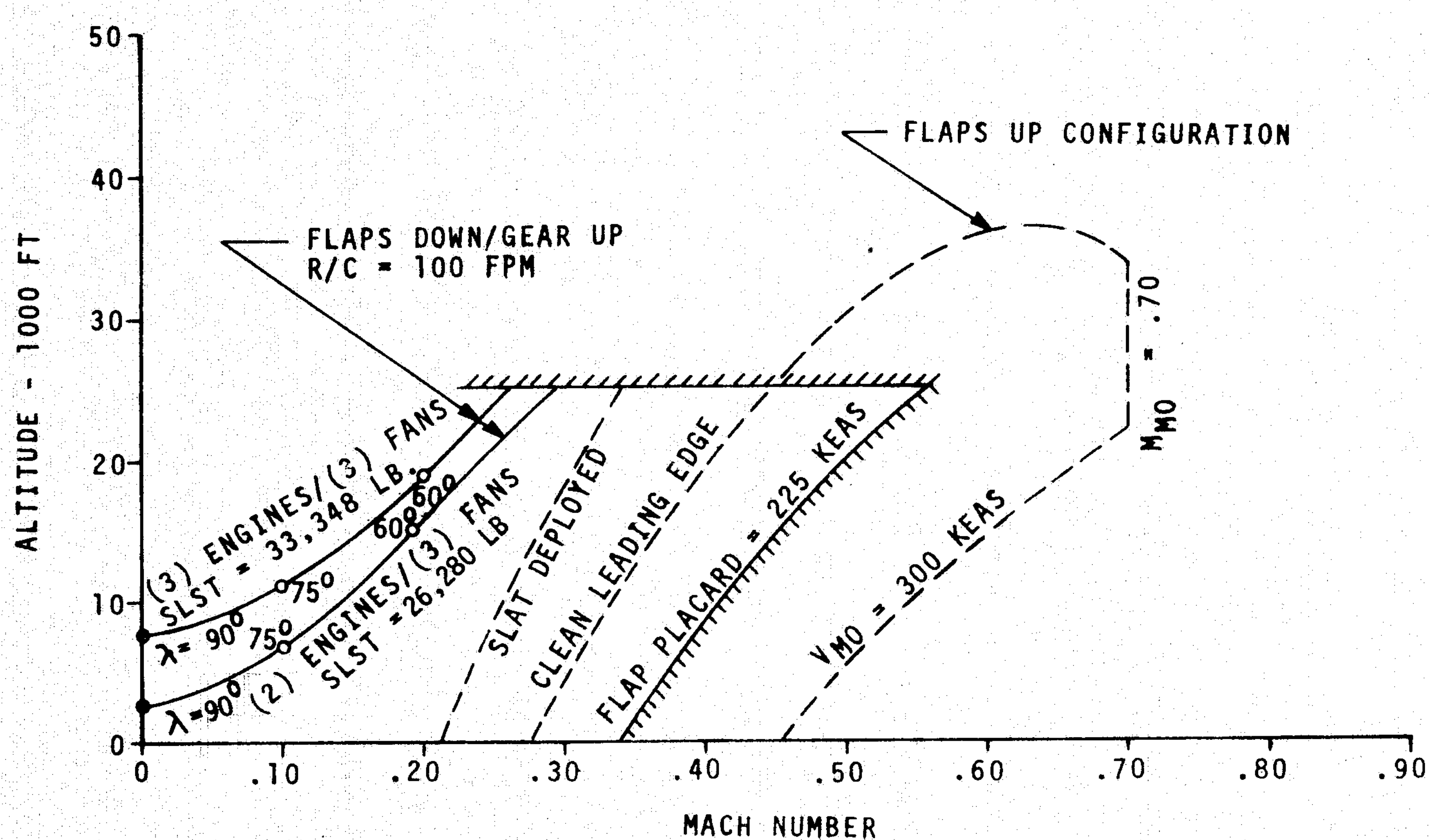


FIGURE 1.13-1

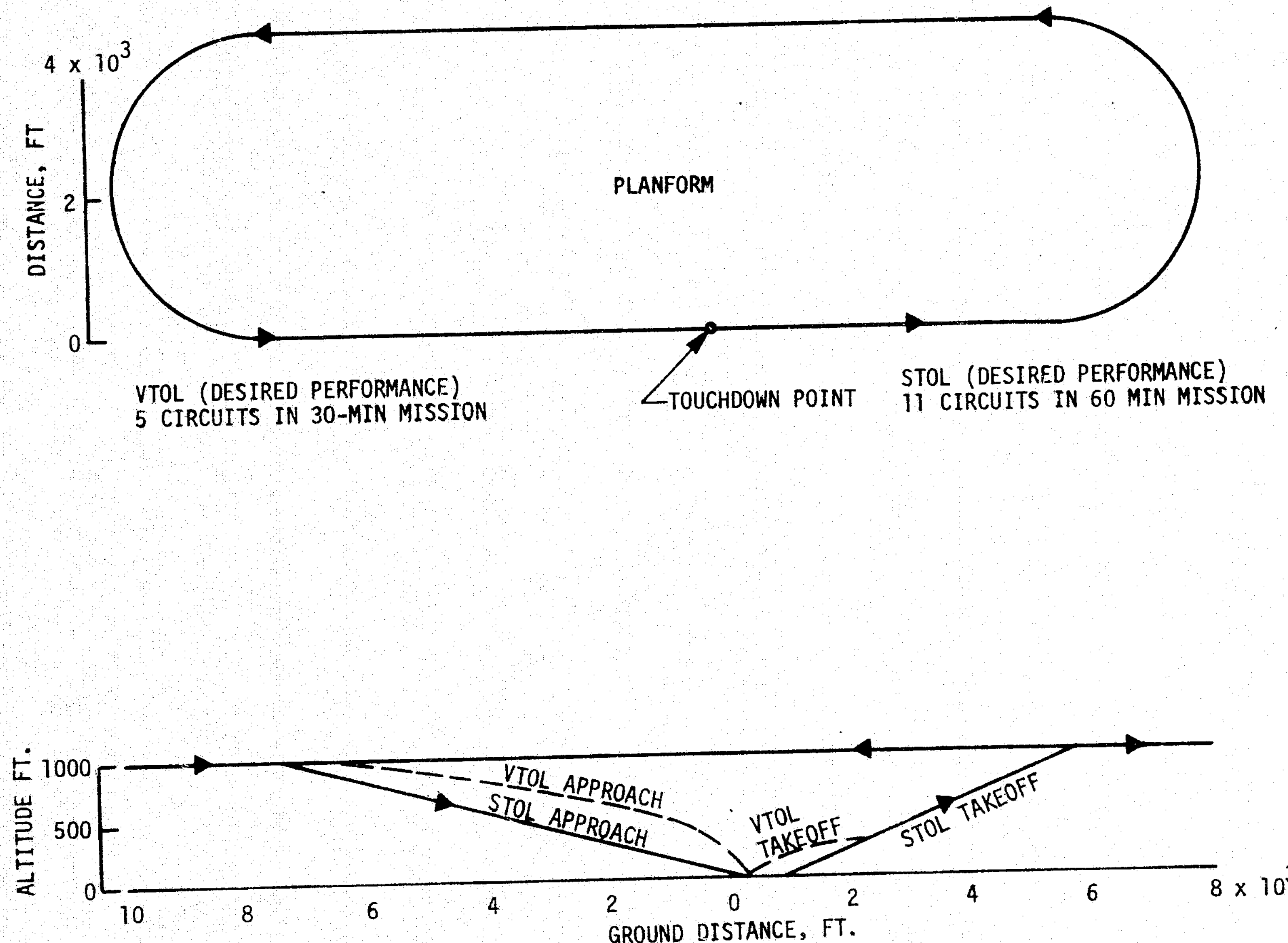


FIGURE 1.13-2 LCFA TECHNOLOGY AIRPLANES, TYPICAL TERMINAL AREA TEST MISSIONS

1.14 TESTING

The V/STOL Research Technology Demonstrator (RTA) airplane will be subjected to a step-by-step sequence of tests of increasing severity or complexity. Each test will be more comprehensive than the previous one. This approach will minimize risk by exposing deficiencies before they can impact the total system; it will permit personnel to become thoroughly familiar with the system on a timely basis, easing peaks in manpower requirements, and it will generate a body of baseline data to support subsequent flight-test and development programs.

The low-risk progression of tests will begin with appropriate component tests performed by the respective manufacturers. The propulsion system and that portion of the flight control system that is required for powered lift flight will be subjected to full-scale integration testing on an "iron-bird" rig. The scope of the rig test will permit demonstration of propulsion system integrity and baseline installed performance. A realtime simulation of airframe dynamics will be used to close the control loop to check out the flight control laws, electronics, and software.

The engines, fans and power train will be subjected to approximately 400 hours of operation under simulated hover and low speed flight conditions during the rig test program. This assumes one-shift operation of the facility. An optional alternative program would provide 600 hours of propulsion system operation on the iron bird by performing additional endurance testing on the second shift while primary test goals are pursued on the first shift.

The flight controls and propulsion system will be installed in the modified airframe and subjected to comprehensive functional checkout on the ground.

Sufficient flight testing will be conducted to demonstrate that the aircraft is capable of supporting the NASA Research Program. Basic CTOL performance will be demonstrated in a 30-hour test sequence. An optional 100-hour flight test would include demonstration of powered lift.

1.14.1 Fan and Engine Testing

All fans, engines and associated fan and engine gearing will be Government Furnished Equipment (GFE) for flight testing and for "iron bird" and development testing. Manufacturers will conduct development and qualification testing prior to delivery of the fan and engine to provide hardware qualified for prototype flight testing. Preliminary estimates from DDA in August 1975 indicated 16 months from go-ahead to start lift/cruise fan engine development testing and 30 months from go-ahead to provide the first Prototype Preliminary Flight Rating Tested (PPFRT) lift/cruise system. (If later estimates cause this delivery schedule to change, adjustment to the test schedule may be necessary).

The XT701 engine has already passed PPFRT but qualification of the engine and fan systems must be accomplished for the demonstrator airplane instaallation and operation effects.

The most significant test verification features are as follows:

A) Design thrust and power,
B) Design fan pressure ratio and flow rate,
C) Design RPM,
D) Fan blade pitch thrust response,
E) Engine response,
F) Distortion tolerance,
G) Reduction and bevel gear performance, and
H) Fan and engine compatibility and reliability

During the development test program, the fan and engine manufacturers will be supplied by the airframe contractor with analytical and test data as required to optimize the propulsion system installed performance. Specific areas of support include inlet distortion data, fan nozzle area and contour, yaw control blade design and control system requirements. Inlet distortion data from the model testing are given in this document and further data will be available from the inlet/Q-fan test in the 40 x 80 NASA Ames wind tunnel.

Based on fan pressure ratios and areas obtained from fan and engine company performance analyses, the airframe contractor will develop fan nozzles and engine core nozzles to optimize installed performance. Model testing will be performed both statically and in the wind tunnel to determine nozzle performance for the center engine, front lift fan installation, and the lift/cruise installation.

1.14.2 Flight Control System

This complete flight controls system comprises environmental flight controls, employed during wing borne flight and CTOL, and integrated flight propulsion controls employed during VSTOL maneuvers, hover and transition.

The control system test program consists of:

A) Hardware and flight software components,

B) Development and integration testing on the iron bird, and

C) Functional testing of the flight control system when installed in the airplane.

The laboratory test program will verify the flight control electronics hardware and software in conjunction with a computer simulation that duplicates the airplane and propulsion system dynamics. The actual propulsion hardware will replace the propulsion simulation in the iron bird rig tests. Forces and moments generated by the propulsion system will be measured and integrated by a real-time computer model of the airframe to evaluate the powered-lift control laws, hardware and software under realistic operating conditions. Final verification prior to flight will be conducted in ground and taxi tests with the actual flight controls electronics and software, avionics, and flight test instrumentation installed in the airplane.

1.14.2.1 <u>Flight Controls Test Requirements.</u> The VSTOL capability of the RTA airplane imposes the requirement for operating the aircraft in two distinct operating modes - powered lift (V/STOL) and aerodynamic lift (cruise and CTOL). The flight control system must accordingly be tested in both modes of operation. The test program will be organized to reflect the dual requirement.

A) Wing Borne Flight Controls Testing

The propulsion iron bird rig will incorporate the power actuators for the aerodynamic conctrol surfaces. A series of laboratory tests and special purpose tests, conducted with various flight control assemblies and corresponding airplane control surfaces, are required for flight control system and control actuator loop stability verification and validation. The following principal needs are covered by this method:

(a) Verification of performance and stability characteristics of the control system actuation systems and associated mechanical system elements before functional test.

(b) Verification of performance characteristics (compliance, freeplay, friction, centering, linearity, etc.) of the mechanical flight control system,

(c) Partially integrated performance verification testing of the EFCS, MFCS, hydraulic power systems, and electric power systems before functional test.

An organizational test structure diagram is developed in Figure 1.14-1. Its purpose is to show the integration of the various flight control test programs which demonstrate that performance and flight safety requirements are satisfied. The laboratory and special purpose test programs will cover many of the engineering and functional test requirements prior to installation on the EFCS electronic flight control system on the airplane. Additional engineering tests on the airplane after the installation will ensure that the installed equipment meets the design requirements. Each of the test categories is described in further detail in the subsequent paragraphs.

B) Powered Lift Flight Controls Testing

The ability to maintain precise, positive, and reliable control of the airplane through modulation of propulsion forces must be developed and demonstrated prior to commiting the airplane to powered lift flight. The following items must be explored in depth.

ORGANIZATION TEST STRUCTURE

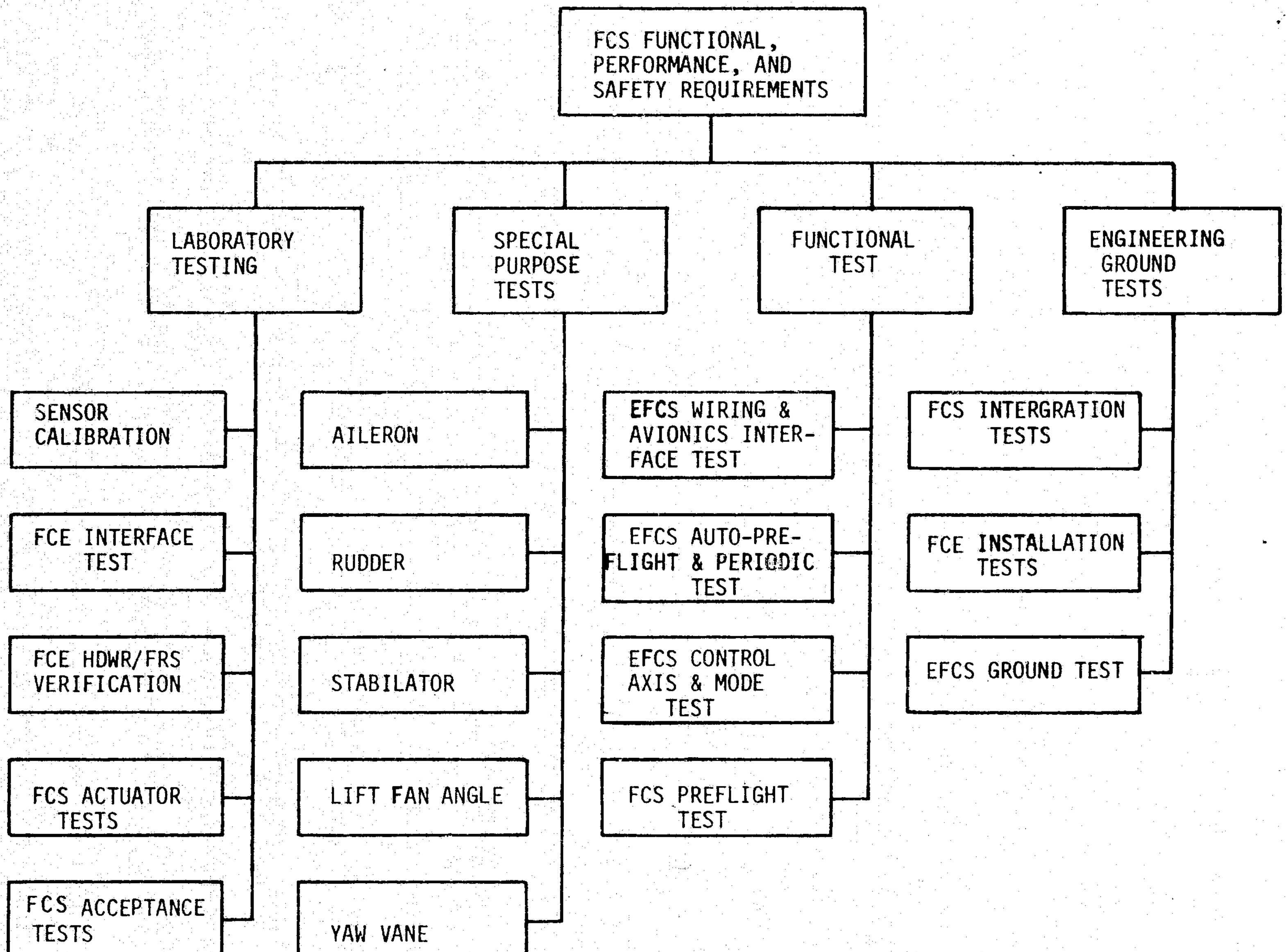


FIGURE 1.14-1

(a) Response and accuracy of propulsion system position loops - fan pitch, nacelle rotation, yaw vanes,

(b) Propulsion system response - power transients control forces and moments,

(c) Flight control electronics - hardware and software,

(d) Powered Lift Control Laws and Flight Procedures, and

(e) Crew interfaces - displays and controls.

These requirements will be addressed through a combination of laboratory, simulation, and propulsion rig tests. Components must be flight qualified through analysis, tests by manufacturers, or by use of components qualified for other programs. Large scale real-time simulations with man-in-the-loop capability (including the NASA-Ames flight simulator) must be used to develop and demonstrate control laws, software, and selected portions of the electronics hardware. A significant portion of the propulsion rig test program will be devoted to testing of powered lift flight controls. (See Section 1.14.4).

1.14.2.2 Flight Controls Laboratory Test. The objectives of the laboratory test are to validate the Flight Control Electronics (FCE) and software and to support the RTA airplane ground and flight tests. These objectives will be achieved by testing of the FCE with a development version of the flight software, using simulated sensors, output devices, and airframe dynamics to close the control loops. The following tests will be performed:

A) No load performance tests of the flight control actuator servos will be conducted using simple hold-down fixtures.

B) Closed-loop tests of the FCE and software will be conducted using simulated aircraft dynamics, aerodynamics control surface servos, propulsion control servos, aircraft motion sensors, and pilot command sensors to close the various loops. These tests will be used for development and verification of flight control laws and redundancy concepts.

C) Electrical interface testing of flight control sensors, output servos, and flight test aircraft data acquisition will be conducted. Sensor tracking and failure monitoring of critical flight control sensors will be studied in these tests.

1.14.2.3 Special Purpose Tests. A series of tests will be conducted on the installed electric command (EC) servos, the manual flight control elements, and the surface and lift fan power control actuators prior to installation of the triplex digital flight control electronics on the airplane. The objective is to obtain performance data on certain critical control elements and to demonstrate and correct any electrical or mechanical problems at an early point in the RTA modification program. The combination of special purpose electronics and selected servos defined as special purpose tests, are as follows:

1) Aileron control system.
2) Elevator control system.
3) Rudder control system.
4) Lift fan angle control system.
5) Yaw vane control system.

These tests will be conducted at various stages of the airplane assembly and will require interfacing the analog electronics with the airplane wiring and electro-hydraulic control elements.

1.14.2.4 Functional Testing. Functional tests will be conducted on both of the airplanes to ensure that equipment and mechanism meet functional requirements in each airplane in the actual airplane environment.

The EFCS wiring and avionics interface tests will ensure that the EFCS wiring is validated for power distribution and that the interface components between the FCE and avionics are properly installed. Automatic preflight and periodic tests will demonstrate that the EFCS self interrogation modes are fully operable. Control axis and mode tests will use the EFCS preflight test capabilities. The tests will determine that EFCS control axis modes are properly installed, interconnected and operate as designed. The functional tests will comprise operation in wing borne and powered lift modes, with propulsion system in operation, when appropriate.

Preflight and periodic test program will be used for the RTA airplane functional test requirements. Since these tests are of great importance, an outline of them follows:

A) Preflight Tests

(a) Startup

- o Test initiation
- o Software program identification checks
- o Clearing of erroneous failure identification

(b) Automatic Tests

- o Interchannel loop tests
- o Sensor checks
- o Hardware overflow checks
- o Analog loop tests
- o Discrete output tests
- o Discrete input tests

(c) Manual Action Tests

- o Vertical gyro tests
- o Air data tests
- o Stabilator trim
- o EFCS master caution
- o Flap position
- o Sensor (α, β vanes, etc.)

(d) Manual End to End Tests

- o Stabilizer trim through pilot and copilot trim switches
- o Pitch and roll tests:
  - o Stabilator series electrical command servo (SECS)
  - o Aileron SECS
  - o Flap position
  - o Stabilizer position
  - o Engine position
  - o Yaw vane position (3 fans)
  - o Lift fan blade angle position (3 fans)
  - o Throttle position
  - o Rudder position

B) Periodic Tests

Until experience has been gained, the periodic tests will be required as preflight tests, as a preliminary to all flights as follows:

- o Unscheduled trim warning
- o FCE interlocks
- o Column trim limit switches
- o Stabilator trim limit switches
- o Vertical gyros (cut-outs)
- o Air ground switches
- o Engine position limit switches

1.14.3 Testing of Propulsion Drive System

The objective of the propulsion drive system tests is to verify that the installation will perform reliably in a simulated flight environment. The testing outlined here will, in conjunction with the integrated system testing described in the next section of this document, provide a comprehensive life cycle and functional check-out under realistic operating conditions. Successful completion of this test program will provide confidence that airplane installation will support the projected 500 hour flight test program.

1.14.3.1 Selection of a Test Method. A full-system ground test is considered a necessary preliminary to flight. This approach is consistent with past practice in development and testing of shaft-driven helicopter drive systems. Figure 1.14-2 shows a YUH-61A tiedown aircraft used in the development of the dynamic system. Figure 1.14-3 shows an iron-bird concept used in the development of the HLH drive train. The iron-bird consists of a structural framework, in which the components are mounted in their correct orientation relative to the aircraft and which are connected thru locally simulated aircraft structure.

FIGURE 1.14-2 YUH-61A TIEDOWN AIRCRAFT

FIGURE 1.14-3 IRON-BIRD CONCEPT HLH DRIVE TRAIN TEST RIG

1.14.3.2 Capabilities. The test rig will have the following capabilities.

- o Full power absorption at any fan
- o One engine inoperative
- o Control response rate (power transient) comparable to the aircraft
- o Full length drive shafting
- o Lube cooling system comparable to the aircraft. (Ducts, intakes, compartmentation around cooler and transmissions).
- o Full tilting provisions for engine transmission pods
- o Simulated aircraft structure around mount points
- o Comparable pod mounting stiffness

1.14.3.3 Special Test Conditions. The possibility of conducting tests under some of the following conditions will be considered.

A) Flight attitudes and accelerations affect the lubrication and scagenge system primarily by potentially unwetting the pump intakes and oil drain paths. It may be possible to mount the lube system development stand on gimbals so as to simulate operation of the extremes of roll and pitch angles. Accelerations due to aircraft maneuvers must be recognized during design, and can be verified only during actual aircraft operation.

B) Fuselage deflections affect shaft flexible couplings. Individual component relative motion can be calculated and the coupling mount points can be offset in the test rig to provide good simulation of the operating conditions.

C) This altitude and forward speed primarily affect oil cooling and foaming. The oil cooling system will be designed primarily for hovering at sea level on a hot (125^{o}F) day. At higher altitudes and forward speed conditions will generally be less critical because temperatures are lower and total power requirements are much less. The greater air-to-oil temperature differential at higher altitudes compensates for the reduced mass flow of air. All conditions will be examined and critical conditions simulated prior to flight.

1.14.3.4 Drive System Test Program. The drive system will be developed in individual component tests where possible, and complete system testing will be employed where interactions need to be tested. The following tests will be planned.

A) One specimen of each gear will be excited electromagnetically or by air-blast to verify the calculated resonant frequencies and mode shapes.

B) One gear box assembly of each type will be instrumented with gear tooth strain gages and loaded statically to various levels to full torque. The gears will be rotated slowly through mesh and the load distribution across the teeth will be recorded. (See Figure 1.14-4.

C) One gear box assembly of each type will be mounted in a no-load lube system development stand as shown by Figures 1.14-5 and 1.14-6. The gear box will be operated at full rpm to the extremes of attitude. Gear gaffles and jets will be adjusted to optimize lubrication and reduce windage losses.

D) One aircraft set of gearboxes and shafts will be instrumented with gear tooth strain gages, torque gages and readout equipment. The system will be operated at 25, 50, 75 and 100% torques at stabilized temperature for sufficient time to record gear tooth stresses. The system will also be operated at essentially zero torque and shaft motion will be monitored.

E) One aircraft ship set of gear boxes and shafts will be mounted in the integrated test rig, (Figure 1.14-7) and subjected to the load schedule of Figure 1.14-8. The load cycle will be:

Hover	60 min
OEI #3	18 min
OEI #2	9 min
Climb	24 min
OEI #1	9 min
	120 min (2 Hours)

STRAIN SURVEY FIXTURE

"T"- BOX

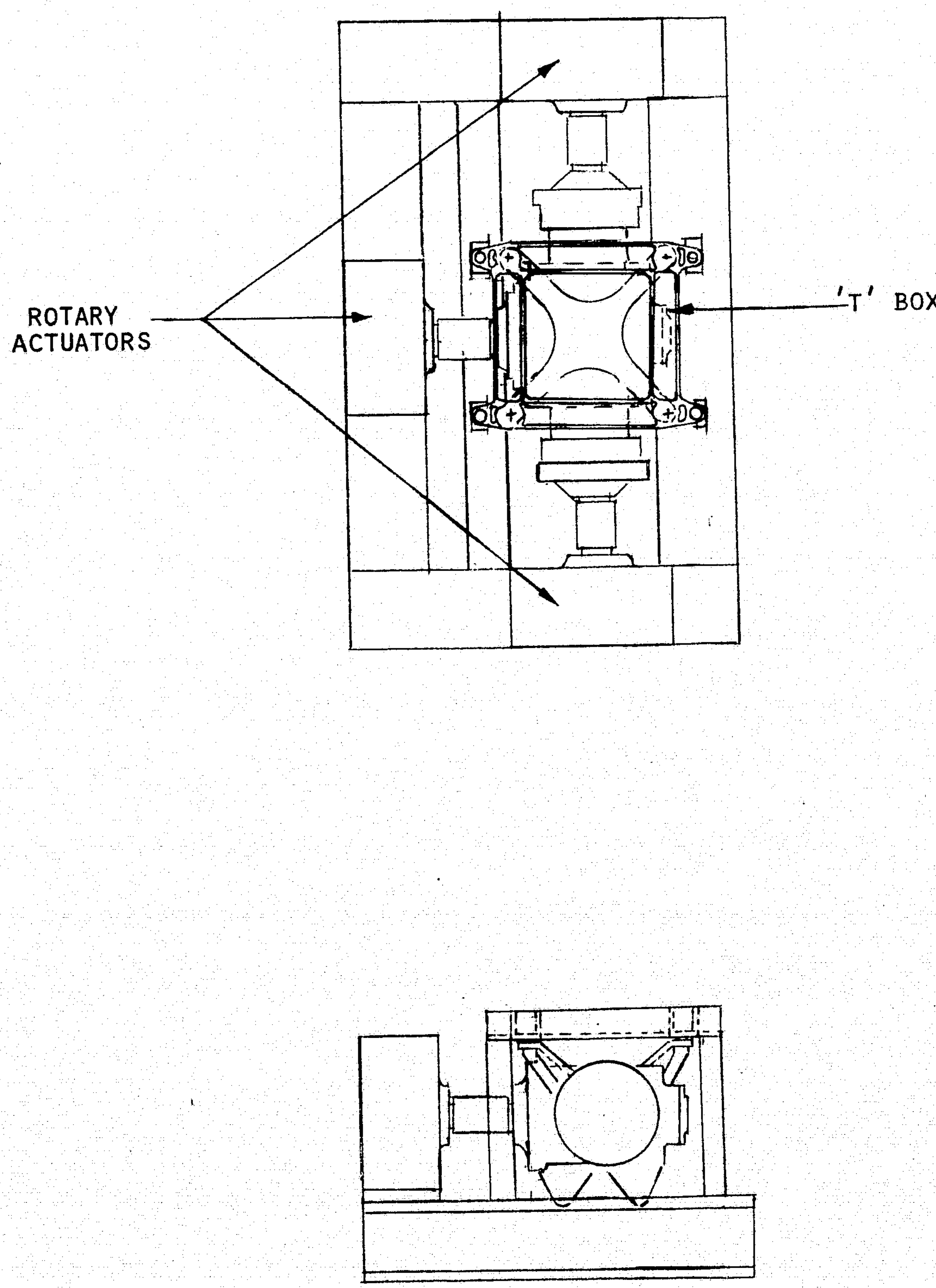


FIGURE 1.14-4

GREEN RUN TEST STAND

"T" - BOX

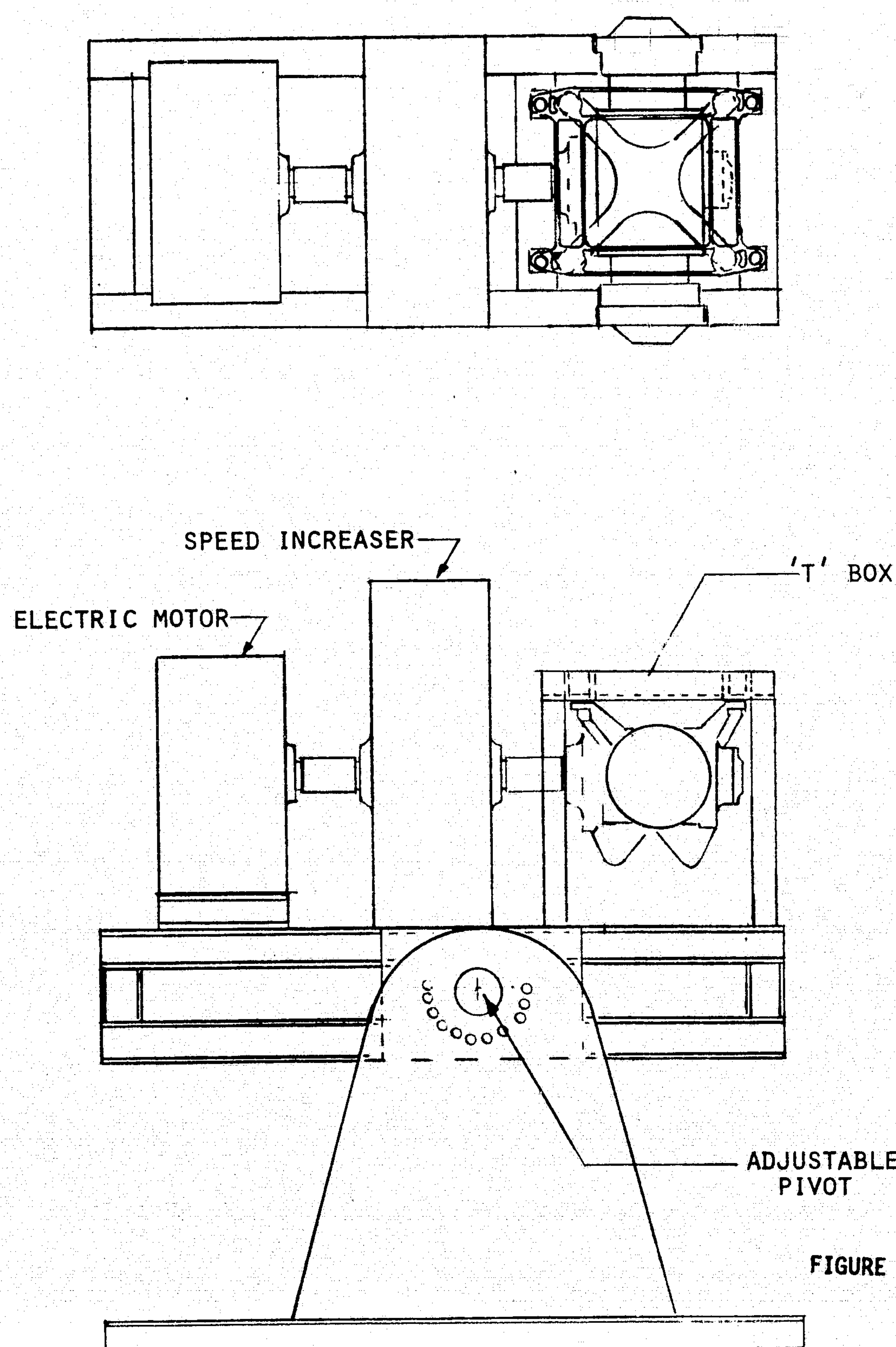
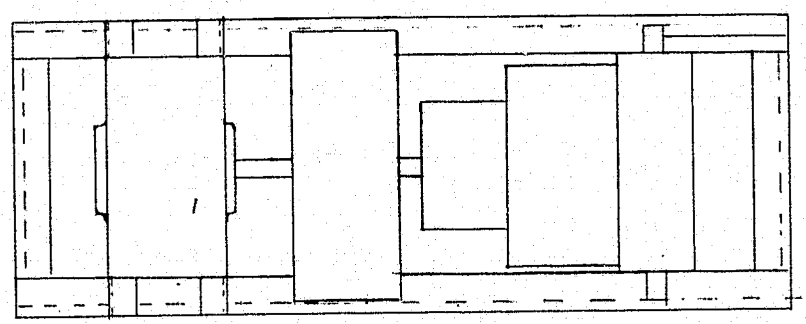


FIGURE 1.14-5

GREEN RUN TEST STAND

DROP BOX



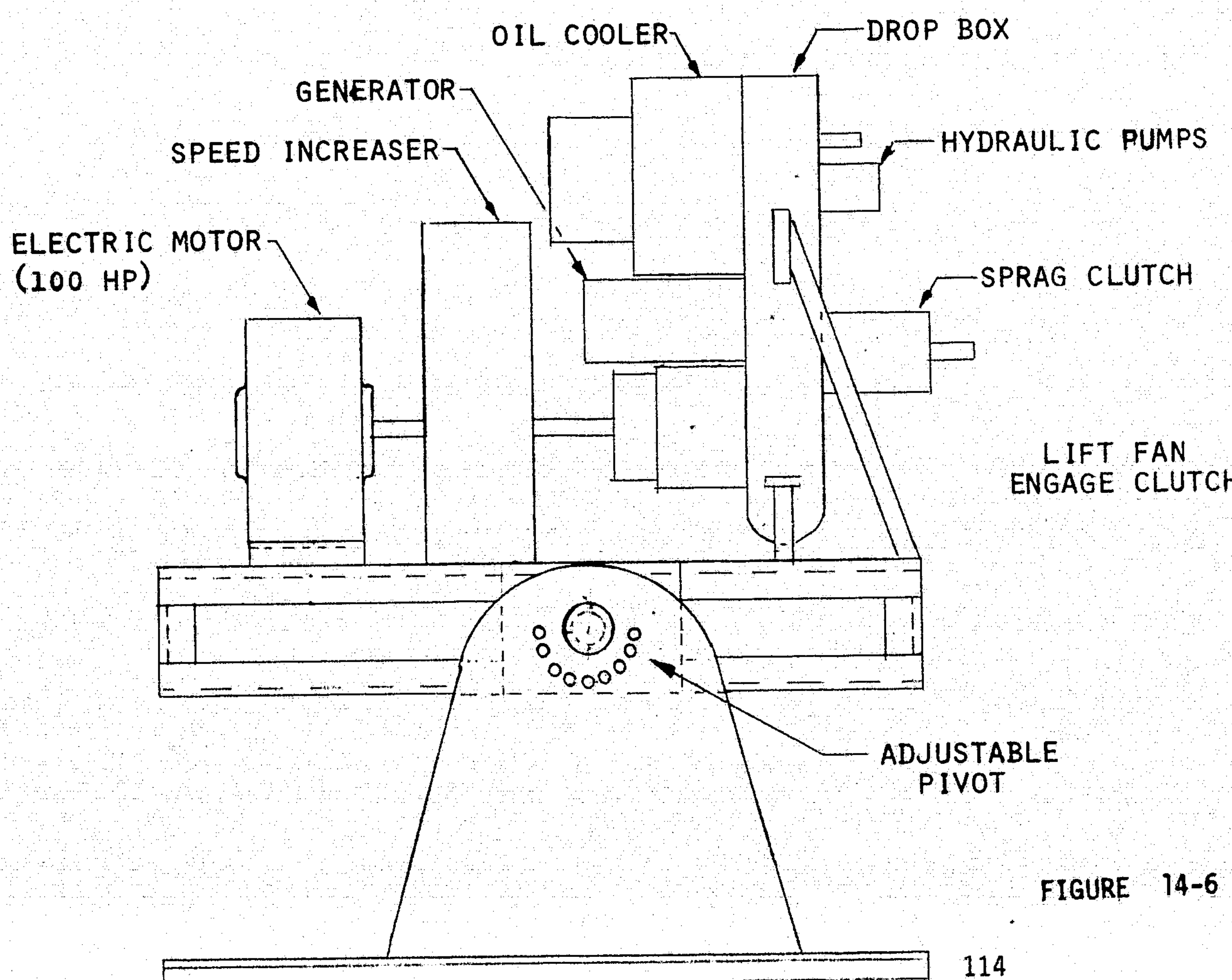
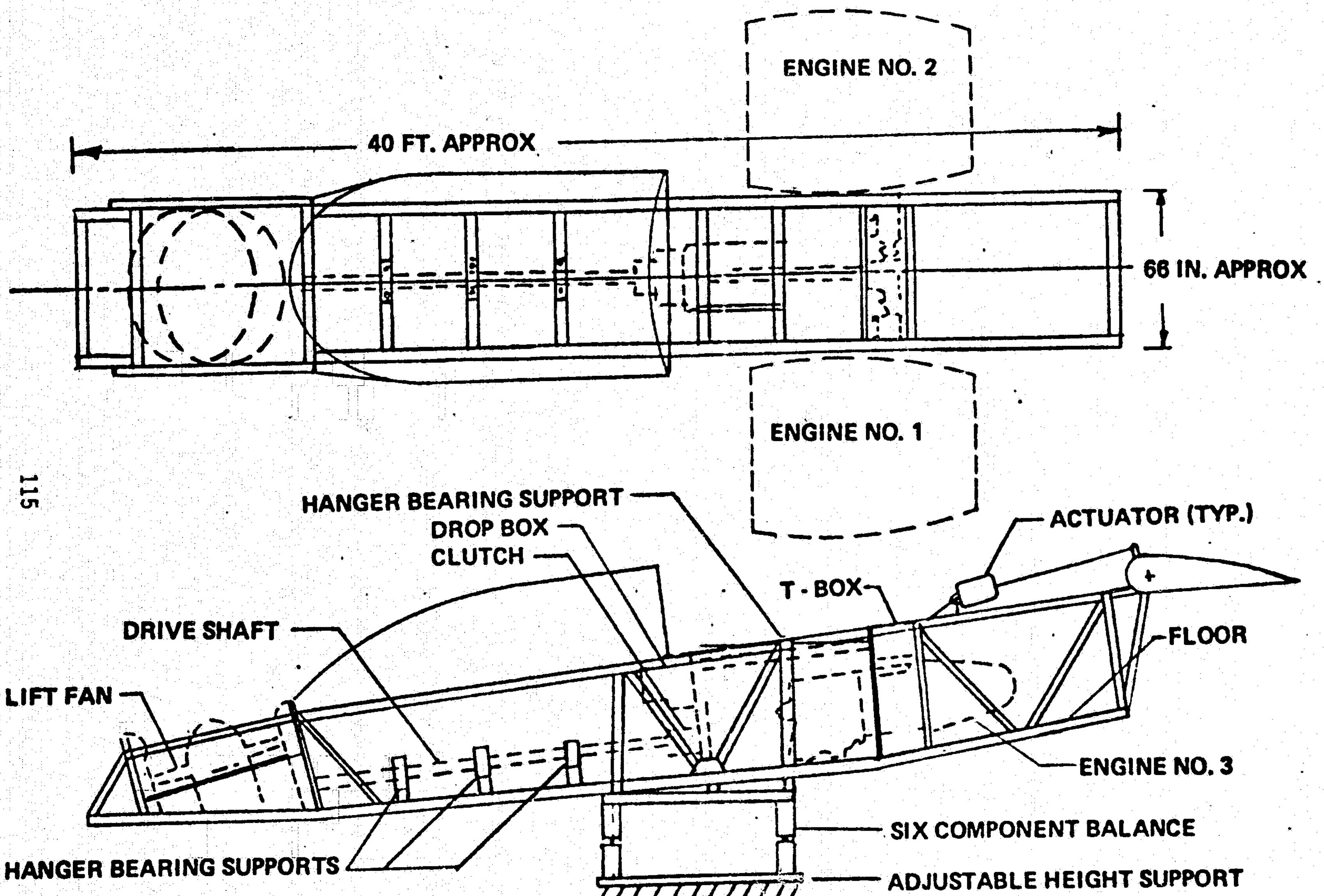


FIGURE 14-6

IRON BIRD TEST RIG



Minimum torque will be added to the first 12 cycles at 10 minutes per cycle. Fans will be in flat pitch and balanced thrust condition.

Following each 2 hour cycle the loads will be repeated. At the completion of 52 hours a teardown inspection of all gear boxes will be made. After examination a limited clearance for aircraft ground run up may be made. At 102 hours of test time a further component inspection will be made. Flight clearance will be based upon satisfactory condition upon teardown inspection at the appropriate point in the test sequence.

FIGURE 1.14-8 TEST TABLE

	CONDITION	TEST HOURS BEFORE FLIGHT	TOTAL TEST HOURS
1	Hover	50	100
	Climb	20	55
2	OEI 3-Hover	15	15
2	OEI 2-Hover	7.5	15
2	OEI 1-Hover	7.5	15
	Min. Torque	2	2
		102	202

1 Perform one reengagement clutch cycle per hour for a total of 100 cycles. Apply maximum pitch and roll control loads each 1/2 hour for a total of 200 cycles.

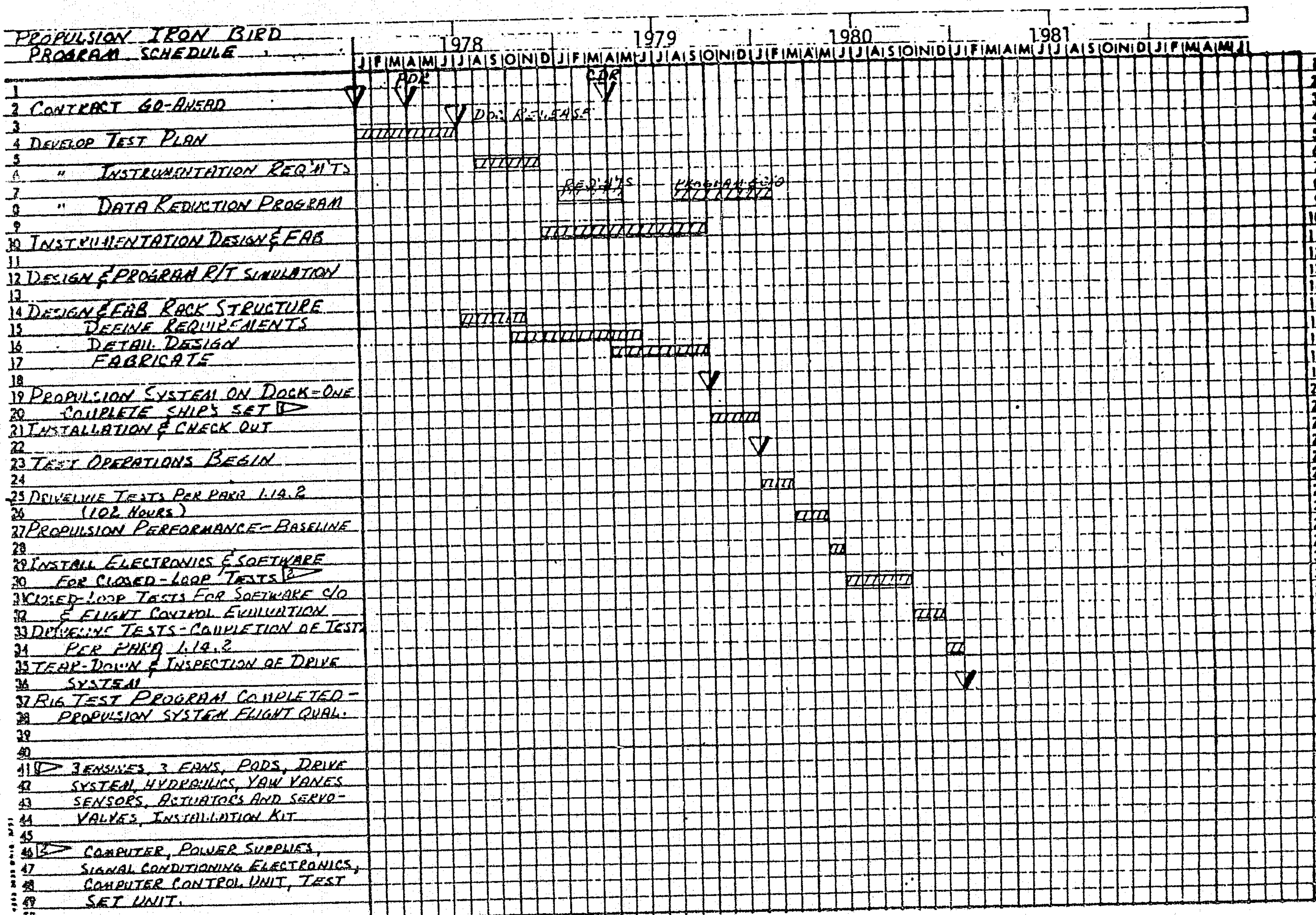
2 Overrun mode for respective engine.

After completion of the first phase of the drive system tests have been completed, the use of the propulsion iron-bird rig will be devoted to propulsion performance testing and flight controls testing for a period of seven months as shown by the schedule in Figure 1.14.9. The test articles, engines, fans and drive system - will accumulate an estimated 200 hours of operation during the propulsion performance and flight-controls testing. A final 100 hours of testing will be conducted to complete the drive system test program.

The baseline rig test program outlined above will provide 400 hours of propulsion system operation, of which 200 hours is devoted specifically to drive system tests. The schedule is predicted on one-shift operation of the test facility. An optional alternative program would provide an additional 200 hours of test time by operating the rig on the second shift during the period devoted to flight-controls testing. This second-shift testing would be devoted entirely to endurance testing of the propulsion drive system. It will be noted that the 600-hour program would provide more test time on the propulsion drive system than the projected 500 hour flight test program, thus generating a high level of confidence in the integrity of the system.

1.14.4 Integrated System Testing (Iron Bird)

A comprehensive system-level test program conducted with a full-scale propulsion test rig (iron-bird) is considered essential to the timely development of a safe, reliable flight vehicle. Experience has shown that low risk, step-by-step system tests, with complexity increasing at each step, is a cost-effective way to develop flight systems that consistently meet their cost, schedule, and performance objectives. The propulsion iron bird was introduced in the previous section as a method for testing propulsion drive systems that has become accepted in the helicopter industry. The iron bird concept is developed further in this section as the preferred method of performing integrated engineering, development, and demonstration tests on those subsystems that are active in powered-lift flight.



PROPULSION IRON BIRD TEST PROGRAM SCHEDULE
FIGURE 1.14-9

1.14.4.1 Propulsion Rig Test Goals. Specific goals to be pursued during the iron bird test program are listed below. It will be noted that this list includes several items that would require extensive and hoc tests if the iron bird were not available.

A) Establish the fit and compatibility of the total hardware system including engines, fans, gear boxes, drive line, mounts, lubrication system, cooling system, yaw vanes, variable area nozzles, flight control actuation, propulsion control system.

B) Demonstrate hardware operability and integrity.

C) Measure installed propulsion system performance, steady-state and transient with two and three engines in operation.

D) Perform software checkout under operational conditions.

E) Demonstrate and evaluate control law design and implementation. Identify and correct any instabilities or limit cycles caused by propulsion - flight control interactions.

F) Evaluate total system response to engine failure.
   1. Engine-out VTOL performance.
   2. Impact of engine failure during VTOL operation.

G) Crew familiarization and development of flight test procedures, including emergency procedures.

H) Develop and evaluate flight-test instrumentation, recording, and data reduction procedures.

I) Identify, support, and demonstrate any design modifications required for the airplane (including the drive, flight control and propulsion systems) to achieve its design goals.

1.14.4.2 Propulsion Iron Bird Facility Requirements. The test facility should simulate the operating environment of the flight propulsion system as closely as is feasible. Instrumentation, recording equipment, and other apparatus required to achieve the test goals listed above should be incorporated in the facility. These plant services (electrical power, utilities, security and fire protection) fuel supply, ground support equipment for the engines as well as the test rig itself. The principal rig facility requirements are as follows:

A) General

The test rig shall provide mounting points for all propulsion, drive system, and control components such that the geometry and compliance of the airframe are simulated as closely as possible.

B) Instrumentation

Instrumentation for measuring for the sum of the forces and moments executed on the rig by the propulsion system shall be provided. Measurements will be made in real time and shall be in a format suitable for presentation both to a recorder and to a real-time computer simulation of airframe dynamics. All relevant propulsion variables will be measured and displayed and/or recorded as appropriate.

C) Safety

The rig assembly shall be safe to maintain and operate.

D) Physical and environmental protection for electronics, sensors, instrumentation, recorders, cables, and operating crew shall be provided.

E) Access

Physical access to all components, both test articles and instrumentation, shall be possible without significant effort beyond removal of protective covers.

F) Facilities

A control room of sufficient size for convenient conducting of test operations shall be provided along with crew interfaces, displays, monitoring equipment, recorders, desks, etc., required to achieve the test goals. The control room shall provide a safe and comfortable working environment for the test crew. Air conditioning shall be provided to ensure reliable operation of the electronics housed in the control room. The air conditioning system should keep the control room temperature to 70°F or lower under a heat load equivalent to 10 personnel and 5000 watts dissipation by electronic equipment.

1.14.4.3 Integration Plan. The integration plan is summarized by Figures 1.14-9 and 1.14-10 which show preliminary schedules for development and documentation of detailed requirements, the procurement or fabrication of hardware and software, the development of real-time simulations to close control loops, and the actual test operations. The program is time-phased to distribute the work as uniformly as possible over the period of interest to ease peak manpower requirements and to allow maximum continuity of personnel. The progression of activities is generally from the less complex to the more complex, consistent with the low-risk step-by-step approach.

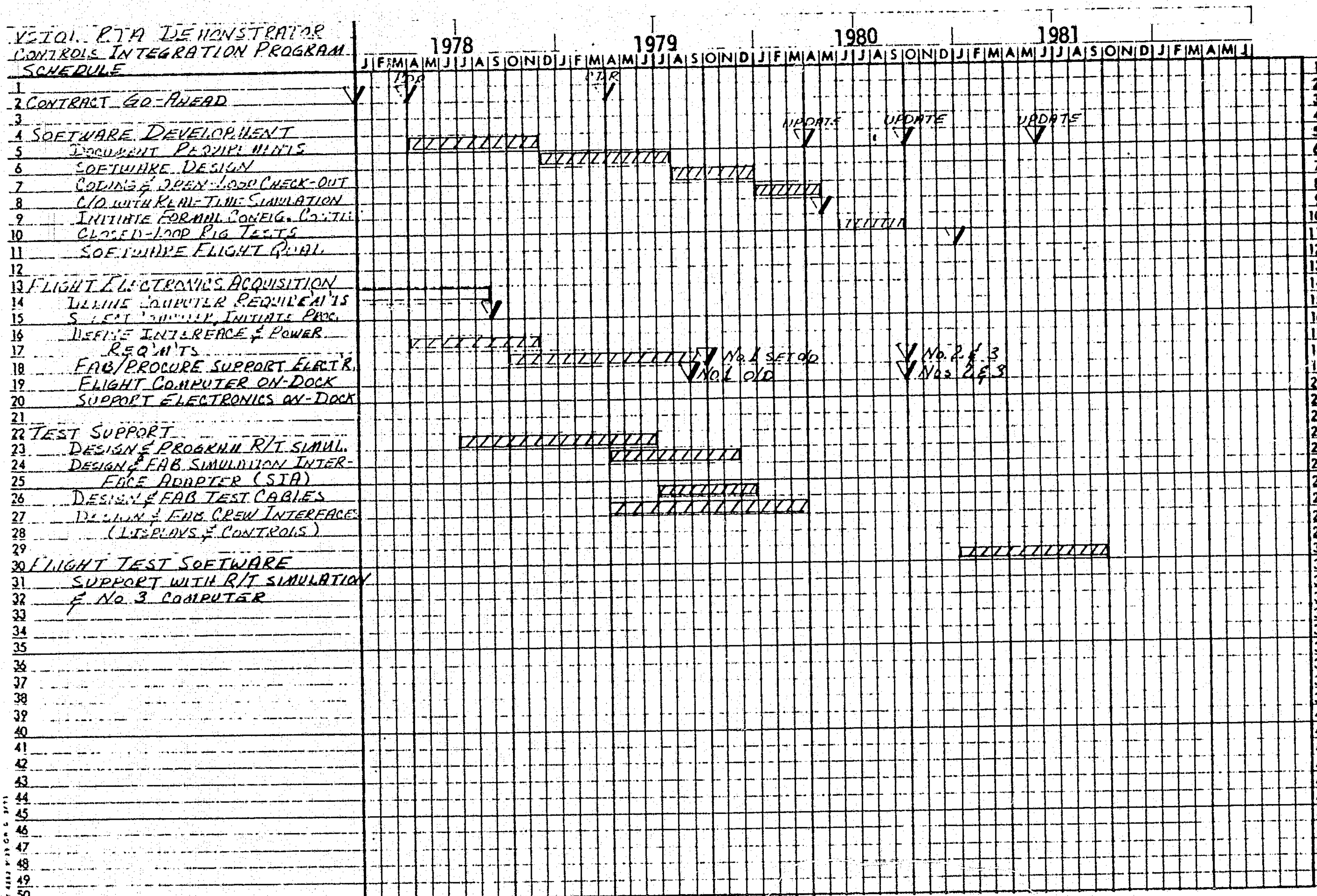
A four test series is planned.

A) Functional Testing

The first test period (two months) is devoted to functional testing of the drive train. A detailed discussion of the driveline test is given in Section 1.14.3. This activity is scheduled early in the program to allow maximum lead time for any hardware revisions indicated by test results. This will also allow time for check-out of instrumentation, preparation of flight control hardware and software, and crew training required for later, more complex tests. The system will be operated approximately 102 hours during this first test series.

B) Propulsion System Performance

The second test period (also two months) is used to evaluate installed propulsion performance as described in Section 1.14.4.4. The data recorded during this period will be used to refine the flight control laws prior to the flight control/software check-out phase. It will also provide a performance baseline for evaluating flight test results. Approximately 50 hours of test running will be required to obtain the required propulsion performance data. Allowing a 50% margin for start-up, shut-down, setting up test conditions, and re-running bad points, about 75 hours of operating time will be accumulated during the propulsion evaluation test phase. The propulsion performance test events are listed in Table 1.14-1.



VSTOL RTA DEMONSTRATOR CONTROLS INTEGRATION PROGRAM SCHEDULE

FIGURE 1.14-10

TABLE 1.14.1 PROPULSION SYSTEM PERFORMANCE TEST EVENTS

I. Steady-State Performance

Operate at matrix of 3 fan speeds and 5 fan blade pitches (15 power settings*). Stabilize for three (3) minutes at each power setting before taking data. Sweep pods and yaw vanes through limits:

- o 10 pod settings with yaw vanes centered
- o 5 yaw vane settings each at 3 pod settings

Allow one minute to stabilize at each point, 30 seconds to record data.

Run the above sequence on three engines, 2 lift-cruise (L/C) engines, and 1 L/C engine + lift engine.

Estimated test time for steady-state tests - 33 hours.

II. Propulsion Transient Performance

A. Step Response: (L/C Pods Vertical)

1. At each power setting in I, command two (2) steps to higher pitch and two (2) steps to flatter pitch. Magnitude of steps TBD. Stabilize for fifteen (15) seconds, step back to original setting, stabilize thirty (30) seconds before running next point.

   Run above sequence on 3 engines, 2 L/C engines, and 1 L/C engine + lift engine (180 points; 3 hours)

2. At each power setting in I, make two (2) steps in yaw vane setting from E to each side of netural. Run with three engines only (60 points; 1 hour).

A one-month period is scheduled after propulsion evaluation to install the flight control electronics and software for the closed loop control tests which follow.

C) Closed Loop Control

Four months are allowed for closed-loop evaluation of flight control laws and check-out of controller software in an operational environment. The test rig will be fitted with strain gauges on the struts to measure forces and moments during these tests. A real-time simulation of the aircraft will integrate the measured forces and moments to provide on-line indication of aircraft status, which will be displayed to the crew. The crew will control the system via a boiler-plate version of the cockpit controls as indicated by the diagram in Figure 1.14-11. The flight control/software evaluation test events are listed in Table 1.14-2.

The propulsion rig will accumulate approximately 125 hours of operating time during the flight-controls test.

D) Drive System Development

An additional two months is scheduled for drive system testing to complete the work described in Section 1.14.3.

A final month is allowed for tear-down and detailed inspection of the drive system. This inspection will be completed prior to the first flight of the No. 1 aircraft. The propulsion system and its control system will be qualified for powered-flight upon completion of this inspection.

COCKPIT CONTROLS SCHEMATIC

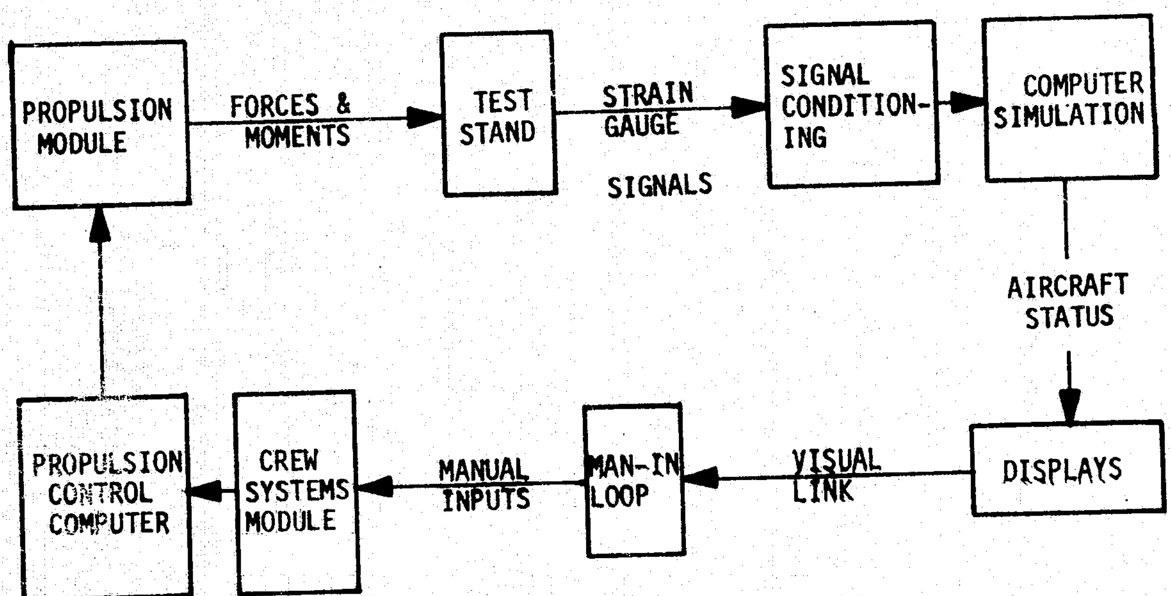


FIGURE 1.14-11

B) Sinusoids

1. At each power setting in I, inject sinusoidal pitch command of amplitude and frequency TBD (8 points at each power setting). Continue one (1) minute for each test point. Run with 3 engines, L/C pods vertical. (120 points; 2 hours).

2. At each test condition in 1 above inject sinusoidal yaw vane command of amplitude and frequency TBD. Continue one (1) minute at each test point. Run with three engines, L/C pods vertical (120 points, 2 hours).

3. At each test condition in one above, inject sinusoidal roll commands of amplitude and frequency TBD. Continue for one (1) minute. Run with 3 engines, L/C pods vertical and at 45^{o}. (240 points, 4 hours).

III. Failure Response

Propulsion system response to engine failure shall be estimated in two ways: (a) by stepping the gas generator speed command to idle to simulate a failed engine, and (b) by shutting off fuel to one of the engines. Use of the fuel shut-off shall be minimized to minimize exposure to the extreme thermal transients associated with engine shut-down from high power settings.

All test points shall be initialized with three engines operating L/C pods set at 90^{o} (vertical). Stabilize on point for three (3) minutes before initiating the failure. After simulated failure, the power setting on the reamining engines shall be reduced to the pre-failure level after to (10) seconds or after transients decay, whichever is the sooner. The failure shall be assumed to occur when the clutch of the affected engine disengages. (30 points, 3.5 hours).

A. Gas Generator Step Command

1. At each power setting in I, step the gas generator command of one L/C engine to idle.
2. At each power setting in I, step the gas generator command of the lift engine to idle.

B. <u>Gas Generator Shut-Down</u>

Failure simulation by fuel cut-off shall be conducted at power settings corresponding to hover at the maximum and minimum hover weights. The two failure series shall be conducted once with a L/C engine shut-down and once with the lift engine shut-down for a total of four (4) events. Approximately 50 hours operating time to evaluate propulsion performance.

TABLE 1.14-2 SOFTWARE CHECK-OUT/FLIGHT CONTROL TEST EVENTS

1. Propulsion transients without flight simulation.
2. Lift-off, hover maneuvers, land.
3. Lift-off, accelerate* to wing-borne speed.
4. Wing-borne maneuvers.
5. Decelerate*, hover maneuvers, re-accelerate to wing-borne speed
6. Decelerate, hover maneuvers, land.
7. STOL Take-off, accel to cruise speed.
8. Simulate engine failure during wing borne flight, decelerate and land in VTOL mode.
   a. Failure of lift engine
   b. Failure of L/C engine
9. Simulate engine failure during VTOL flight.

* Several different types of acceleration and deceleration maneuvers will be investigated. For example, constant glide slope decelerations, constant altitude, or constant sink rate decleerations may be explored. Similar possibilities for accelerating transition exist and warrant examination.

1.14.5 Emergency Escape System

A minimum amount of testing will be necessary due to the previous qualification tatus of the Stencil S111S-3 seat. Testing will be conducted to ensure safe integration into the RTD airplane. The tests will be to simulate representative VTOL operation in accordance with MIL-E-9426A.

Two track tests will be conducted using a simulated R.T.A. vehicle. The first first will be at 120 KEAS and the second at 300 KEAS. The test conditions, objectives, equipment and procedure are summarized in Table 1.14-3.

TABLE 1.14-3 ESCAPE SYSTEM TEST SUMMARY

TEST CONDITIONS	OBJECTIVE	EQUIPMENT	PROCEDURE
Preoperative	Validate initiation control operation.	Ejection seat mounted in aircraft	Simulate initiation with catapult dct
Track Test 120 KEAS	Validate low speed escape system performance	Track test vehicle	Measure velocity and accelerations Photograph trajectory
Track Test 300 KEAS	Validate high speed escape system performance	Track test vehicle	Measure velocity and accelerations. Photograph trajectory

1.14.6 Flight Testing

A brief flight test will be conducted prior to delivery. The intent is to clear the airplane as being airworthy and not to pursue research goals. The following test events will be performed:

- o Structural demonstration
- o Performance and handling
- o Flutter clearance
- o Full functional check-out in flight

Two flight test programs are offered. The baseline (100 hour) program would exercise all flight modes, including hover and transition to and from wing-borne flight. The low-cost alternative program (30 hours) would demonstrate only wing-borne (CTOL) flight.

1.15 USE OF EXISTING COMPONENT SUMMARY

Wherever possible, existing components, or off-the-shelf hardware has been identified for use on the RTA. A preliminary summary is listed below of the items identified to date. Notation is provided as to "useable as is", or "modified as described". Modification descriptions of major items appear in section 1.17 of this report.

TABLE 1.15-1

Item	Useable As-Is Or As Procured Off/Shelf	Modified or Adapted As Described
Airframe		
T-39 Wing Structure		X
T-39 Fuselage Structure		X
T-39 Cab Structure		X
T-39 Horizontal Tail		X
T-39 Rudder		X
A-4 Main Landing Gear Struts		X
A-4 Nose Landing Gear Strut		X
A-4 Wheels/Brakes/Tires	X	
T-39 VFR Instrumentation/ Communication	X	
T-39 L. E. Slats		X
T-39 Flaps		X
T-39 Ailerons		X
T-39 ECS		X
Electrical Generators and Back-up	X	
Control System Actuators	X	
Hydraulic Pumps, etc.	X	
Oxygen Syst.	X	

TABLE 1.15-1 (Continued)

Item	Useable As-Is Or As Procured Off/Shelf	Modified or Adapted As Described
Propulsion		
T-39 Wing Tanks		X
Engine Starters	X	
T-Box, D-Box Lubrication/Cooling (HLH)		X
Drive Shaft Supports (HLH)		X
L/C Fan Pod Rotation Actuator (Harmonic Drive)		X
Sprag Clutch (HLH)	X	
Flight Controls		
FCS Computer (HLH or Equivalent)		
FCS Sensors	X	

1.16 EXPERIMENTAL SHOP USE

Costing will be based upon use of experimental shop fabrication, assembly and test. This assumption was used in the costing presented for the demonstrators defined in the 1975 definition study under Contract NAS2-6563. The procedures for engineering, shop, inspection and related functions will follow very closely those used in the Buffalo modification for the Augmentor Wing Research Aircraft and the QSRA Program. The former was accomplished by Boeing for NASA Ames and the latter is now in progress for NASA Ames. Generally, the implications are:

- o Engineers, final assembly craftsmen, flight line mechanics and inspectors having exceptional versatility will be assigned. A team of this composition naturally eases coordination in the conceptual, assembly and operating phases.

- o When practical, drawings for the shop will have details described on layout or assembly drawings. In those cases where layout drawings can suffice for assembly guidance, no new drawing will be made.

- o All drawings will be controlled and change notices will be made for control. As appropriate, all configuration sensitive change notices will be incorporated onto the drawings. Other notices need not be incorporated.

- o Manufacturing engineering specialists will review design layout work on the designer's board and a cooperative derivation of a "least cost" approach will ensue. This entails establishing desirable arrangement, fabrication, process and assembly techniques that match a one-or-two-of-a kind production and developmental shop tooling and fabrication facilities.

- o Shop progress will be expedited by authorized project engineer mark-up of shop drawings on the job. Such change or clarification will be dated and signed on the shop drawing by the engineer and at a convenient time will be covered by the appropriate change notice. Adequate paper records will be maintained for quality control and government inspection acceptance. Where necessary, the engineer responsible for the design will engage in inspection "buy-off" where drawing clarity or special action has been taken to produce the part or assembly.

- Drawing preparation and notes thereon will be done with the idea in mind that only one or two ship sets are to be made and that appropriate engineering assistance will be available to assist. The approach will reduce normal drawing preparation time.
- The project engineering group will be co-located with final assembly so that the designers fulfill the liaison engineering function. Similarly, the designer will serve the liaison function for fabrication at the selected fabrication shops.
- Developmental tooling will be used wherever possible.
- The modification design will avoid requirements for tooling wherever practical. For example, reskinning of a fin or stabilizer will be accomplished in a sequence that uses the basic structure as its own jig or fixture.
- Developmental shop assembly craftsmen will be used.
- Developmental shops will do the majority of the required parts fabrication.
- Use of the main production fabrication shops will only be made in discrete instances requiring their special capability for welding or honeycomb structure.
- Functional tests for acceptance will be delineated and attended by the design project engineers.
- As appropriate, the final assembly craftsmen will service the aircraft during functional, ground and contractor flight test.
- It is expected that qualified engineering, inspection, and flight line mechanic personnel will spend 6 months or more in residence at the customer's operating facility to adequately train the government crew that will take over. This provision was made for the Buffalo Augmentor Wing Research Aircraft Program and proved quite successful. Here again the engineering definition of operating, maintenance and drawing descriptions can be tailored to meet this expected turnover. Personalized participation and follow-through by the actual designers, flight line crew and inspectors reduces the extent of required formal definition.

- Shop planners and schedulers will be selected from our developmental organization and will be personnel that have been through the mill on previous Boeing experimental aircraft developments. This approach is used on the QSRA program. Many Buffalo Augmentor Wing Research Airplane personnel are assigned to that program.

1.17 PRELIMINARY MODIFICATION WORK STATEMENT

The following paragraphs explain the approach and extent of modification necessary in the restructure of the T-39A.

1.17.1 Airframe

A) Landing Gear (see Figure 1.5-1). Use A-4 nose gear, fork, wheel, axle, tire, steering cylinder and shock strut with metering pin modification mounted in reinforced existing T-39 nose support beams. Aft retraction will be accomplished by knuckled drag bracing struts which straddle the front fan drive shaft when retracted. A modified length A-4 gear telescoping link will be used to reduce gear length for retraction. This provision minimizes airframe modification.

The main landing gear consists of the A-4 wheel, tire, brake and shock strut assembly with a modified metering pin. The shock strut is installed in a new design "support jacket" to provide the necessary landing gear length. The A-4 gear swivelling mechanism is eliminated since inboard retraction is planned and the wheels will be fully enclosed within the T-39 wing/body contour. The support jacket is hinged at an upper end fitting, located between the swept rear wing spar and an added transverse beam extending through the body to the opposite rear wing spar. A trailing edge wing/body fillet is added to fair the necessary auxiliary beam depth. The lower end of the support jacket is supported at the apex of a folding "V" brace which transfers gear side, drag, and torque load to the body structure.

Use will be made of T-39 or A-4 gear up/down locks, door latches, retraction/extension actuators, gear position indicators, gear-up warning system, and door actuation linkages.

Nose gear doors will be provided. As shown in the inboard profile (Section 1.5), a blister contour in the door covers the nose wheel.

Main gear strut doors will be provided and attached to the struts. Clamshell wheel doors are provided on the body.

B) Subsystems and Conventional Controls

B-1) The electrical system will generally retain ship's wiring with special cable provisions for test instrumentation. Two new 20KVA generators will be installed to accommodate the additional load. They will be mounted on the drive system drop-gear box. The old generator installation will be removed.

B-2) Hydraulic System - This system is completely revised. It will consist of dual main 3,000 psi systems for operating landing gear, nose wheel steering, ailerons, rudder, horizontal slab tail, nose fan doors, nose fan vanes, lift/cruise pod pitch angle, lift/cruise pod yaw vanes, L/C fan nozzle and nose fan Beta control. Dual pumps are driven from the drive system drop-gear box. Two new fluid reservoirs will be provided along with necessary control and shut-off valves, relief valves, filters, indicators, regulators, etc. that fulfill the new system requirements for the added elements to be powered.

B-3) Environmental Control System - Existing cooling packs are candidates for adaptation. Cabin pressurization is not to be provided.

B-4) Fire, Heat Detection, and Fire Extinguishing Systems - New systems will be provided to accommodate engines, gear boxes, clutch, and associated cooling systems.

B-5) Anti-icing/Deicing Systems - None provided.

B-6) Oxygen System - Modify existing provisions to accommodate pilot and co-pilot only. Delete other provisions.

B-7) Slats - Retain existing system, but tailor outboard segment to match wing cut-off.

B-8) Ailerons - Retain existing ailerons but tailor outboard region to match wing cut-off. Note that wing cut-off occurs outboard of outboard aileron hinge, thus the existing aileron support and actuation is retained. Aileron actuation will be hydraulic through a modified YC-14 central aileron actuator with cable runs to the ailerons.

B-9) Flaps - The existing flap system will be retained. A cut-out and a local auxiliary hinged surface will be provided on the flap to accommodate the motion of the lift/cruise fan pod in its extreme angular position for vertical flight operation.

B-10) Rudder - The existing rudder will be used but the fin is new. New hydraulic rudder actuation is provided.

B-11) Horizontal Slab Tail - The existing horizontal stabilizer and elevator have been modified to serve as a flying slab tail. Hydraulic actuation is provided for control and electrical screwjack actuation for trim. This is a completely new actuation system.

C) Cockpit

Engine instruments will be replaced. Additional position indicating instrumentation will be provided for control surface and thrust vectoring controls (rudder, slab tail, nose fan vector, L/C fan nozzle, L/C fan vector). The wheel controls will be replaced with dual stick controls for the pilot and co-pilot. Longitudinal and lateral trim controls will be provided on each stick. Throttle control will be modified to provide necessary detents for mode selection. A new control for manual setting of L/C pod angle is provided. Provisions will be made for SAS and its mode selection controls. The overhead cabin structure will be modified to provide a clear through path for ejection through a frangible canopy. The side door and its frame are modified to allow the addition of a canted overhead frame. This frame is to provide body bending strength that was previously supplied by the cockpit overhead centerline structure.

D) Ejection Seats

Both pilot and co-pilot seats will be removed to be replaced by Stencil SIIIS-3F16 (Harrier type) ejection seats. The bulkhead aft of the cabin section will be modified and strengthened to carry the new escape loads and accommodate the 17 degrees required for ejection rail installation.

E) Wings

Wing structural modifications are confined to four areas. The first is reducing the semi-span by 22 inches and providing a new tip fairing. The second is incorporation of the transverse landing gear beam and attachment provisions for the gear and its actuation. The third is recompartmenting of the internal fuel cell system to provide three tanks. This entails providing feed-through plumbing through the wing centerline rib and two tank bulkheads each side of centerline to create a center fuel tank. The addition of left and right tank bulkheads outboard of these center tank bulkheads then provide left and right outboard tanks. The remainder of the existing wing tankage is retained for use in providing ferry fuel capacity. Minor modifications are required to accommodate fuel feed lines, venting, and boost pump installations. The fourth is a fairing to accommodate the landing gear gransverse beam.

F) Fuselage

F-1) A new nose section will be required to accommodate the lift fan installation, flight test instrumentation, an instrumentation nose boom, and ballast provisions.

F-2) In the cockpit region local beef-up is required to accomodate the loads resulting from the new nose section, installation of the nose gear, and cut-away of overhead structure for ejection egress. Also, modification is necessary to accommodate ejection system installation and loads. Provisions for forward fan drive shaft, and intermediate bearing support and protective fairing are required. The fairing should provide containment features for the shafting to protect the crew.

F-3) The forward cabin must be adapted to installation of flight test instrumentation racks, hydraulic reservoirs, drive shaft bearing supports and drive shaft protective fairing and containment.

F-4) Aft cabin and fuselage must accommodate the third engine installation including its air intake provisions, drop-gear box, shroud for cooling and fire zone delineation and firewall shroud over burner section and tail pipe as well as tail pipe. Engine mounting and drop-box mounting will be as shown in Figure 1.5-3 . An engine accessory access panel and its structured frame will be provided on the left hand side. It should be noted that fuselage pressurization will not be provided.

The T-Box installation and lift/cruise pod structural cross tie is above the engine. A notch cut-out in body structure is provided along with new longeron elements for body load carry-through around the notch. Additionally, a VEE-frame structure is provided at the top of the fuselage to bridge the notch and provide body bending structure and torsional strength. The VEE-frame is removable to permit T-Box access for service and/or removal. Capability is provided for installation of the entire left and right hand lift/cruise pod, cross shaft drive, T-Box and interconnecting lateral structure as a unit. Additionally, the arrangement permits removal and replacement of just the T-Box or a cross shaft drive member. The lateral box beam that interconnects the left and right pod assemblies is shown in Section B-B in Figure 1.5-3 .

An intermediate drive shaft bearing support is provided for the interconnect shaft running fore and aft between the T-Box and the drop-box.

Aft fuselage beef-up is made to provide additional bending and torsional strength and rigidity to accommodate increased loading from the T-tail. Further detail of this modification remains to be determined. This strengthening along with some fuselage frame strengthening is also provided locally to permit introducing the vertical fin front spar loads into the fuselage.

Provisions are made to install and remove the fuselage engine (3rd engine) through a framed aperture at the rear, lower, fuselage. A structural door covers the aperture but includes a hole for the third engine tailpipe. When in place, the door provides torsional strength for asymmetric fin loads and body bending.

G) Empennage

G-1) The horizontal tail is made up of the existing T-39 tail panels with a new center section for adapting the tail to the vertical fin. The center section also provides the anhedral which is desirable for flutter avoidance. Since a flying tail is to be used, the existing

T-39 elevator surfaces are locked in their zero deflection mode.

G-2) A new vertical fin is planned for accommodating the new loading imposed by the T-tail configuration. It's outline approximates that of the T-39 fin and it is planned to utilize the existing T-39 rudder assembly and its hinges.

H) Miscellaneous
None at this time.

1.17.2 Propulsion System

A) The lift/cruise fan pod structure, structural cross beam, T-Box, cross shaft, clutch, pod thrust vectoring drive, electrical service, hydraulic service, control routing, and fuel feed installations are all new construction and are illustrated in Figure 1.17-1. The installation of this and the tie to the fuselage was briefly described in section 1 (F).

B) The drive shaft from the T-Box forward is installed with one intermediate support.

C) Installation of the drop-box has previously been described. The output from the drop box is through a disconnect clutch which is supported from a pad on the drop-box case.

D) The drive shaft forward to the nose fan has three intermediate support bearings that are supported from the deck structure. The nose fan installation concept is shown in Figure 1.5-2 . Details as to tying the fan case to the nose structure are in work in coordination with the Hamilton-Standard designers. Philosophically, the fan will be supported so that its case structure is isolated from airframe loading.

E) Nose fan entrance doors, exit doors and thrust vectoring concepts are shown in Figure 1.5-2 . These are all new structural items.

F) Installation concept for the lubrication and cooling systems for the T-Box, drop-box and clutch are shown in Section 2 of this report.

G) Thrust vectoring vanes are installed in the lift/cruise fan efflux. These are integrated with the support structure for the variable area fan nozzle.

LIFT FAN ROTATION MECHANISM

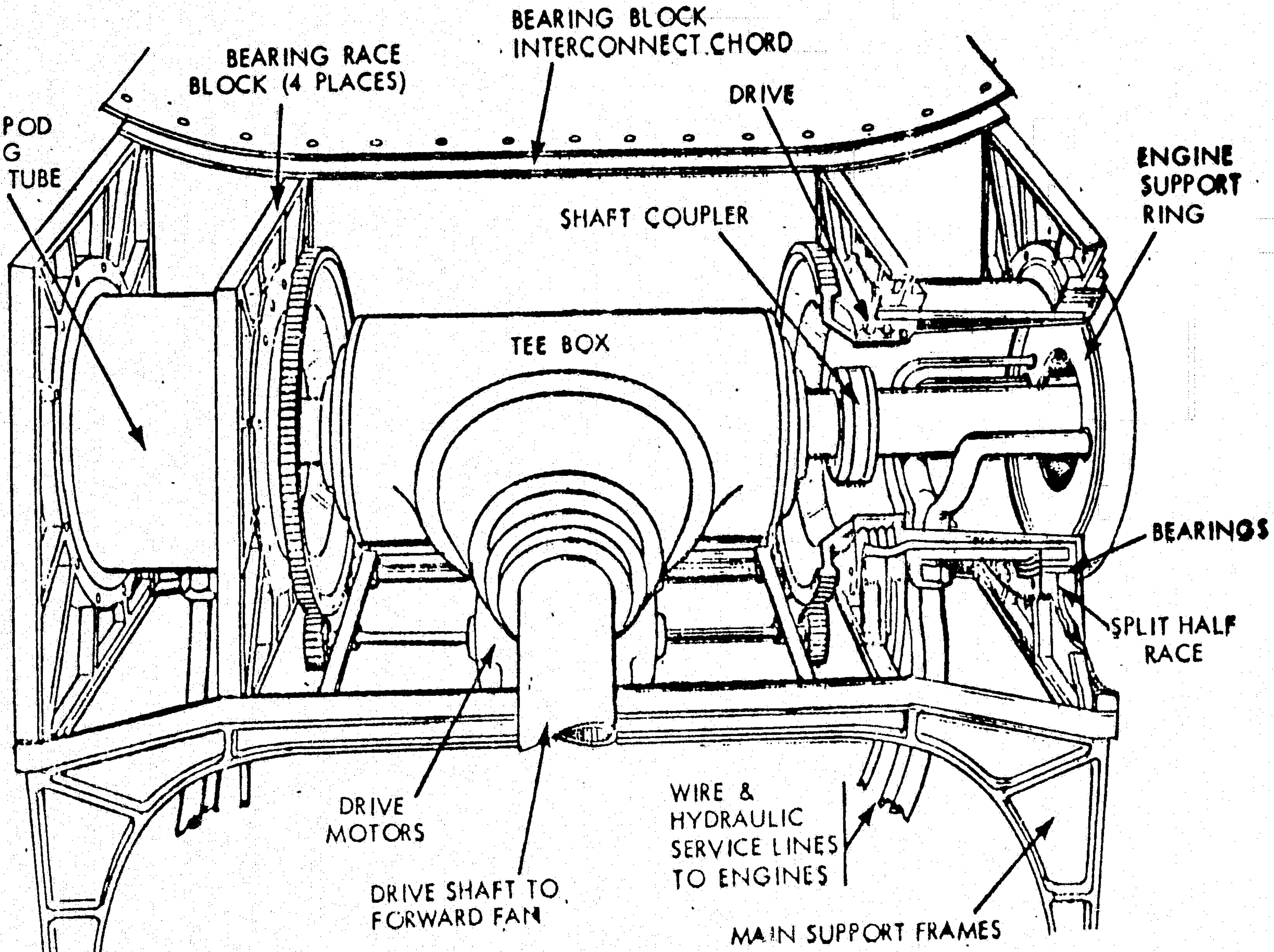


FIGURE 1.17-1

H) Miscellaneous

H-1) The fuel system will be modified to provide three integral wing tank compartments, their boost pumps, interconnects, transfer plumbing, controls and vents. Installation concept will not be shown. A schematic and description of the fuel system is in work and will be provided at a later date.

H-2) Fire zone compartmentation and protection are to be provided. Their description is beyond the scope of this study.

1.17.3 Control System

Roll, pitch and yaw axis flight control systems are shown schematically in Figures 1.10-2 thru 4. Fly-by-wire systems mixture with some mechanical system is shown.

The design concept is to provide a direct mechanical link between the pilot and Conventional Controls (rudder, stabilizer and aileron); a direct electrical link between the pilot and VTOL controls (fan blade angle, yaw vane angle, nacelle tilt angle); a direct electrical link to engine controls (fuel, fan nozzle); and a SAS with limited authority. The airplane systems revert to the direct link mode of operation in the event of a SAS Channel failure.

2.0 TASK 2- DESIGN DETAIL OF PROPULSION DRIVE SYSTEM

2.1 BASELINE SYSTEM DESCRIPTION

The baseline drive system is presented in Figure 2.1-1 (SK27132). It consists of shafting, gearboxes and clutches that provide a direct load path from any of the three engines to any of the three fans.

The clutch mounted on the forward face of the drop-box provides the capability of engaging the forward lift fan for V/STOL modes and disengaging it for conventional flight modes. This clutch is activated at pilot command.

There are three, one way (overrunning) clutches in the drive system. One each is located in the engine supplied gearbox and one is in the drop box at the input from the third engine. These clutches automatically disengage any time an engine operates at an RPM less than the system RPM. Thus, in the event of an engine failure, the remaining engines provide the power required to drive the fans. The failed engine is disengaged by the clutch.

Damage to a lift/cruise fan or malfunction of any one of the gearboxes (e.g., lubrication system failure) in the propulsion drive system could result in the need for disengaging the malfunctioning device from the drive system to permit conventional landing of the airplane. To provide this capability, each input gear in the T-box is equipped with a sliding spline coupling. An external hydraulic actuator will disengage one (to isolate a L/C fan or associated gearbox) or both (to isolate the Tee or drop box) of these couplings upon pilot command. The coupling is re-engaged on the ground with the system at rest.

2.1.1 T-Box (SK-27097)

The T-box, Figure 2.1-2 is a 90^0 spiral bevel gearbox with a ratio of 1:1. It is so designed and situated that it can be driving and/or a driven gearbox: i.e., engines #1 and #2 can drive into the box, to the drop box and thence to the forward lift fan. Under other conditions, the #3 engine can drive back through the drop box, to and through the T-box to the lift/cruise fans.

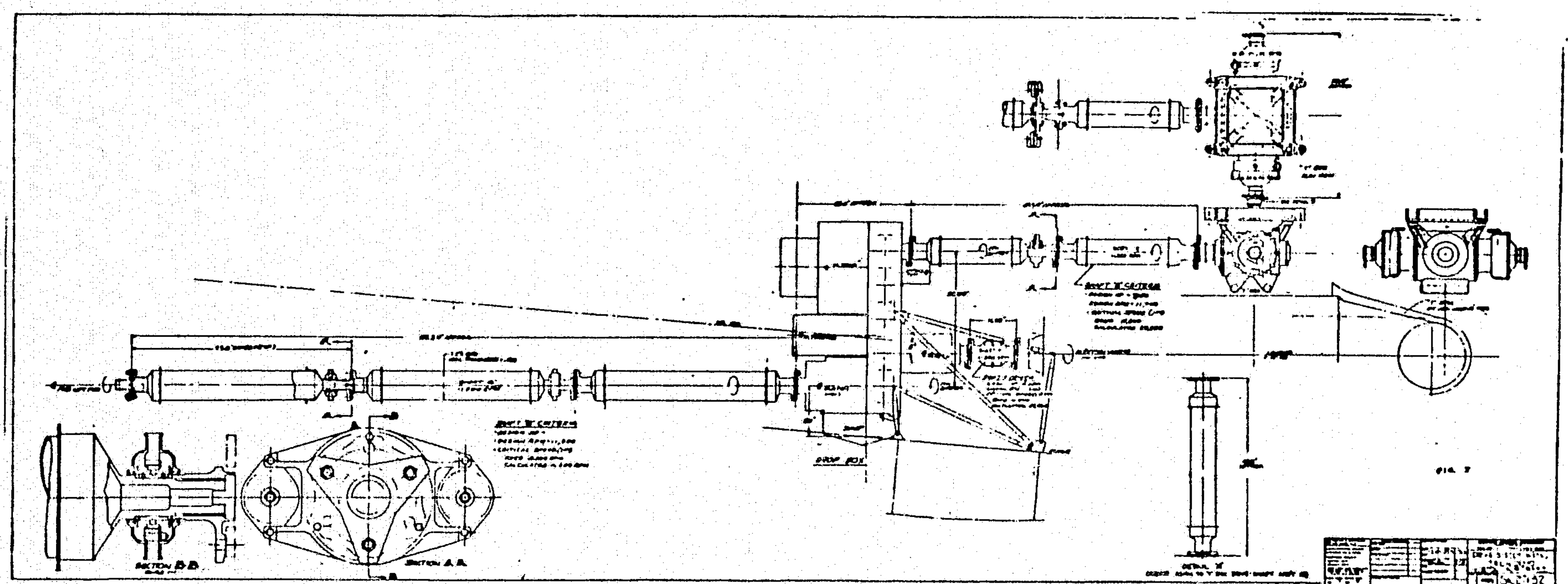


FIGURE 2.1-1 DRIVE SYSTEM INSTALLATION

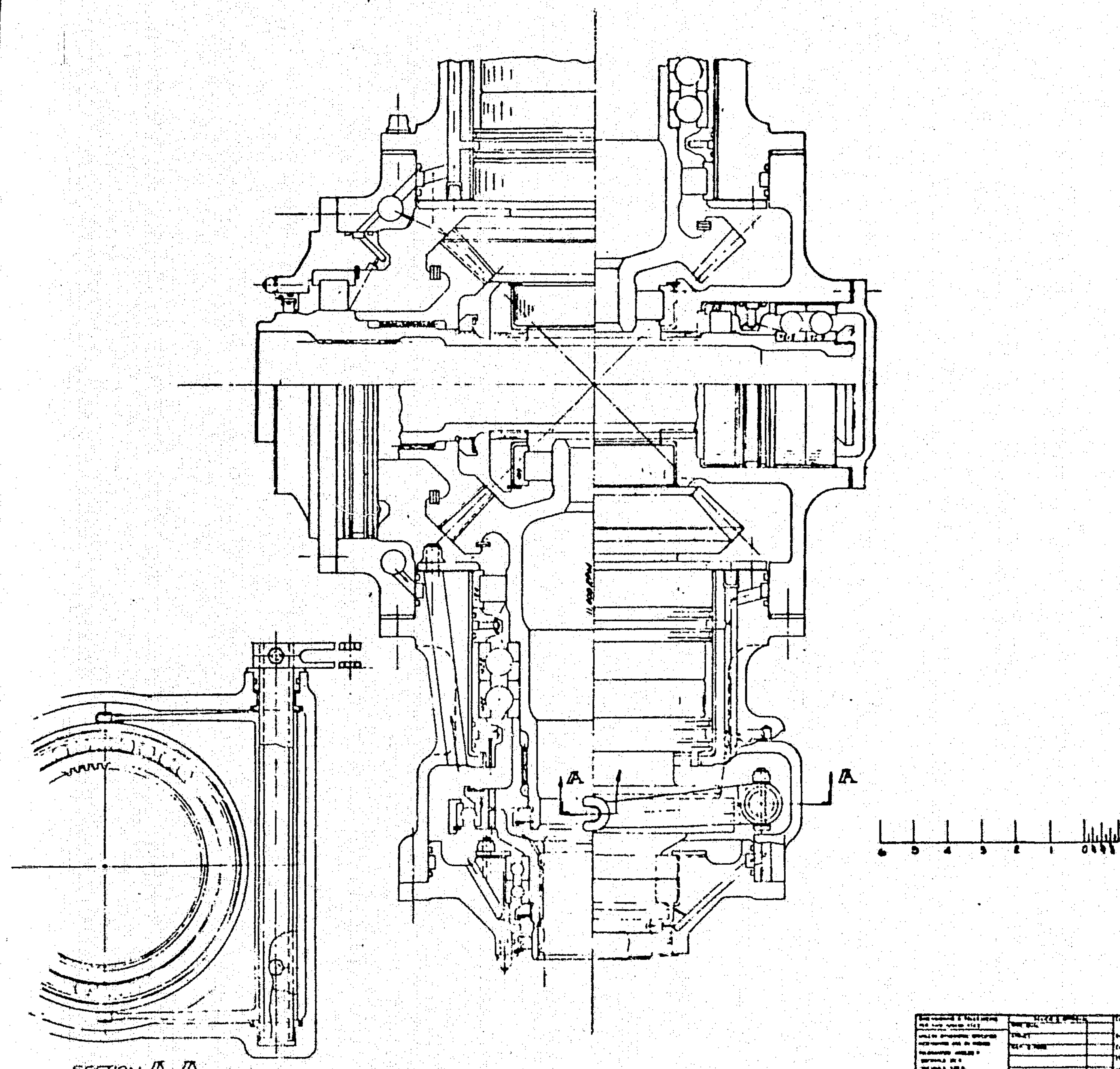


FIGURE 2.1-2 T-BOX LAYOUT

Because of the proximity of the #3 engine to the bottom of the T-box, there is no space for an oil sump. Therefore, the T-box is a dry sump (scavenged) gearbox and utilizes a separate oil tank for its oil supply. Two oil collection areas are provided in the bottom of the gear case. Each of these is scavenged by a separate scavenge pump element.

The three bevel gears in the transmission are conventionally straddle mounted between a pair of cylindrical roller bearings. The thrust bearings are designed to take thrust in either direction thus providing the driving or driven capability required of the box.

The detail design of the gear rims and webs is such that its resonant frequencies are removed from its operating speed. In addition, the gears are equipped with damping rings. These damping rings do not change the natural frequency of the gear but mechanically attenuate the resonance to a low (non-damaging)level.

The T-box is mounted in the aircraft from an "H" frame as shown in Figure 2.1-1. The "H" frame is an aluminum hog-out or forging which is designed to prevent loads resulting from aircraft deflection from being transmitted to the gearbox housing.

The gears in the T-box are made from VASCO X-2 steel. The housings will be magnesium hog-outs or castings whichever is cost effective. The bearings are CEVM-M-50 steel.

Oil is supplied to the gears and bearings via a combination of finger jets and lubricators.

The T-box weight of 380 lbs includes 43 lbs of oil and a 13 lb oil tank. Cooler and plumbing weight is excluded.

2.1.2 Drop Box

The drop box, Figure 2.1-3 and 2.1-4 is a double helical gearbox. Like the T-box, it is also a driving or driven gearbox. However, its primary function is to provide power to the forward lift fan for V/STOL operation. There is no speed change across the drop box gears. Each of the five herringbone gears is identical as far as the gear tooth configuration is concerned. Each gear is supported in identical roller bearings. This commonality minimizes tooling and special set-up requirements.

The primary power train gears will be made from VASCO X-2 steel whereas the accessory drive gears will be made from 9310 steel. The housings will be hog-outs or castings whichever is cost effective. The bearings will be CEVM-M-50 steel.

Oil is supplied directly to the gears by lubrication jets. The bearings and clutches are lubricated by centrifugal force with oil supplied to the bores of the gear/clutch shafts. Lubrication by this means is cost effective because of casting simplification and delivers oil more efficiently than external lubricators.

The transmission will be mounted in the aircraft utilizing an open truss arrangement as shown in Figure 1.5-1 and schematically shown in Figure 2.1-1. This provides satisfactory alignment between the engine and transmission.

The clutch at the forward end of the drop box is required to engage and disengage the forward lift fan as required by the flight mode. This clutch will be developed under separate contract by the US Navy.

In addition to its function of transferring #1, #2 or #3 engine power to the forward lift fan or transferring #3 engine power to the T-box, the drop box also mounts the following drive system and aircraft accessories:

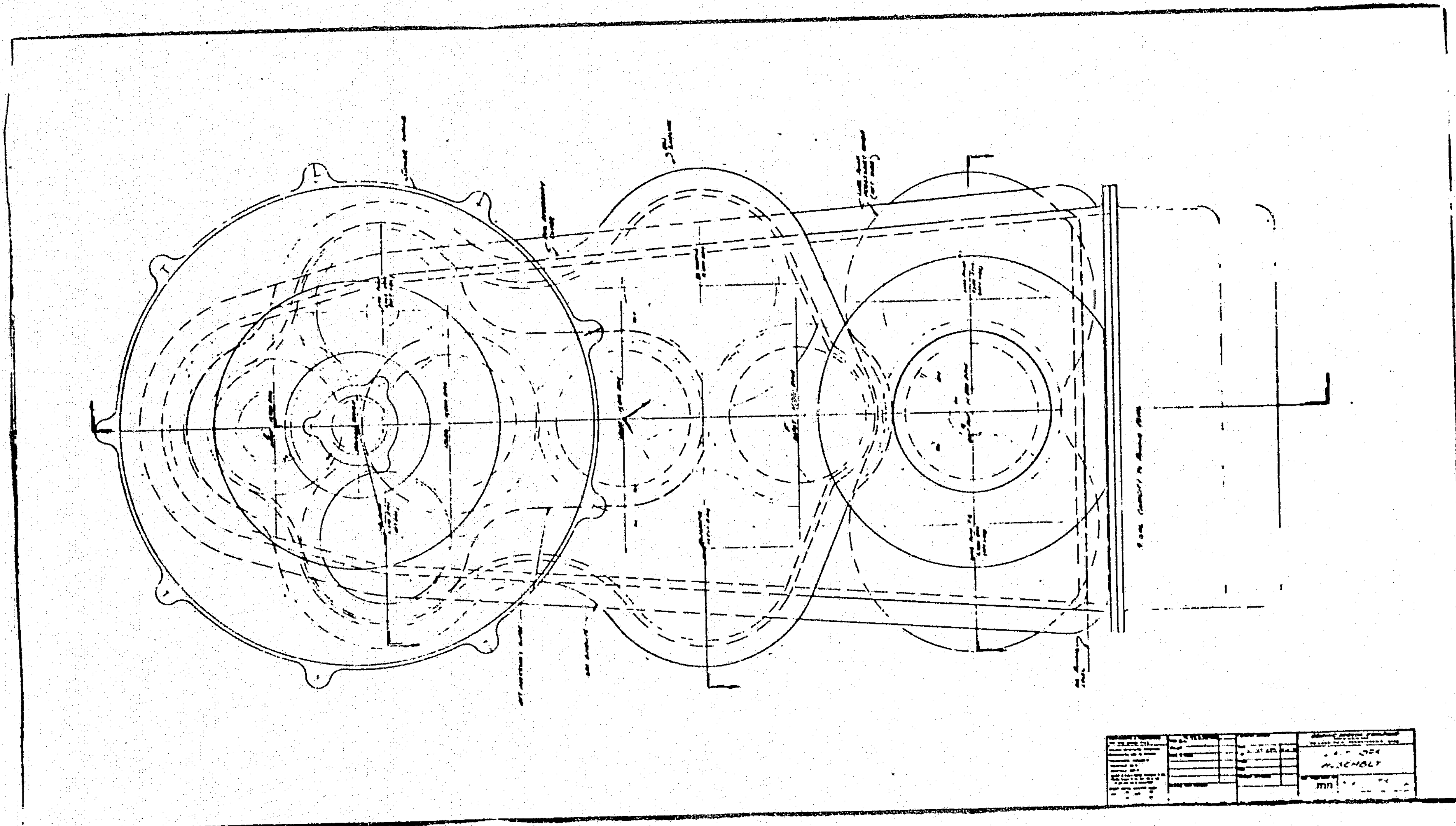


FIGURE 2.1-3 DROP BOX LAYOUT (External)

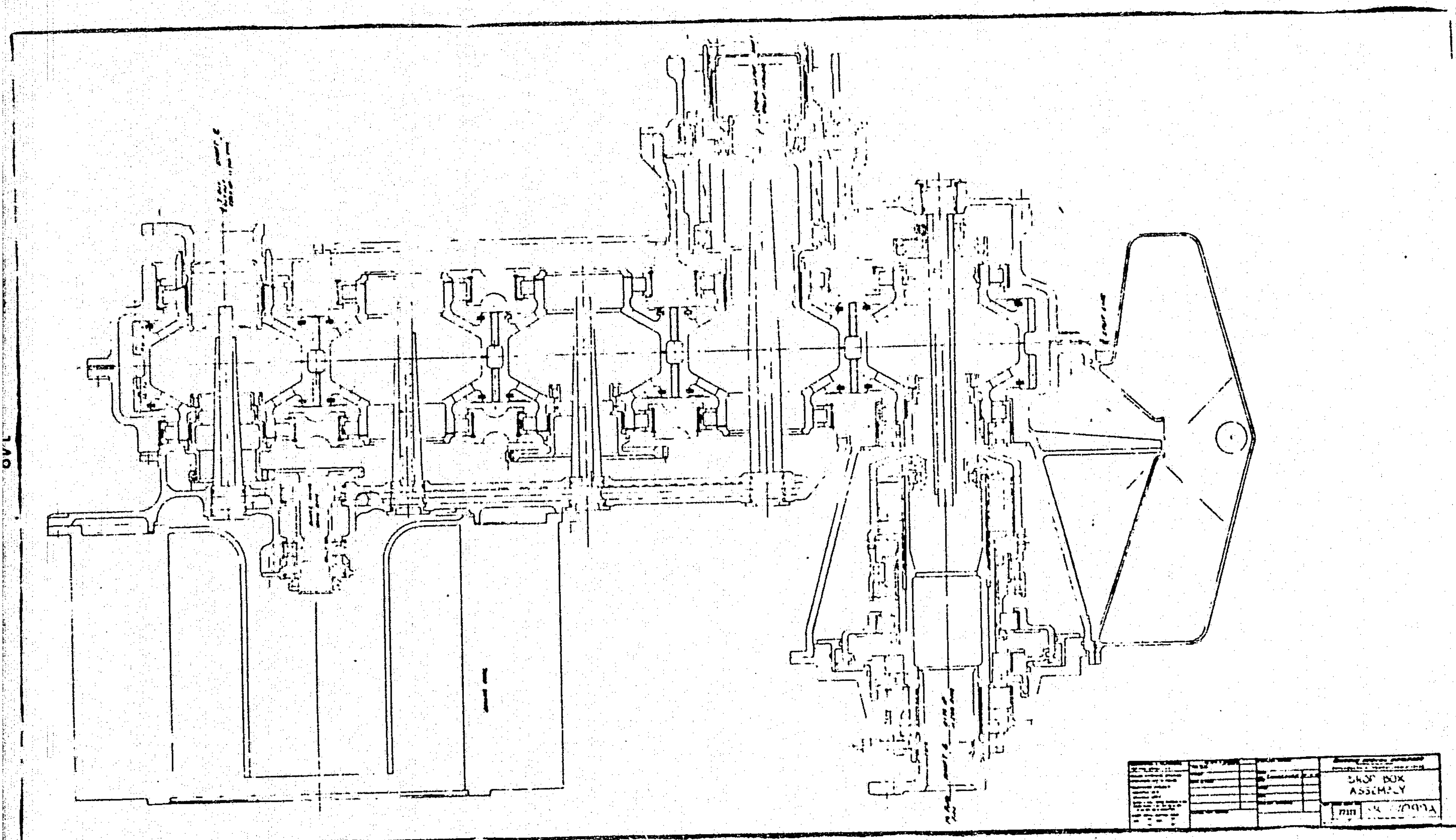


FIGURE 2.1-4 DROP BOX LAYOUT (Internal)

(1)	Oil Cooler - Drop Box	(67 Lbs)
(1)	Blower for Cooler	(22 Lbs)
(2)	Lubrication Pumps - Drop Box and T-Box	(25 Lbs)
(2)	20 KVA IDG's	(76 Lbs ea)
(2)	Hydraulic Pumps - 25 GPM (3000 PSI)	(11.5 Lbs ea)

The drop box has an integral oil sump, thus no scavenge pumps are required.

The drop box also includes an overrunning clutch which automatically disengages engine #3 from the drive system in the event of failure of #3 engine. The sprag clutch is the same hardware developed for the HLH (XCH-62). The drop box weighs 560 Lbs and includes 60 Lbs of oil.

2.1.3 Shafts and Bearings

The drive shafting shown in Figure 2.1-1 is large diameter, thin wall aluminum tubing, supported at the intervals shown for subcritical whirl mode operation. The bearings in the supports are grease lubricated. Each of the shaft sections is connected together by a balanced assembly of thin stainless steel discs to accommodate shaft misalignment. The splined couplings that connect the drive shaft assemblies to the transmissions are grease lubricated.

The drive shafting is identical in design to that developed and tested for the Boeing Vertol HLH (XCH-62). It is similar in concept, although larger in diameter, to that used successfully in the CH-46 and CH-47 helicopter rotor interconnect and engine drive locations. A typical XCH-62 drive shaft assembly is shown in Figure 2.1-5.

During the preliminary design of the shafting system for this aircraft, the drop box to lift fan shafting used 6.00 inch dia. tubing. In order to meet the subcritical operational speed requirements, four sections of shafting and three support bearings were required. A trade study was made using 7.25 inch dia. tube (HLH synchronizing shaft) and

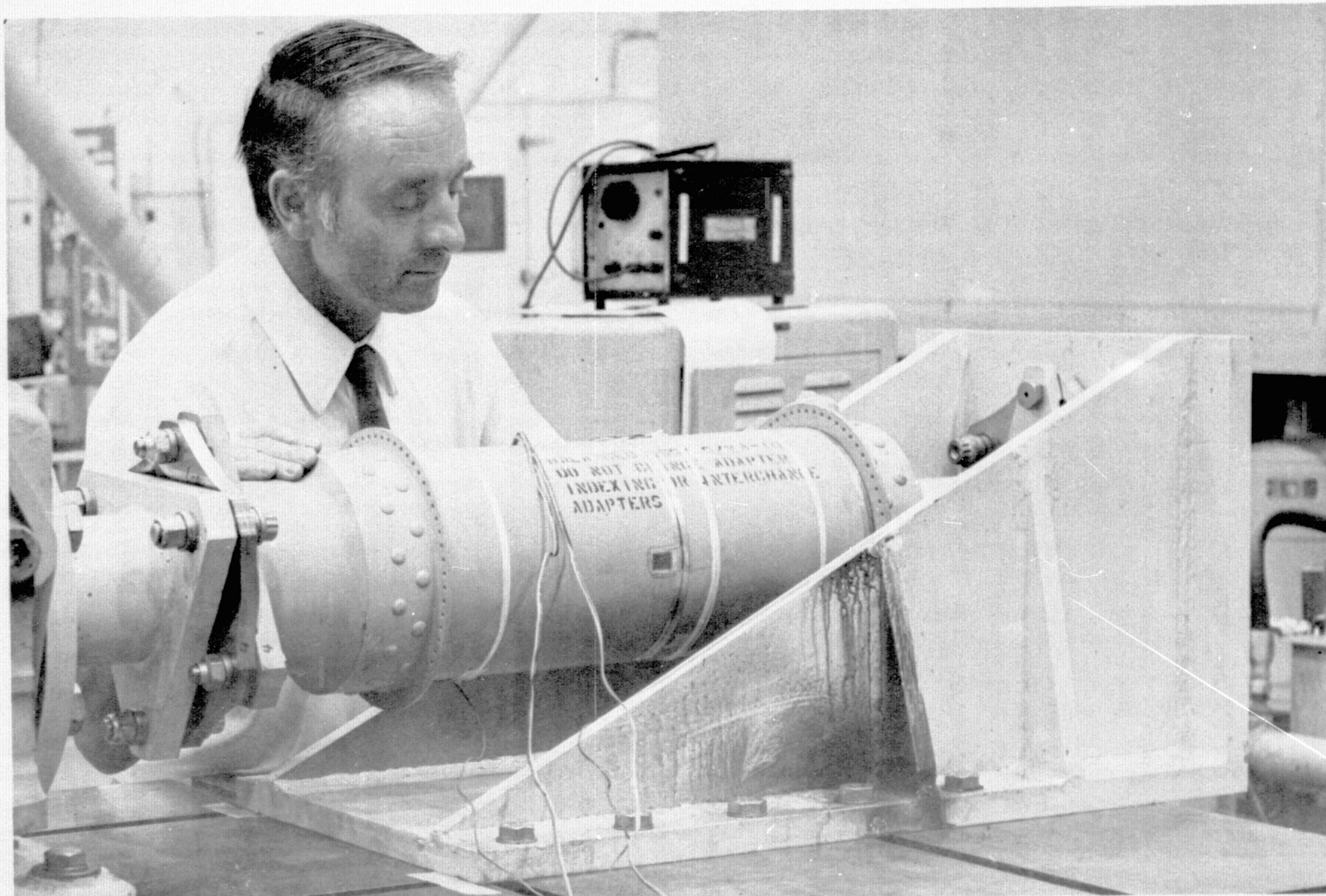


FIGURE 2.1-5 TYPICAL HLH DRIVE SHAFT ASSEMBLY

it was determined that only three shaft sections and two bearing supports would be required for subcritical speed operation. A weight savings of 23 Lbs was achieved by designing with the larger shafting.

The two drive shafts from the pod mounted engines to the T-box and the shaft from the #3 engine to the drop box will be instrumented and equipped with a telemetry system in order to provide the pilot (and flight test engineering) with torque and power distribution data. This system, has been used successfully during HLH and UTTAS drive system development testing. Figure 2.1-6 and 2.1-7 show the system schematically and installed on a UTTAS engine drive shaft.

The total shafting weight including couplings and bearing supports is 296 Lbs.

2.1.4 Lift Fan Engagement Clutch

The lift fan engagement clutch is being developed under separate Navy contract and will be GFE. The clutch will be mounted on the front of the drop box to facilitate actuation and lubrication oil supply. Conceptual design of the clutch was done during preproposal studies in response to the Navy RFP N0019-76-0-0058. The clutch weight was estimat 80 pounds. The clutch concept is described in section 2.5.4.

2.1.5 Lubrication & Cooling Systems

Major features are common to each of the lubrication systems for the T and drop boxes. See Figures 2.1-8 and 2.1-9. Each of the pump inlets is protected from debris by screens located in the bottom of the transmission. These screens are designed to detect ferrous and non-ferrous metals and provide an indication to the pilot in the event of a potential problem in the transmission. The oil filters are full flow, single stage, 40 micron. They are provided with a ground inspectable indicator which provides indication of impending bypass and actual bypass. A downstream screen with .015 inch opening at the transmission

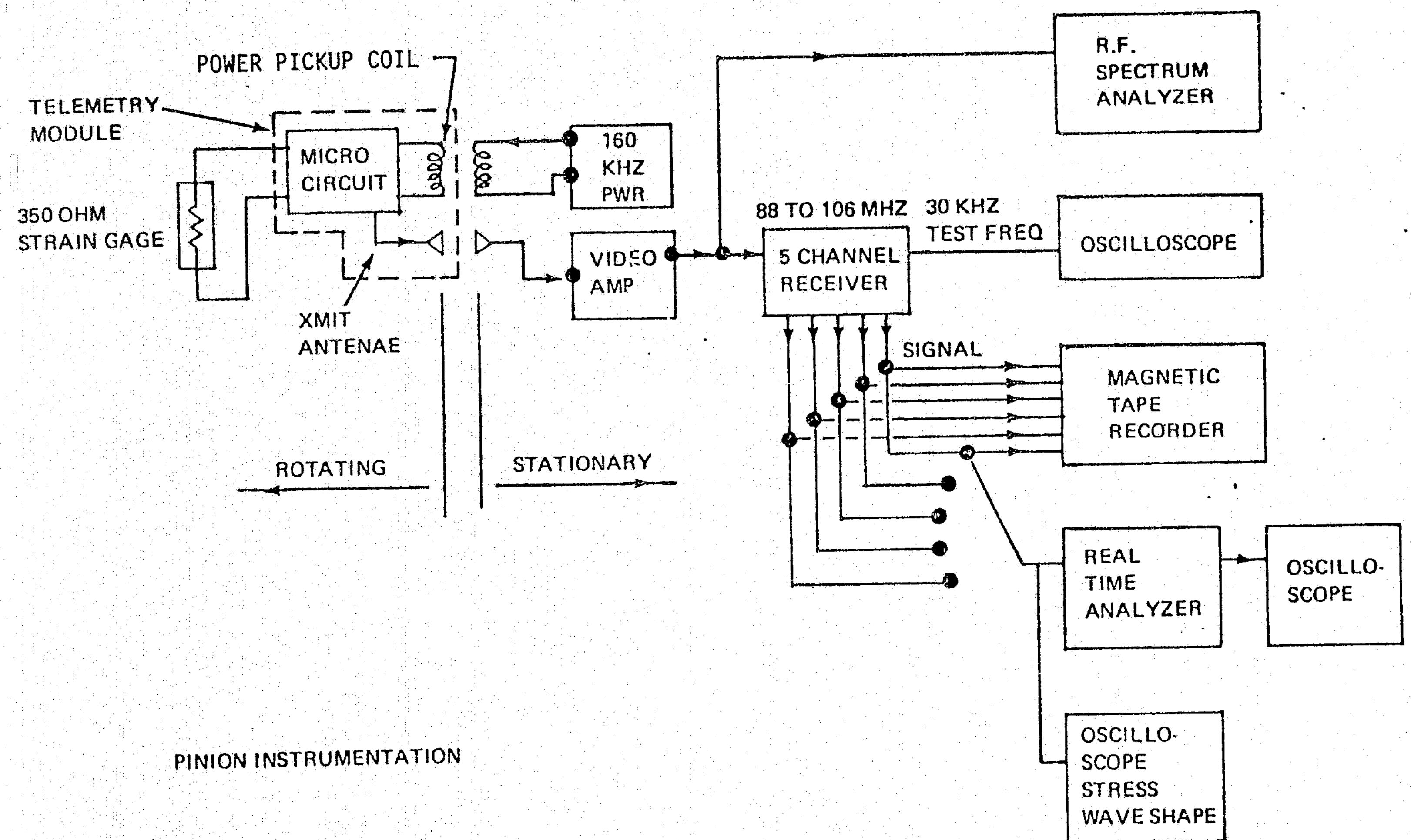


FIGURE 2.1-6 TELEMETRY SYSTEM FOR SHAFT TORQUE MEASUREMENT - INSTRUMENTATION BLOCK DIAGRAM

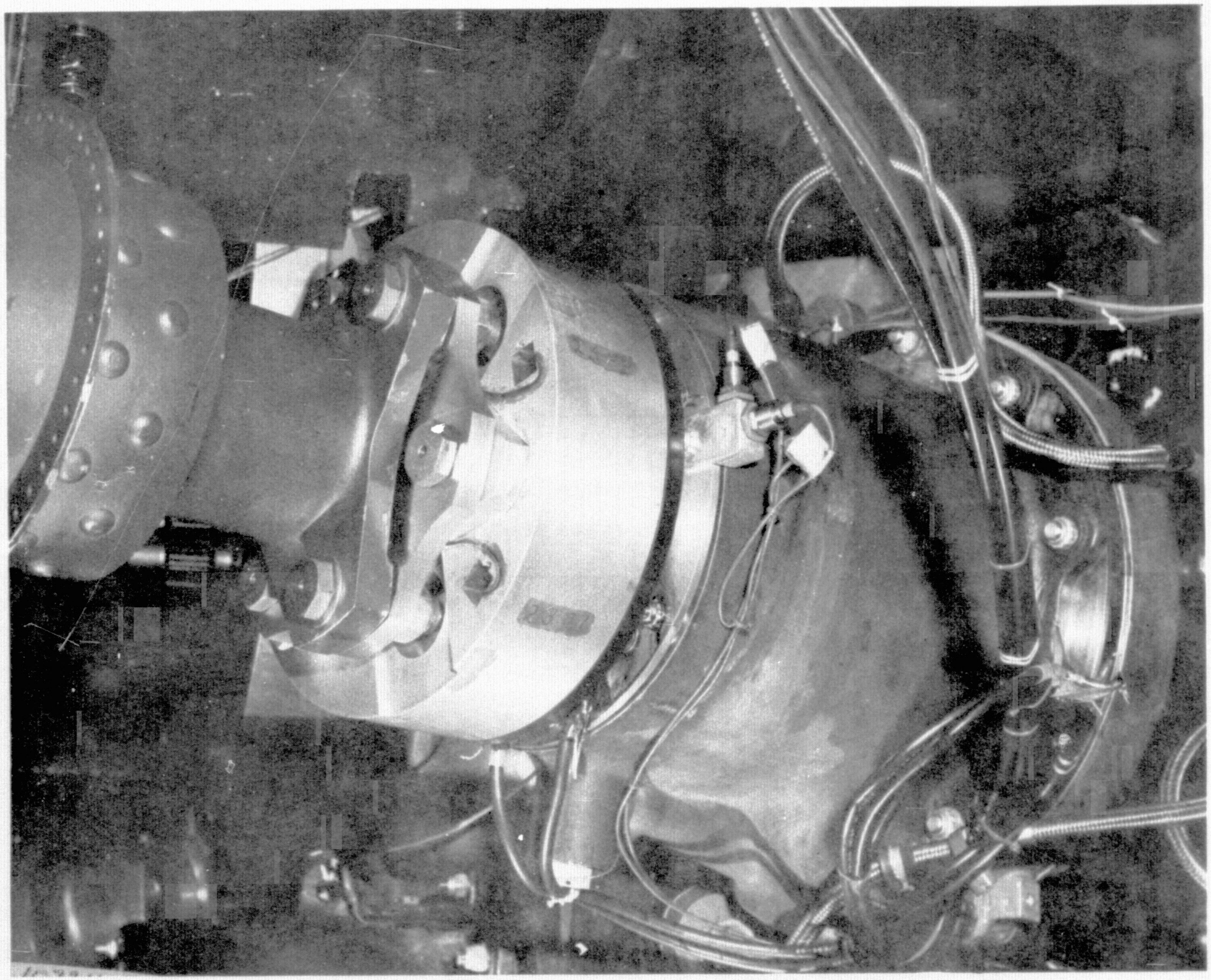


FIGURE 2.1-7 TELEMETRY SYSTEM FOR SHAFT TORQUE MEASUREMENT

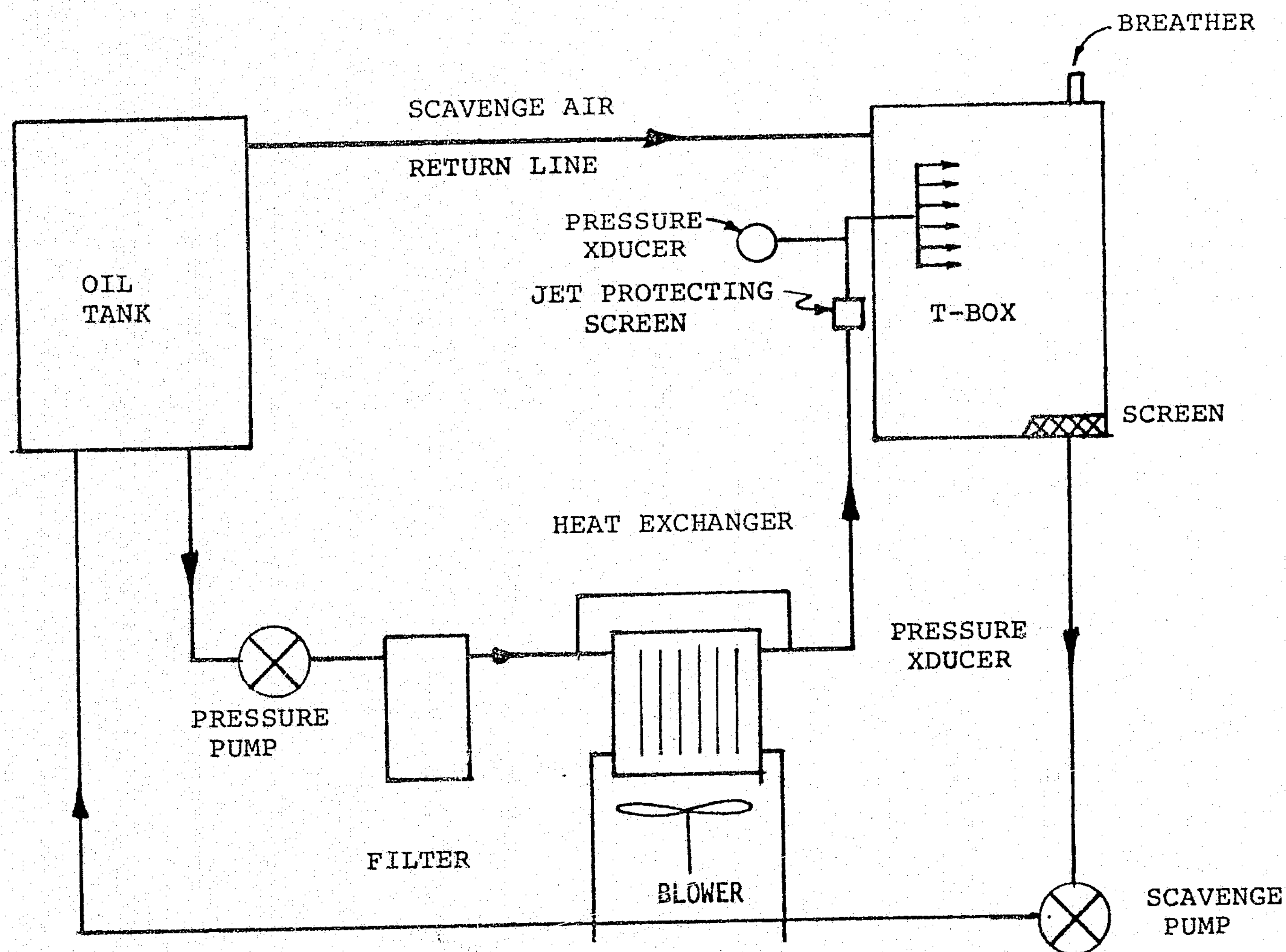


FIGURE 2.1-8 T-BOX LUBRICATION SYSTEM

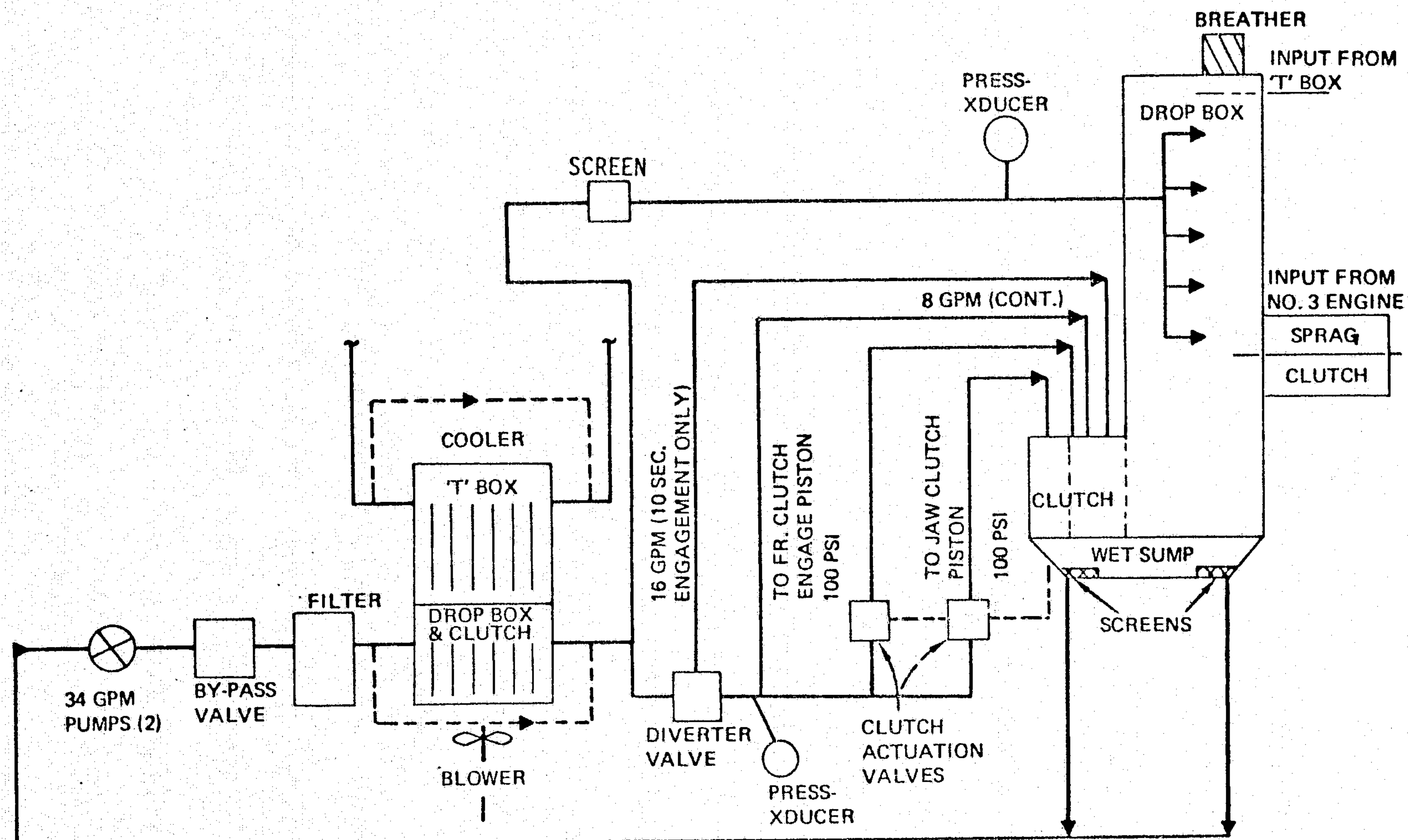


FIGURE 2.1-9 DROP BOX LUBRICATION AND CLUTCH ACTUATION - COMMON OIL SUPPLY

is provided to preclude filter bypass material clogging lube jets. This screen does not have a bypass. Debris build-up is indicated by the oil pressure gage in the cockpit.

The T-box is mounted in the aircraft just above engine No. 3. The proximity of the engine to the transmission prevents the utilization of an integral oil reservoir (sump). Thus a remotely mounted oil tank will be used. The T-box is scavenged from two low points, one forward and one aft to provide positive scavenging for 45^{o} nose up and 45^{o} nose down attitudes. Because of space limitations, the lubrication pump for the T-box is mounted on and driven by the drop box. It is a 3 element, positive displacement GEROTOR pump; one 18 GPM pressure element and two 45 GPM scavenge elements. Hoses are used to connect the pump to the T-box, the oil tank and the oil cooler. A schematic of the T-box lubrication system is shown in Figure 2.1-8. The lubrication and cooling system is sized for the STO climb Case I (Figure 2.1-10). This design point was selected because it is the most severe steady state standard operating condition. The heat rejection analysis is shown in Figure 2.1-11 based on power levels in section 2.3.1.

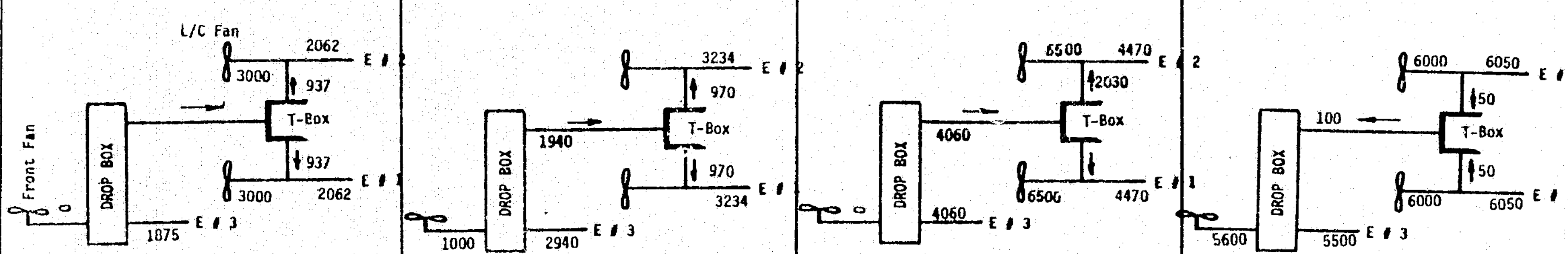
The drop box is a wet sump gearbox, thus no scavenge pumps are required. The inlet to the pressure pump will be designed to accommodate the 45^{o} nose up and down aircraft attitudes. The lubrication and cooling system for the drop box is also sized for the STO climb Case I (Figure 2.1-10,) its heat rejection analysis is presented in Figure 2.1-12.

The drop box and T-box are cooled by air, utilizing the HLH oil coolers and blower which mount on the transmission. The blower is gear driven by an accessory drive gear in the drop box.

LIFT FAN V/STOL TRANSMISSION OIL COOLING

DESIGN POINT ANALYSIS

CONDITION - A	CONDITION - B	CONDITION - C	CONDITION - D
CRUISE	APPROACH (CASE I)	CLIMB (CASE I)	HOVER - A.E.O.



Note:
Numbers shown are
Power Levels (Horsepower)
Transmitted by that
Element. Arrows indicate
direction of power flow

FIGURE 2.1-10

LIFT FAN V/STOL "T-BOX" OIL HEAT REJECTION

MESH DESIGNATION	DESIGN HP	MESH WINDAGE LOSS	CONDITION - A (Cruise) TRANS-MITTED HP	MESH POWER LOSS	TOTAL MESH LOSS	CONDITION - B (Approach) TRANS-MITTED HP	MESH POWER LOSS	TOTAL MESH LOSS	CONDITION - C (Climb) TRANS-MITTED HP	MESH POWER LOSS	TOTAL MESH LOSS	CONDITION - D (Hover) TRANS-MITTED HP	MESH POWER LOSS	TOTAL MESH LOSS
# 1	5450 HP	27 HP	940 HP	5 HP	32 HP	970 HP	5 HP	32 HP	50 HP 2030 HP	10 HP	37 HP		.3 HP	28.3 HP
# 2	5450 HP	27 HP	940 HP	5 HP	32 HP	970 HP	5 HP	32 HP	50 HP 2030 HP	10 HP	37 HP		.3 HP	28.3 HP
HP TOTALS / BTU/MIN								64 HP			74 HP			56.6 HP

CONDITIONS/NOTES:

1. 50% Windage Loss/50% Power Loss
2. 1.0% Total Loss per Mesh
3. Size Lubrication & Cooling System for Condition C (STO Climb Case I)
4. Oil/air Heat Exchanger required to cool T-Box.

OIL FLOW REQUIREMENTS

$$Q = \frac{BTU/MIN}{SP.HT \times \Delta T}$$

$$Q = \frac{3140}{.51 \times 50^{o}} = 123 \text{ LB/MIN}$$

$$GPM = \frac{123}{7.4} \times 1.1 = 18 \text{ GPM}$$

AIR FLOW REQUIREMENT

$$W = \frac{BTU/MIN}{SP.\ HT \times \Delta T}$$

$$W = \frac{3140}{.24 \times 65} = 201 \text{ LB/MIN}$$

$$CFM = \frac{201}{.061} = 3300 \text{ CFM}$$

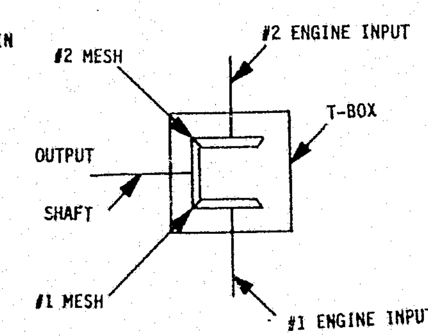


FIGURE 2.1-11

LIFT FAN V/STOL "DROP BOX" OIL HEAT REJECTION STUDY

MESH DESIGNATION	DESIGN HP (See Table 2.3-2)	MESH WINDAGE LOSS	CONDITION - A (Cruise) TRANS-MITTED HP	CONDITION - A (Cruise) MESH POWER LOSS	CONDITION - A (Cruise) TOTAL MESH LOSS	CONDITION - B (Approach) TRANS-MITTED HP	CONDITION - B (Approach) MESH POWER LOSS	CONDITION - B (Approach) TOTAL MESH LOSS	CONDITION - C (Climb) TRANS-MITTED HP	CONDITION - C (Climb) MESH POWER LOSS	CONDITION - C (Climb) TOTAL MESH LOSS	CONDITION - D (Hover) TRANS-MITTED HP	CONDITION - D (Hover) MESH POWER LOSS	CONDITION - D (Hover) TOTAL MESH LOSS
#1	8174	24 HP	0 HP	0 HP	24 HP	1000 HP	3 HP	21.6 HP	0 HP	0	24 HP	5600	17 HP	41 HP
#2	8174	24 HP	1875 HP	5.6	24.6 HP	1940 HP	5.8 HP	29.8 HP	4060 HP	12.2 HP	36 HP	100	.3 HP	24.3 HP
#3	8174	24 HP	1875 HP	5.6	29.6 HP	1940 HP	5.8 HP	29.8 HP	4060 HP	12.2 HP	36 HP	100	.3 HP	24.3 HP
#4	8174	24 HP	1875 HP	5.6	29.6 HP	1940 HP	5.8 HP	29.8 HP	4060 HP	12.2 HP	36 HP	100	.3 HP	24.3 HP
			150 HP	.9	.9	150	.9 HP	.9 HP	150 HP	.9 HP	.9 HP	150	.9 HP	.9 HP
Acces. Drive	150 (60 Hyd/ 90 Elect)													
HP TOTALS BTU/MIN		96 HP			113 HP			112 HP			133 HP			115 HP

CONDITIONS/NOTES:

1. 50% windage loss/50% power loss
2. 0.60% total loss per mesh
3. Size Lubrication & Cooling System for Condition C (510 Climb Case I)
4. Use HLH Fan Assy & Oil Cooler to Air Cool Drop Box

OIL FLOW REQUIREMENTS

$$Q = \frac{BTU/MIN}{SP.\ HT.\ x\ \ T}$$

$$Q = \frac{6357}{.51 \times 50} = 249\ LB/MIN$$

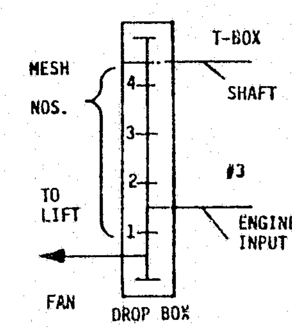
$$GPM = \frac{249}{7.4} \times 1.1 = 37\ GPM$$

AIR FLOW REQUIREMENTS

$$W = \frac{BTU/MIN}{SP.\ HT.\ x\ \ T}$$

$$W = \frac{6357}{.24 \times 65^0} = 408\ LB/MIN$$

$$CFM = \frac{408}{.061} = 6680\ CFM$$



* Includes Friction Clutch Residual drag of 17 HP (Cruise Mode)

FIGURE 2.1-12

In addition to lubricating the drop box and the forward lift fan engage clutch, the system is also designed to provide the hydraulic pressure required to actuate the friction and jaw clutches (see Figure 2.1-9) and during friction clutch engagement, divert a flow of oil of 22 GPM to the friction clutch for the 10 second engagement cycle.

2.2 STUDY APPROACH

Design objectives of the Technology Demonstrator drive system are characterized as follows:

- o Minimize development risk through conservatism in design allowables.
- o Minimize cost through maximum use of existing components and by design and fabrication techniques which reflect cost conscious technical approaches
- o Trade off weight to assist in achieving objective consistent with attainment of aircraft operational requirements

The design approach achieves the first objective by selecting component arrangements that avoid development risk areas and by using design allowables that are within current experience. The power requirements of the 3 engine, technology demonstrator aircraft are such that component velocities are also within current experience limits at the stress levels selected. The objective achieved of minimum cost is by providing maximum possible commonality between parts and by using existing components where possible to avoid engineering and development costs.

The drop box illustrates both a component arrangement that minimizes risk and also allows the use of common components. Placing #3 engine input and the reengagement clutch on alternate shafts allows introduction of lubricating oil to both members from the open shaft ends. Combining these components into one shaft would necessitate development of a high speed hydraulic slip ring to pass oil into the shaft interior. The cost of the drop box is minimized by common bearings, common gear sizes and profiles, and similarity in gear shaft design through the five gear locations.

Drop box gear are designed as high contact ratio helicals rather than spur gears to reduce dynamic loads and potentially damaging resonances that degrade life and reliability. To provide a lighter and less expensive bearing system the gears are double helicals, with opposing thrusts that eliminate the need for thrust bearings and the extra complexities of housing and lube system needed to support them.

Gear stress levels in the drop box are compared to experience in Figure 2.2-1. High speed parallel shaft gearing is limited in helicopter operational experience, however the pitch line velocities of 24,000 fpm does not present any known problems when compared to aircraft bevel gear experience and to commercial helical gearbox experience.

The T-box arrangement of two input pinions was selected so that both engine nacelle transmissions would be identical. An alternate gear arrangement using one pinion would have required opposite directions of rotation from the nacelles, and thus would have made left hand and right hand nacelle gearboxes necessary. In the technology demonstrator aircraft program, every attempt will be made to limit spare components. Common parts, and interchangeability between parts, will ease support requirements and minimize spares inventory.

The reengagement clutch design was selected from Boeing's recent proposal to the Navy (see reference 3). It represents a state-of-the-art design in all important parameters (disc peripheral speed, thermal and mechanical loading) at the design power of 11,000 h.p. In the technology demonstrator the requirement is 8174 h.p. or 75% of the design power. Therefore the clutch is operating at a significant margin below its rated capability.

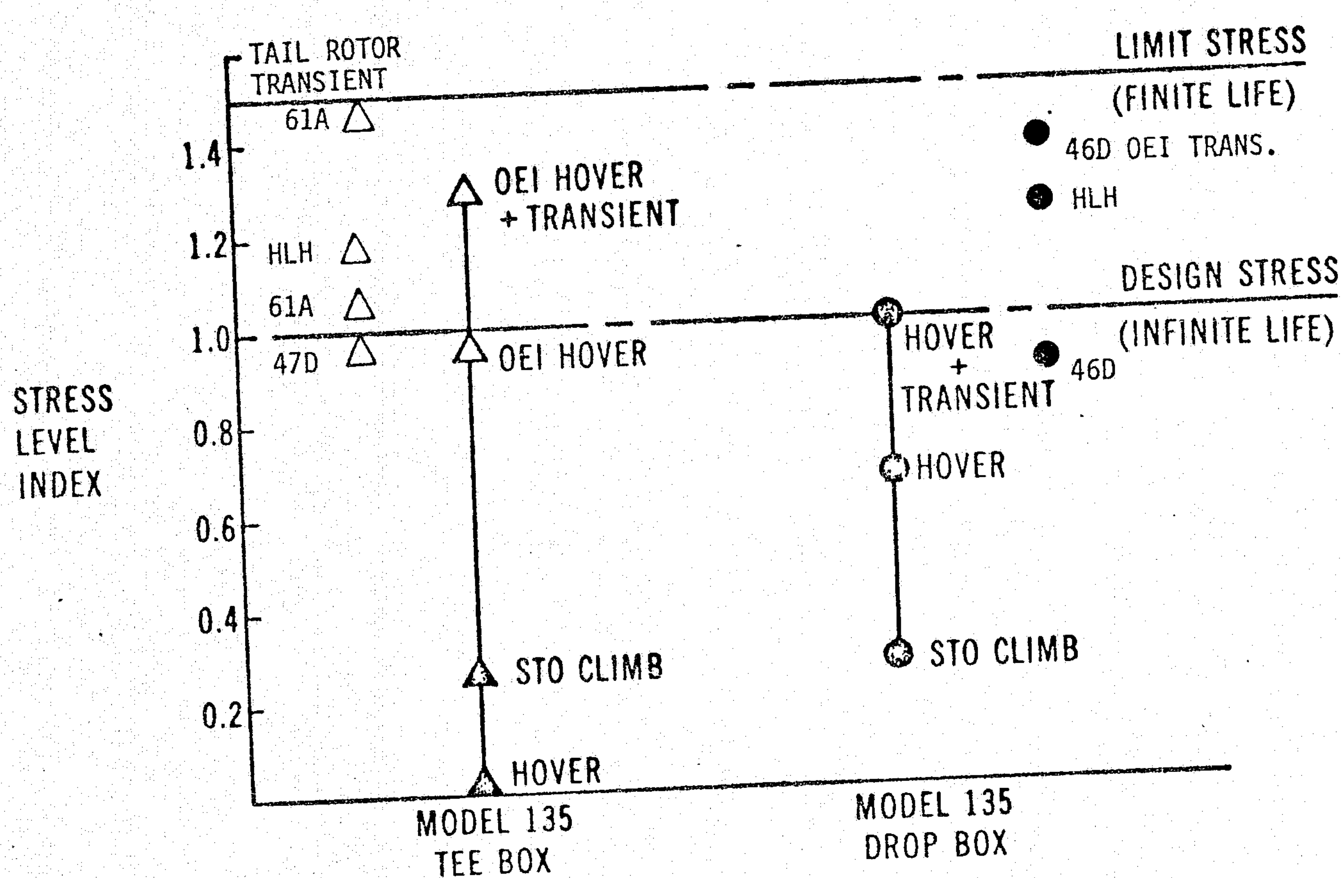


FIGURE 2.2-1 GEAR STRESSES RELATED TO EXPERIENCE

Drive shafting detail design is taken directly from HLH (XCH-62A) development program. Engine drive shafts were designed to connect Allison XT701 engines to a combiner gearbox at the same power and rpm required for the Technology Demonstrator. Development cost and risk is minimized by this previous work, which included over 200 hours of power-on rotation, a torsional fatigue test of shaft and adapters, and a several-thousand-hour endurance test to qualify the shaft support bearings for flight.

The overrunning clutch is also taken from the HLH program, where identical requirements prevailed. The HLH program provides overrunning endurance testing, torsional fatigue and ultimate test data to substantiate the basic clutch design. Since the operational concept of the Technology V/STOL aircraft is to keep all three engines on the line during all flight modes, the overrunning to be expected is confined to an engine-out, or Level II, condition.

Selection of design allowables, and resulting component velocities, are within experience limits as shown in the accompanying figures. In the T-box, spiral bevel bending stress levels are within CH47 fleet experience (Figure 2.2-1) and so are pitch line velocities (Figure 2.2-2). Comparison is made to the CH-47C engine nose box, the relative size of which is shown overlaid on the T-box (Figure 2.2-3). Note that the input shaft rpm of the CH-47C box is higher (16,000 vs 11,500). The maximum current rating of the nose box is 4200 h.p.

Bearing sizing and velocities are directly influenced by load, speed and life requirements. Because the V/STOL mission includes operation at comparatively low powers, the prorated (cubic mean) load is lower than is normal for a typical helicopter design. In consequence, bearing sizes and velocities fall well within the bounds of development test experience (typified by the HLH), and are in the region of our fleet experience (Figure 2.2-4).

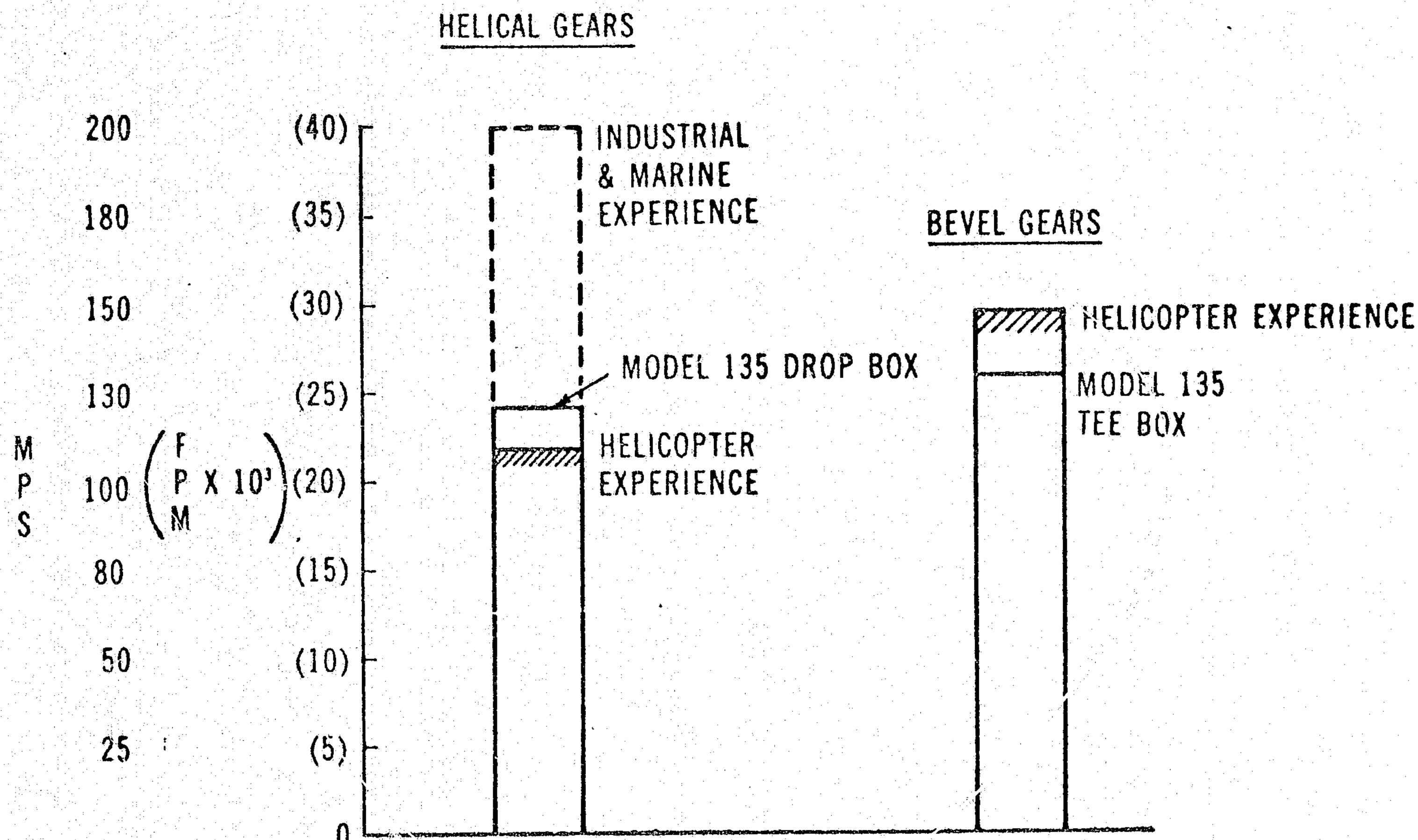


FIGURE 2.2-2 GEAR PITCH LINE VELOCITY RELATED TO EXPERIENCE

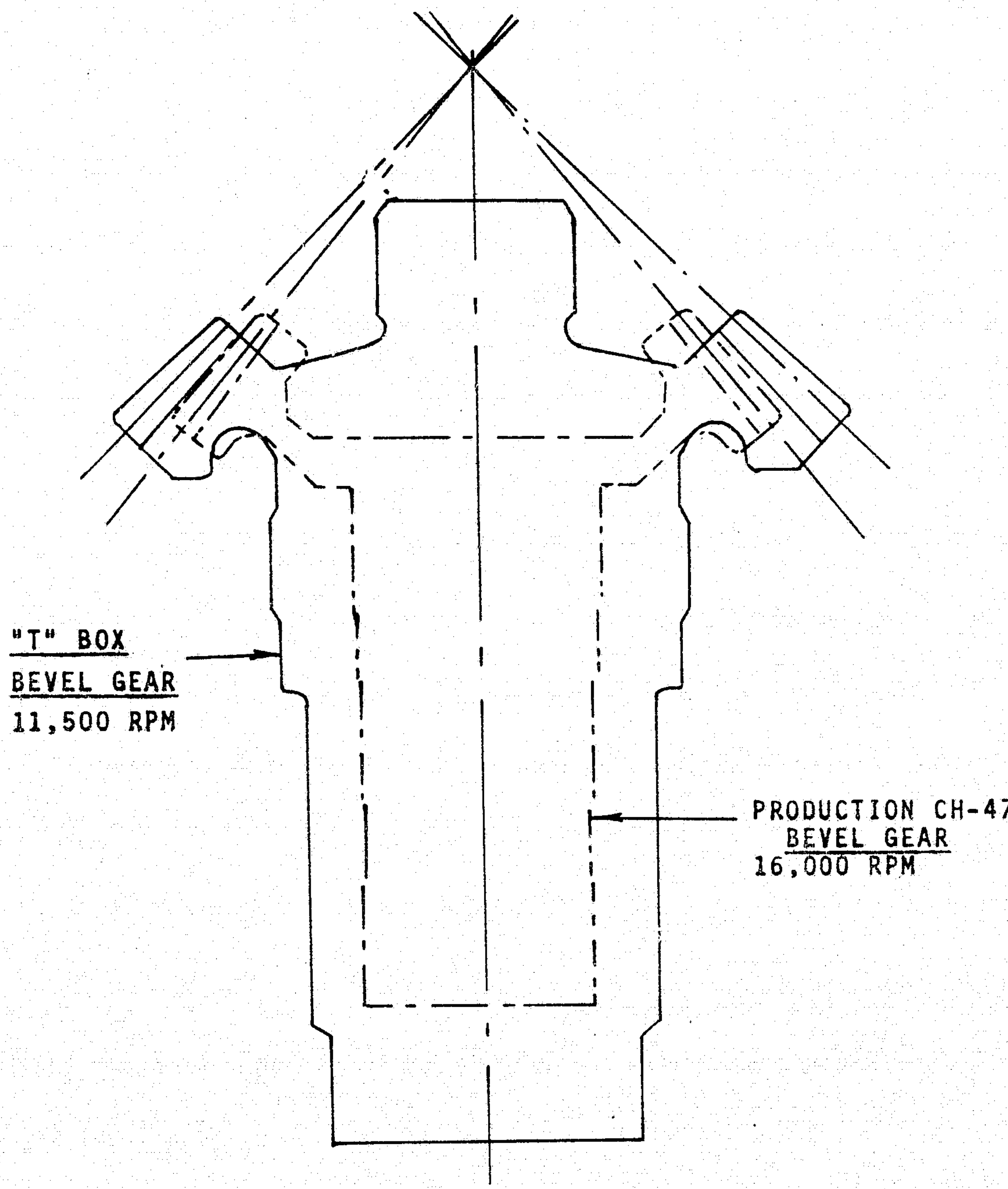


FIGURE 2.2-3 T-BOX BEVEL GEAR SIZE COMPARISON

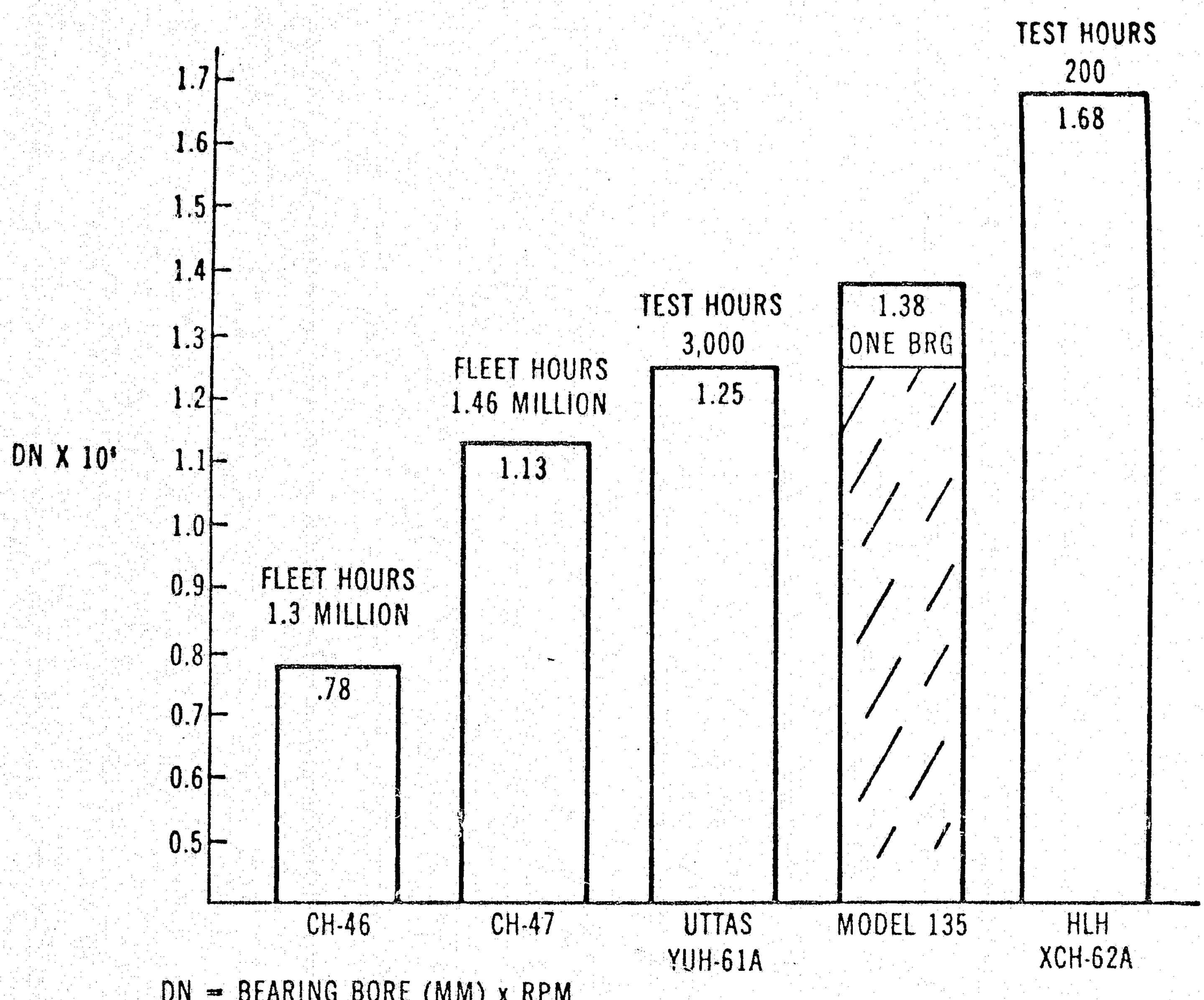


FIGURE 2.2-4 BALL AND ROLLER BEARING VELOCITY RELATED TO EXPERIENCE

Minimization of development risk and assurance of success is dependent upon the depth of analysis. In this study we have performed preliminary gear and bearing sizing using computer tools developed for this purpose and arrived at a Transmission Layout based upon this sizing.

2.3 REQUIREMENTS AND CRITERIA

The requirements and criteria for use in the preliminary design of the major components of the Model 135 RTA drive system are addressed. These include the cross shaft transmission, T-box, the third engine input transmission, drop box, their lubrication and cooling systems; the lift fan engagement clutch and connecting drive shafting. Figure 2.3-1 illustrates the drive system arrangement.

2.3.1 Mission Load Profile

Steady state hover and maximum transient thrust control requirements were calculated for both standard day and 90°F day operation. Thrust requirements for all engines operating, and any single engine out were computed against the work statement Level I and Level II criteria. The resulting simultaneous three fan thrust requirements for various control maximum are shown in Table 2.3-1. These data, in conjunction with anticipated airplane duty cycle, based on the statement of work mission requirements, were used to determine the RTA mechanical interconnect power transfer requirements for both levels of operation.

RTA airplane weights for this analysis were 26,300 lb for standard day VTO operation, 25,050 lb for hot day and 25,400 lb for one engine inoperative (OEI). Power levels were determined from required thrust assuming a thrust to horsepower ratio of 1.5 for steady state thrust levels, 1.36 at maximum thrust all engines operating and 1.67 O.E.I. Maximum thrust at any fan was determined assuming the fan will transition to the beta stop. Further, the beta stops on all fans are set the same and at the position required for the 90°F day maximum thrust condition. From Table 2.3-1, the maximum required thrust for 90° F day

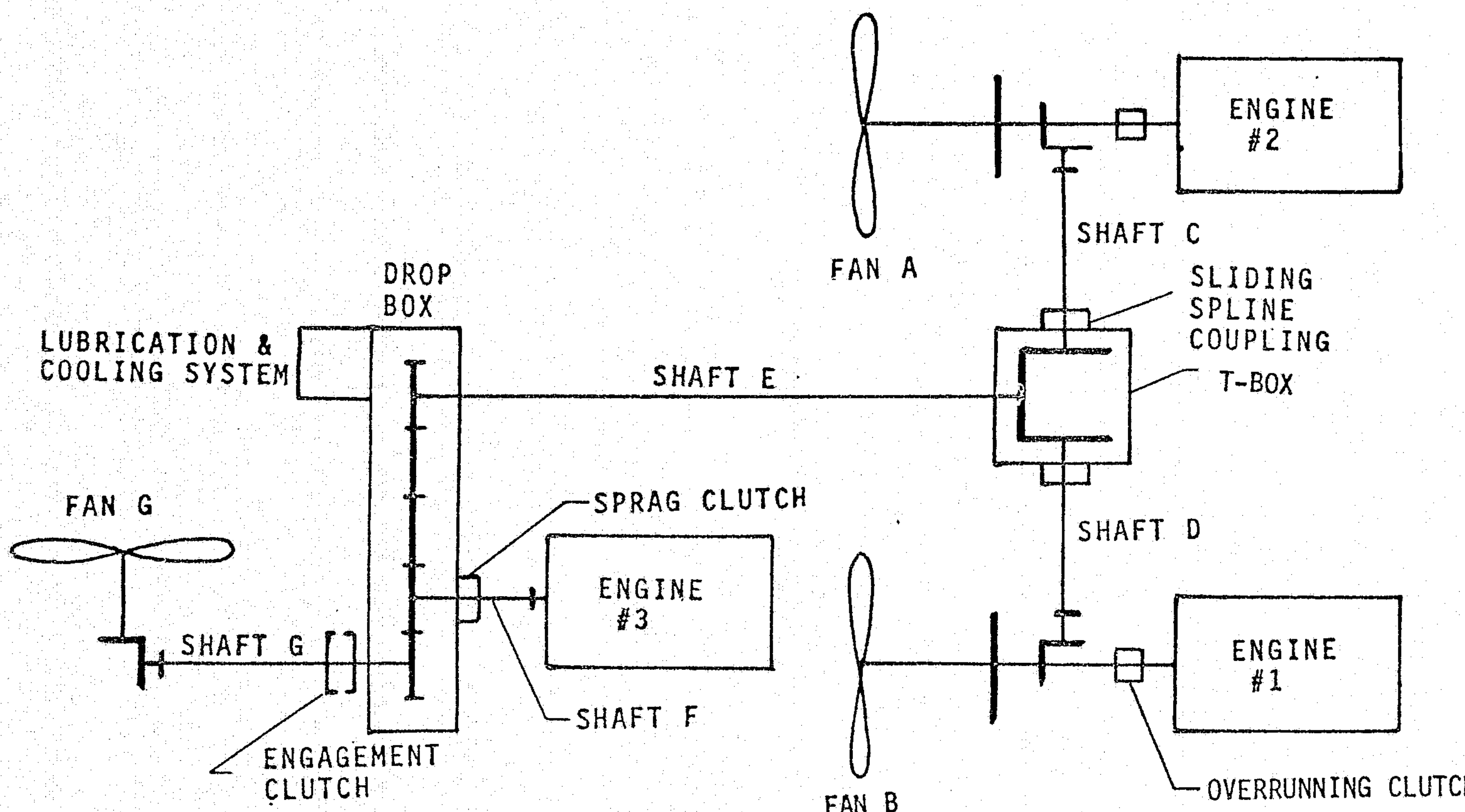


FIGURE 2.3-1 POWER TRAIN SYSTEM SCHEMATIC

TABLE 2.3-1 MAXIMUM THRUST REQUIREMENTS

CRITICAL CONDITION \ SIMULTANEOUS ATTITUDE CONTROL REQUIREMENTS		G. W. = 26,300 Lbs (Std. Day A/P Wt.)						G. W. = 25,050 Lbs (90° Day A/P Wt.)					
		LEVEL 1			LEVEL 2			LEVEL 1			LEVEL 2		
		L/C-F A	L/C-F B	L-F G	L/C-F A	L/C-F B	L-F G	L/C-F A	L/C-F B	L-F G	L/C-F A	L/C-F B	L-F G
A	L/C Fan Peaked	10908	9106	8917	9954	9054	8608	10432	8662	8483	9503	8629	8192
	L Fan Peaked	9548	8882	10436	9142	8748	9662	9100	8450	9968	8714	8332	9219
B	L/C Fan Peaked	10955	7531	7814	9946	8326	8028	10507	7139	7424	9506	7928	7636
	L Fan Peaked	8447	7295	10494	8533	7923	9780	8045	6915	10052	8128	7536	9348
C	L/C Fan Peaked	10780	7438	8082				10351	7031	7688			
	L Fan Peaked	9842	7216	9601				9019	6819	9173			

	CRITICAL CONDITION (From Statement of Work)		
	A	B	C
LEVEL 1	Hover 100% Height (.1g's) 50%/15%/15% 5" Longit cg trim .5 Lat. c.g. Trim	Hover 0 Height 100/30/30 5" Longit c.g. Trim .5" Lat. c.g. Trim	Hover 25 Kn Crosswind Trim .5" Lat c.g. Trim 50/15/15/
LEVEL 2	Hover 100% Height (0.05 g's) 50%/15%/15% .5 Lat. c.g. Trim 5" Longit. c.g. Trim	Hover 0 Height 100/30/30 5" Longit c.g. Trim .5" Lat c.g. Trim	NO REQUIREMENT

operation is 10507 lb (lift/cruise fan thrust). The thrust then at standard day operation is 10507 x 1.058 = 11116 lb. The maximum transient horsepower at any fan is then:

$$\text{Power} = \frac{11116}{1.36} = 8174 \text{ HP}$$

Maximum transient power levels shown in Table 2.3-2 were determined assuming any one fan is at this power level of 8174 HP with the other fans at steady state condition.

The maximum steady state power values shown in Table 2.3-3 were selected from the complete breakdown at values for each flight condition shown in Table 2.3-4. The estimated percent of total operating time for each condition is also shown in Table 2.3-4.

These power levels were used in the design of the power train components. General requirements for each component are listed in the following paragraphs. The design conditions for the gearboxes were:

- o Design to maximum transient power with all engines operating
- o Maximum O.E.I. Steady State Power not to exceed design power
- o Maximum O.E.I. transient power not to exceed 1.5 times design power

General requirements for each component are listed in the following paragraphs:

TABLE 2.3-2

SUMMARY OF PEAK TRANSIENT

POWER LEVELS

	ALL ENGINES OPERATING	ENGINE 1 FAILED	ENGINE 2 FAILED	ENGINE 3 FAILED
E1	6935	0	8294	8294
E2	6935	8294	0	8294
E3	6300	7540	7540	0
Fan A	8174	8174	8174	8174
Fan B	8174	8174	8174	8174
Fan G	8174	8174	8174	8174
Shaft C	2210	8174	8174	8174
Shaft D	2210	8174	8174	8174
Shaft E	4421	8174	8174	8174
Shaft F	6300	7540	7540	7540
Shaft G	8174 [1]	8174	8174	8174

Design Point for Drop Box

T-box design power level was determined from OEI maximum transient power (8174/1.5 = 5450 HP) which is more than twice the maximum transient all-engines operating power and slifhtly greater than OEI steady state power (see Table 3)

TABLE 2.3-3

SUMMARY OF MAXIMUM POWER LEVELS

STEADY STATE CONDITIONS

	HORSEPOWER LEVELS			
	ALL ENGINES OPERATING	ENGINE 1 FAILED	ENGINE 2 FAILED	ENGINE 3 FAILED
E 1	6050	0	8206	7830
E 2	6050	8206	0	7830
E 3	5500	7460	7460	0
Fan A	6000	5350	5350	5350
Fan B	6000	5350	5350	5350
Fan G	5600	4960	4960	4960
Shaft C	2210	2850	5350[1]	2480
Shaft D	2210	5350[1]	2856	2480
Shaft E	4420	2500	2500	4960
Shaft F	5500	7460	7460	0
Shaft G	5600	4960	4960	4960

1 T-box maximum steady state power

Table 2.3-4 - STEADY STATE POWER LEVELS AND TIME AT EACH OPERATING CONDITION (DESIGN POINT)

OPERATING CONDITION	SHAFT *	POWER (SHP)	% TIME
1) STO-Ground Roll & Lift Off			
a) Initial Roll	C	2210	
	D	2210	3
	E	4420	
	F	5420	
	G	1000	
b) Lift Off	C	50	
	D	50	3
	E	100	
	F	5500	
	G	5600	
c) Transition	C	947	
	D	048	4
	E	1894	
	F	4594	
	G	2700	
2) High Power Climb	C	2030	
	D	2030	
$V > 1.2\ V_S$	E	4062	5
	F	4062	
	G	0	
3) Cruise	C	937	
	D	937	
a) @ 10,000 Ft, M = .7	E	1875	10
	F	1875	
	G	0	
b) Loiter	C	500	
	D	500	
	E	1000	5
	F	1000	
	G	0	

* Ref. Fig. 2.3-1

TABLE 2.3-4 (continued)

OPERATING CONDITION	SHAFT	POWER	% TIME
4) Approach (1.2 V_s to 40 knots)			
a) $V = 1.2\ V_s$	C	970	
	D	970	
	E	1940	13
	F	2940	
	G	1000	
b) V = 40 Kt	C	50	
	D	50	
	E	100	13
	F	5500	
	G	5600	
c) Transition	C	572	
	D	572	
	E	1143	14
	F	3843	
	G	2700	
5) Hover @ 0 Ft Std Day	C	50	
	D	50	
	E	100	20
	F	5500	
	G	5600	
6) Ground Check (Each fan will be run up in power such that the drive train will be subjected to power levels not to exceed 70 percent of design maximum).			10
7) One Engine Inoperative	C	2850	
	D	5350	
a) # 1 engine failed	E	2500	
	F	7460	
	G	4960	
b) # 3 engine failed	C	2480	
	D	2480	
	E	4960	
	F	0	
	G	4960	

2.3.2 Gearboxes

Design methods and allowables will, in general, conform to current state-of-the-art technology as defined by the Boeing Vertol Heavy Lift Helicopter (HLH) Drive System Final Report, D301-10319-1.

1) T-Box Design Data
   - a) Maximum power: @ shaft E_a See Table 2.3-2
   - b) Ratio: 1:1
   - c) Design RPM: 11,500
   - d) Max RPM: 1.2 x 11,500 = 13,800

   Means of decoupling either input shaft from the T-box shall be provided.

2) Drop-Box Design Data
   - a) Maximum power: @ any shaft See Table 2.3-2
   - b) Ratio: 1:1
   - c) Design RPM: 11,500
   - d) Max RPM: 1.2 x 11,500 = 13,800

3) Alternating Torque - 12% of steady

4) Limit Torque - 150% of steady

5) Ultimate Torque - 150% of limit

6) Crash Loads - The following ultimate load factors will be used

Forward	20 g
Vertical	20 g
Lateral	10 g

7) Gears and shafts shall be designed for unlimited fatigue life under maximum steady power (i.e.: all engines operative with maximum transient control loads).

8) Gears and shafts shall be capable of sustaining maximum abnormal loading (i.e.: one engine inoperative (OEI) and maximum transient control loads without yielding or permanent deformation.

2.3.3 <u>Gears</u>

1) Material: AMS 6265 (9310 CEVM) per BMS 7-6 or Vasco X-2 per BMS 7-223. Vasco X-2 shall be used where increased scoring allowable is necessary.

2) Allowables:

<u>Spiral Bevels</u>

Stress	
S_b = 37,000 psi	S_b = 44,000 psi
S_c = 235,000 psi	S_c = 165,000 psi
Temperature	
T_f = 500^{0}F	T_f = 300^0

3) Methods of Analysis:

- Computer Program

4) Rim thickness shall be a minimum of one tooth depth.

5) Wherever possible, gear rims and webs will be sized to avoid resonant frequencies at operating speeds. Damping rings shall be included on every main drive gear to control resonant energy at any rpm to maximum.

BEARINGS

1) Critical bearings shall be CEVM M-50 steel

2) Material factors for above materials:
   Cylindrical roller bearings - 6.0
   Ball bearings - 6.0

3) Methods of Analysis: Computer Program

4) Shaft Slopes (Maximum)
   Ball - .0010 inch/inch
   Roller - .0004 inch/inch

5) Types - Ball and cylindrical roller bearings shall be used.

6) Loading cases - consideration shall be given to maintaining gear position in reverse loading conditions. Maximum steady and OEI loading shall be considered in bearing detail design.

2.3.5 Splines

1) Stresses - tooth bearing shall not exceed the following at steady torque rating.

Fixed	12,000 psi
Working (Non-lubed)	4,000 psi
Working (Lubed)	6,000 psi

2) Length to diameter ratio shall not exceed:

Fixed	1.0
Working	0.5

3) All splines shall be half-depth, involute profile teeth.

2.3.6 Housings

1) Where cost effective, hogouts will be used in place of castings and forgings.

2) Airframe connections shall provide a one-mount-failed capability under the following conditions:

   a) Ultimate Load - Ultimate load is defined as 1.0 factor of safety on limit load.

   b) Fatigue Load - Fatigue load is defined as maximum steady power

   c) The transmission mounting system shall be capable of operating for 10 hours with any one of the four mounting points failed at fatigue load noted above.

2.3.7 Overrun Clutch

1) An overrun clutch shall be provided between the engine and the drive system.

2) The clutch rating shall equal 8174 S.H.P. at 12,000 rpm.

3) No permanent deformation or yield of shaft, springs, or cage may occur at 11,386 ft-lb (300 percent) based on a contact stress of 600,000 psi.

4) No structural failure, overturning, or slippage may occur at 17,078 ft-lb (450 percent).

2.3.8 External Shafts and Couplings

1) Shaft construction shall be aluminum tube riveted end fittings.

2) Couplings shall be multi-plate steel laminated disk type.

3) Alternating stress allowables:

In Tube: $f_s = \pm$ 2,000 psi at rivited end connection net section (Unconcentrated)

In Coupling $f_t = \pm$ 960 psi at bolt hole (unconcentrated).

4) Design shall be such as to assure less than $1/2^o$ steady misalignment thru any coupling.

5) Lateral stiffness of the drive shaft assembly shall be such that first critical speed is 15% or more above 13,800 rp. Shaft shall be analyzed as a simply supported beam, supported at the hanger bearings. Analytical and/or empirical data will be provided to demonstrate that no unfavorable dynamic coupling modes exist when the engine, engine/fan, accessories and transmission components including gears, are operated as a combined dynamic system. Non synchronous whirl modes shall be controlled by radial piloting of spline connections.

6) Hanger bearings shall be greased lubricated ball bearings with purgable housing cavity. They shall be supported from the airframe on low-spring-rate isolation mounts.

2.3.9 Lubrication System

1) Each transmission shall have an integral lubrication system.

2) The lubrication system for each transmission shall be single path (not redundant).

3) External lubrication lines may be used where cost and complexity are reduced.

4) A visual oil level indication shall be provided.

5) Lubricating oils shall be MIL-L-7808 or MIL-L-23699.

6) Design Temperatures:

Oil	300°F	Max before cooler
Air	125°F	Max at cooler inlet

Design oil temperature rise: 50°F

7) Pressure limits shall be:

- Component Proof Pressure	400 psi
- Maximum Pump Pressure (Norm. Operating Temp -65°F)	160 psi max adjustable
- Design Operating Pressure at Oil Jets	90-100 psi

8) At least two jets will be directed at each main power train bearing.

9) Sumps shall be sized for 15 seconds oil retention time.

10) Minimum jet size shall be .030 inch

11) A jet protection screen shall be incorporated before the jets and the pressure pickoff. In addition, a full-flow, single-stage filter shall be incorporated before the cooler. This filter shall have a visible condition indicator.

12) Oil heat rejection shall be calculated on the basis of maximum steady power (excluding maneuver conditions) using the following percentages:

Bevel Gears - .50% per mesh at design rating (windage)
.50% per mesh at highest continous transmitted power

Spur or Helical Gears - .3% per mesh at design rating (windage)
.3% per mesh at highest continuous transmitted power

13) The lubrication system shall provide adequate lubrication and cooling under the following conditions:
30-second operation at zero G. Continuous operation at 45^{o} nose up or down and 30 seconds in any uncoordinated maneuver.

14) As an objective, lubrication system components such as breathers, chip detectors, filters, screens and transducers will be similar to Heavy Lift Helicopter (HLH) gearbox design.

15) Gearbox oil will be used to positively lubricate drive splines.

ACCESSORY DRIVES

1) T-Box

   None

2) Drop-Box
   a) Two lube pumps (T-Box and Drop Box)
   b) Two hydraulic pumps
   c) Two oil cooled 20 KVA alternator/CSD units
   d) Oil cooler blower

3) Accessories shall be easily accessible for maintenance or removal and replacement with the transmission mounted in the airplane.

4) Failure of an accessory shall not result in replacement of transmission (i.e., accessory jamming shall not damage main transmission as spline quill will fail in shear).

2.3.11 <u>Condition Monitoring</u>

Shall include:

1) Oil temperature before cooler
2) Oil pressure before jets
3) Debris Detection
4) Filter Clogging

2.3.12 <u>Engagement Clutch</u>

1) Load - The engagement clutch will encounter load combination of 1.3 slug-ft^2 and an aerodynamic drag of 1000 S.H.P. at synchronous speed. A positive coupling will engage at synchronous speed to carry 8174 S.H.P. steady load.

2) Engagement time: 10 seconds

3) Location - the clutch shall be located as shown in Figure 1.

2.4 PERFORMANCE

2.4.1 Summary

Analyses of major components of the drive system were made to substantiate strength and fatigue life. These components included main drive gears, gear shafts and splins, main drive bearings, housing attachment lugs, and external drive shaft tubes and coupling adapters.

All components analyzed displayed a positive margin in fatigue and ultimate loading conditions.

Gear tooth analysis for bending and contact (hertz compressive) stresses and for flash temperature (scoring) indices showed all main drive gears at or below Boeing Vertol allowables.

Bearing lives were analyzed from a spectrum of loading representative of a typical mission profile. All bearings analyzed show a life expectancy appropriate to the intended usage of the aircraft drive system.

The drop box and the T-box designs (Figures 2.1-2, 2.1-3 and 2.1-4 and the drive shafting Figure 2.1-1 reflect the results of these analyses.

2.4.2 Gearing

Design loads are summarized in Table 2.4-1. Loads for which the shafts and gears were designed are based on the criteria that unlimited fatigue life is required for steady plus transient maneuver conditions with all engines operative, but that the one engine inoperative condition combined with maximum transients may be taken as a limit condition, with finite fatigue life. Design powers are therefore as shown in Table 2.4-2.

TABLE 2.4-1 - POWER REQUIREMENTS

SHAFT (See Fig. 2.3-1)	STEADY STATE POWER WATTS x 10^3 (H.P.)	RPM	FLIGHT CONDITION
C or D (Cross Shafts)	2030	11,500	Climb, AEO.
	5450	11,500	Hover, one lift/cruise engine inoperative
E (Tee to Drop Box)	4063	11,500	Climb, AEO
	4960	11,500	Hover, lift engine inoperative
F (Lift Engine to Drop Box)	5500	11,500	Hover, AEO
	7460	11,500	Hover, one lift/cruise engine inoperative
G (Drop Box to Lift Fan)	5600	11,500	Hover, AEO

TABLE 2.4-2 DRIVE SHAFT AND GEAR DESIGN CONDITIONS

SHAFT See Figure 2.3-1)	DESIGN POWER (H.P.)	LIMIT POWER (H.P.)
C or D	5450	8174
E	5450	8174
F	7460	11,190
G	8174	12,261

These design powers provide adequate margin for transient loads in both AEO (all engines operative) and OEI (one engine inoperative) conditions.

Gear stresses at design loads are summarized in Table 2.4-3. As previously noted, these stresses are at or below allowables used in current Boeing helicopter design. The method of calculation follows recognized AGMA (American Gear Manufacturers) and Gleason Gear Analysis Standards.

2.4.3 Bearings

Bearing loading is based on a pro-rated summary of twelve flight regime load conditions, combined according to the cubic mean load rule commonly used in this type of analysis. Bearings were also reviewed for the maximum load conditions imposed by OEI plus transients to assure the absence of permanent structural deformation of the bearing elements. All main drive bearings were analyzed and all exceeded the lift criterion of 500 hours L_3 (equivalent to 1200 hours L_{10}) by a substantial margin. Further design iteration would tend to reduce bearing sizes in several locations.

TABLE 2.4-3 - GEAR TOOTH STRESS AND FLASH TEMPERATURE SUMMARY

LOCATION	TYPE	NO. TEETH	DIAMETRAL PITCH	PITCH DIAMETER cm (in)	SPEED RPM	PITCH LINE VELOCITY M/S (FPM)	BENDING STRESS N/cm^2 (psi)	CONTACT STRESS N/cm^2 (psi)	FLASH TEMP ^oC (^oF)
T-Box	Spiral BEVEL	30	3.480	21.89 (8.62)	11500	131.84 (25,952)	23,855 (34,600)	157,468 (228,400)	305 (3223)
Drop Box	Double Helical	56	7.000	20.32 (8.000)	11500	122.36 (24,086)	27,647 (40,100)	98,728 (143,200)	324 (342)

1. Stresses are shown for design power
2. Allowables:

	Bending (psi)	Contact (psi)	Flash Temp (^oF)
Bevel	37,000	235,000	500
Helical	44,000	165,000	350

2.4.4 Shafting

Shaft critical speeds were determined using a derivation of the Rayleigh pin-pin critical speed expression. The calculations for critical speed (first elastic body lateral frequency) show the lowest critical to be 42% (Shaft G) above design operating speed. This meets Boeing criteria for drive shafting.

In summary, a preliminary design analysis indicates adequate strength and fatigue life for all major components when subjected to the defined loads.

2.5 Trade Studies

Trade studies of the following major component areas were made for design optimization.

1. Shaft speed and gear ratio selection
2. Reengagement clutch lubrication system
3. Drop box configuration as influenced by clutch placement
4. Reengagement clutch concepts
5. Drive system lubrication cooling concepts

2.5.1 Shaft Speed

The shaft rpm study dealt with the effects of two selected rpm's (12,000 and 8,000) on the design characteristics of the tee and drop box. It was concluded that the weight penalty associated with the lower rpm system did not compensate for less tangible advantages of reduced gear velocity and lower bearing speeds. The Model 1041-135-2A design study does not exceed the state of the art in these or other parameters.

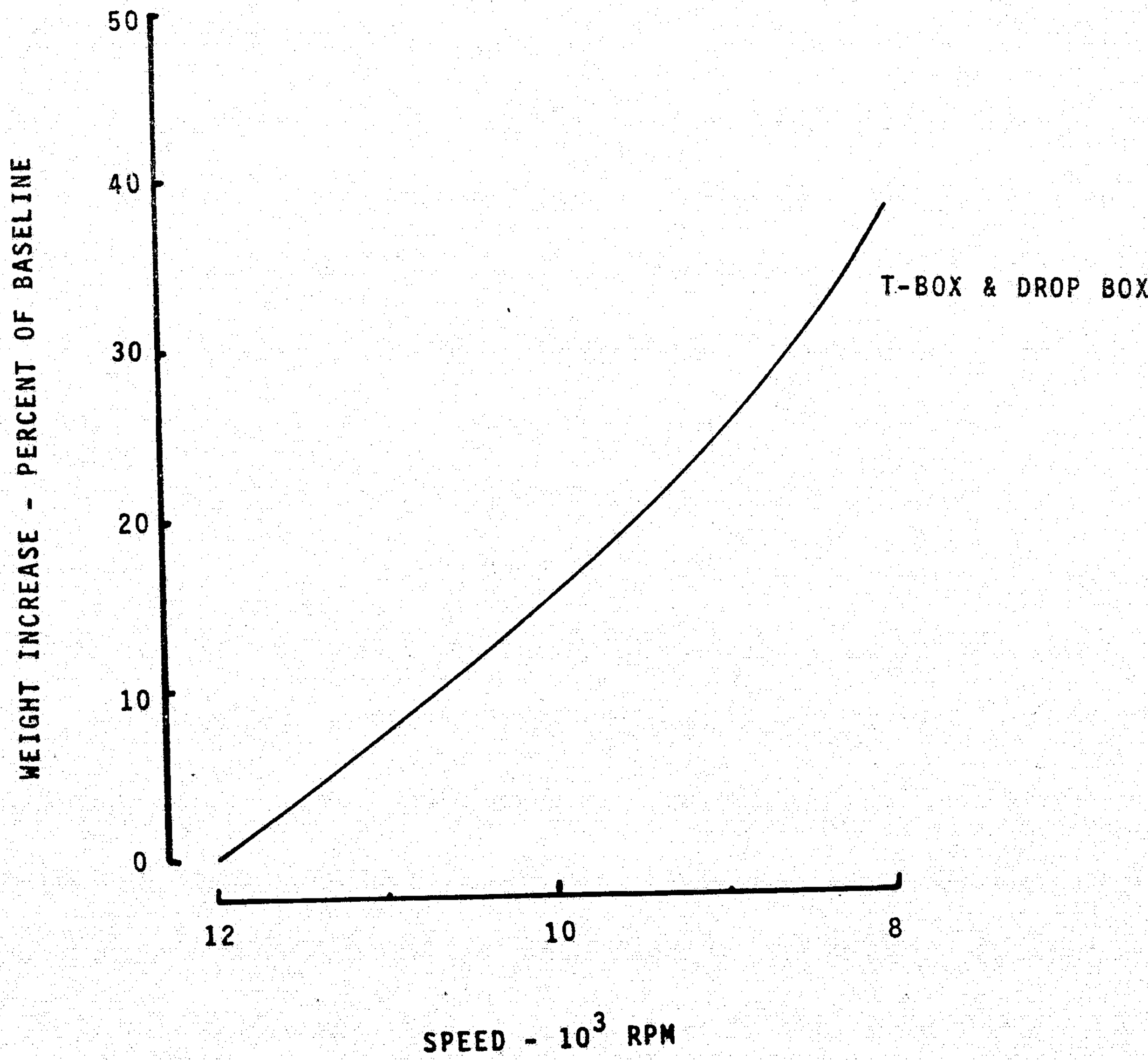
Figure 2.5-1 shows the relationship between speed and weight for the Tee and drop Box.

2.5.2 Re-engagement Clutch Lubrication System

The reengagement clutch lubrication system study evaluated separated and combined (with the drop box) fluid flow and cooling schemes. It was concluded that the design simplification and the approximate 20 lb. weight savings inherent to the combined system made this the preferred approach. Figures 2.5-2 and 2.1-10 represent the systems compared.

2.5.3 Drop Box Configuration

A study (Figure 2.5-3) was made that located engine number three, the sprag (overrunning) clutch, the bottom gear in the drop box and the forward lift fan clutch in line with each other. This results in a weight savings and also reduces power loss caused by windage in the Drop Box. However, each of the clutches require lubrication and cooling oil to be introduced into the innermost bore of the clutch assembly. To accomplish this with the clutches arranged in line would require two sets of dynamic oil transfer seals for the sprag clutch and one set for the forward clutch. It was concluded that the development of these transfer rings



2.5-1 SPEED - WEIGHT RELATIONSHIP FOR TEE AND DROP BOXES

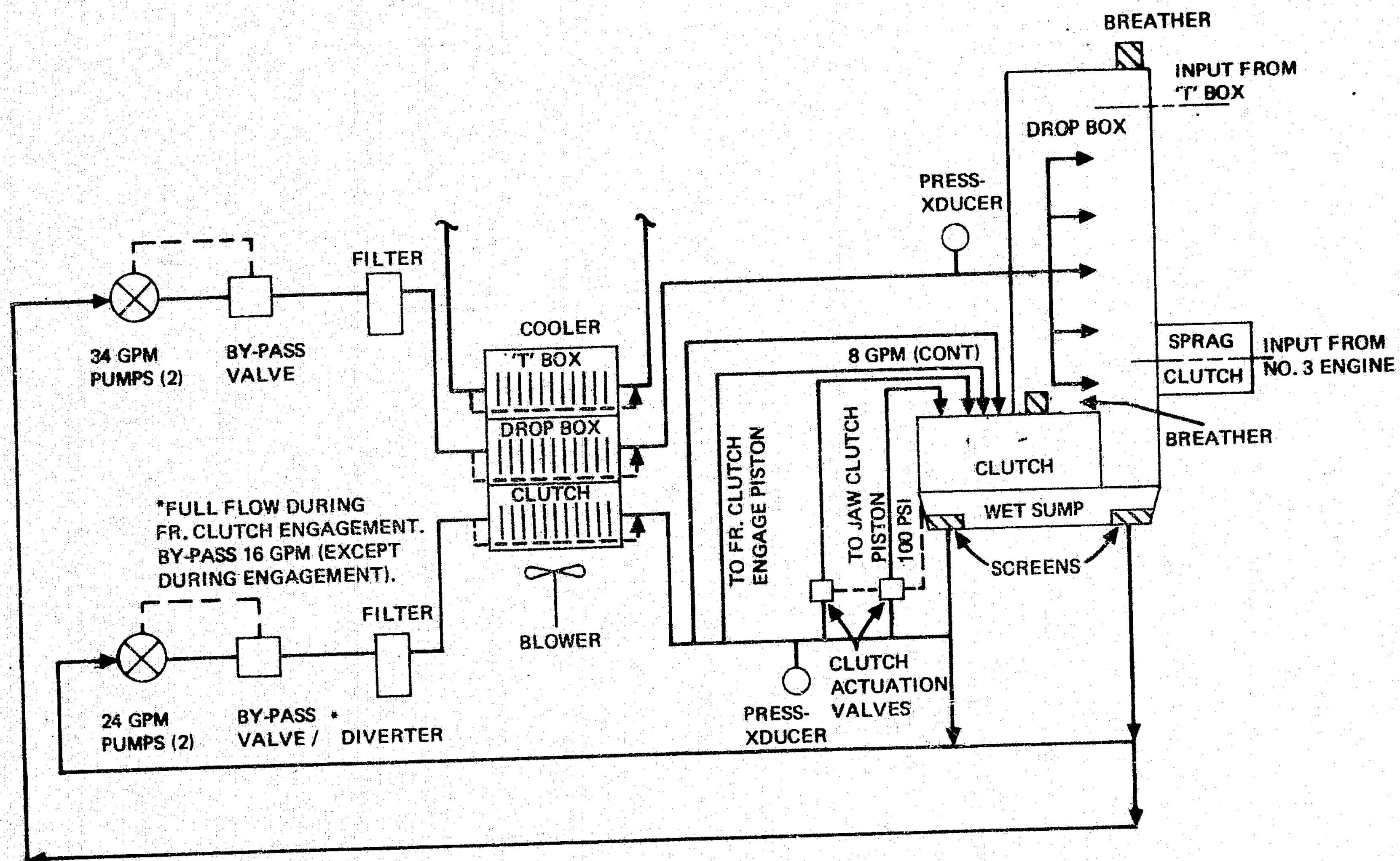
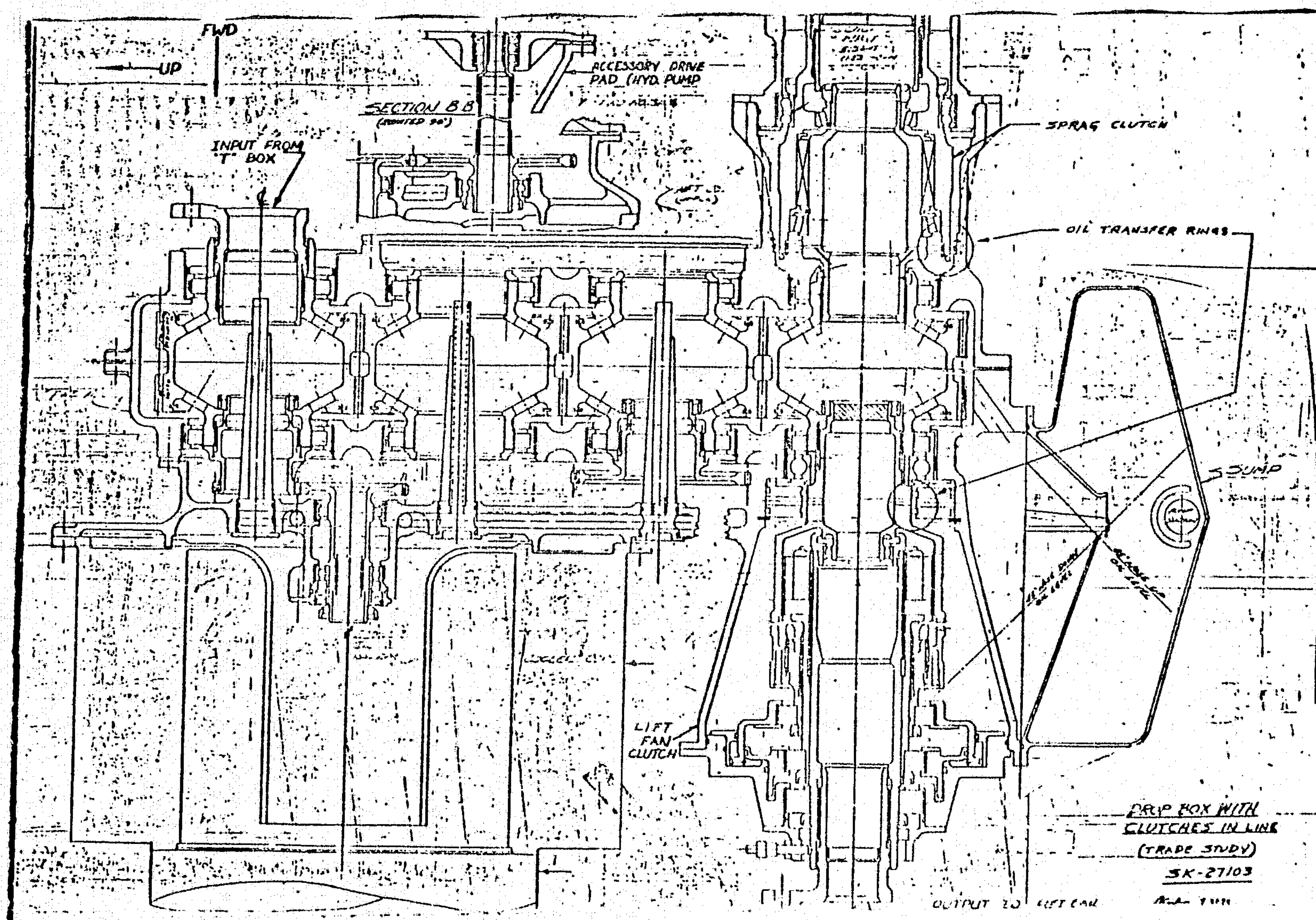


FIGURE 2.5-2 DROP BOX LUBRICATION AND CLUTCH ACTUATION - SEPARATE OIL SUPPLY



2.5-3 DROP BOX CONFIGURATION TRADE STUDY - CLUTCHES IN LINE could be a sizeable technical program. Therefore, the baseline system proposed here has clutches on independent shafts and there is direct access to the bores of the clutches for introduction of oil (see Figure 2.1-4). This is considered the minimum risk and cost approach.

2.5.4 Clutch Concept Evaluation

The various clutch concepts considered for this application include hydroviscous (wet friction) dry friction, fluid coupling and air turbine. A jaw or spline clutch is used in each case for positive engagement to transmit power with the fan under load. The jaw clutch is essential to prevent disengagement of the clutch with failure of the actuation medium.

Nearly all of these concepts have been used, generally under lower power requirements on airplane engine start systems or secondary power drive systems. The hydroviscous clutch was chosen for this application for the following reasons:

- Positive synchronization, despite torque fluctuations in the output side.
- No slip at synchronous speeds.
- A body of experience that indicates feasibility and attainment of life goals at parameters similar to the proposed aircraft design.
- Simple and relatively inexpensive parts.
- Flexibility of basic design to handle changing torque requirements.

The wet or hydroviscous clutch concept carries torque through most of the engagement cycle by viscous shear of an oil film between adjacent plates, alternate plates being driven by the input shaft and the output shaft. The final engagement is obtained by physical contact and pressure between opposing plates. Because the effective friction coefficient is quite low, the oil being a lubricant, a large rubbing area is required to develop the required torque. This is achieved through use of a multiple disc stack. Torque is controlled by varying the pressure on the disc stack through variation in axial force. Synchronous speed is achieved by bringing the torque to a level sufficient to lock the stack. The oil which is fed through the center shaft and forced through the disc stack, largely by centrifugal force, serves as the vehicle for carrying heat from the clutch disc stack. Peak temperatures are maintained at levels below the breakdown point of the oil.

2.5.5 Lubrication and Cooling System

Analysis of power train loads for various flight conditions indicates a climb result in the highest gear box power loads with associated design oil cooling requirements as shown below. "T" box heat rejection = 3140 BTU/Min drop box plus clutch heat rejection = 6357 BTU/Min cooling system concepts using fuel, water, and air as heat sinks were evaluated to meet these requirements. The results are shown in Figure 2.5-4. Fuel flow to the engines is inadequate to meet the total power train cooling requirements without exceeding current engine fuel supply temperature (135^{o}F) as shown in Figure 2.5-5, however the "T" box could be fuel cooled.

2.6 Development Plan

The V/STOL gear and shaft system has been designed at stress levels that are equal to or less than current production designs. From a material allowables viewpoint this design is entirely within the state of the art. Because of the power requirements and other specific considerations, there are, however, areas where development and testing is required to provide a minimum risk drive system. The objective of the test program is to address these areas as early in the development cycle as possible in order to provide assurance that the aircraft will have a reliable and safe drive system.

Areas that will be investigated will include:

2.6.1 Lubrication and Scavenge Systems

Peripheral speeds of gears and bearings are at the high end of experience, especially in the T box. The effect is to create centrifugal forces and windage that distorts the flow of oil to the gears and bear-

V/STOL GEARBOX COOLING CONCEPTS

SYSTEM CONCEPT	CAPABILITY	WEIGHT	COMMENTS
FUEL/OIL HEAT EXCHANGE	5700 BTU/MIN WITH 50° FUEL 3000 BTU/MIN WITH 90°F FUEL	40 LBS	FUEL TEMPERATURE SUPPLIED TO ENGINE LIMITED TO 135°F FOR ENGINE OIL COOLING (1250 BTU/MIN)
HLH AIR/OIL HEAT EXCHANGE AND BLOWER	8320 BTU MIN. MEETS REQUIREMENT	89 LBS EXCLUDES: o EXCHANGE DUCTING o DRIVE BOX DESIGN o INSTALLATION	HX AND BLOWER CAPABLE OF PROVIDING TOTAL SYSTEM COOLING. FAN POWER 35 HP
WATER/OIL HEAT EXCHANGE	MEETS REQUIREMENT	150 LBS FOR STOL TESTING (30 MIN) 433 LBS CRUISE (120 MIN)	ASSUMES T BOX FUEL COOLED BASED ON MAX. HEAT REJECTION SIMPLE INSTALLATION

FIGURE 2.5-4

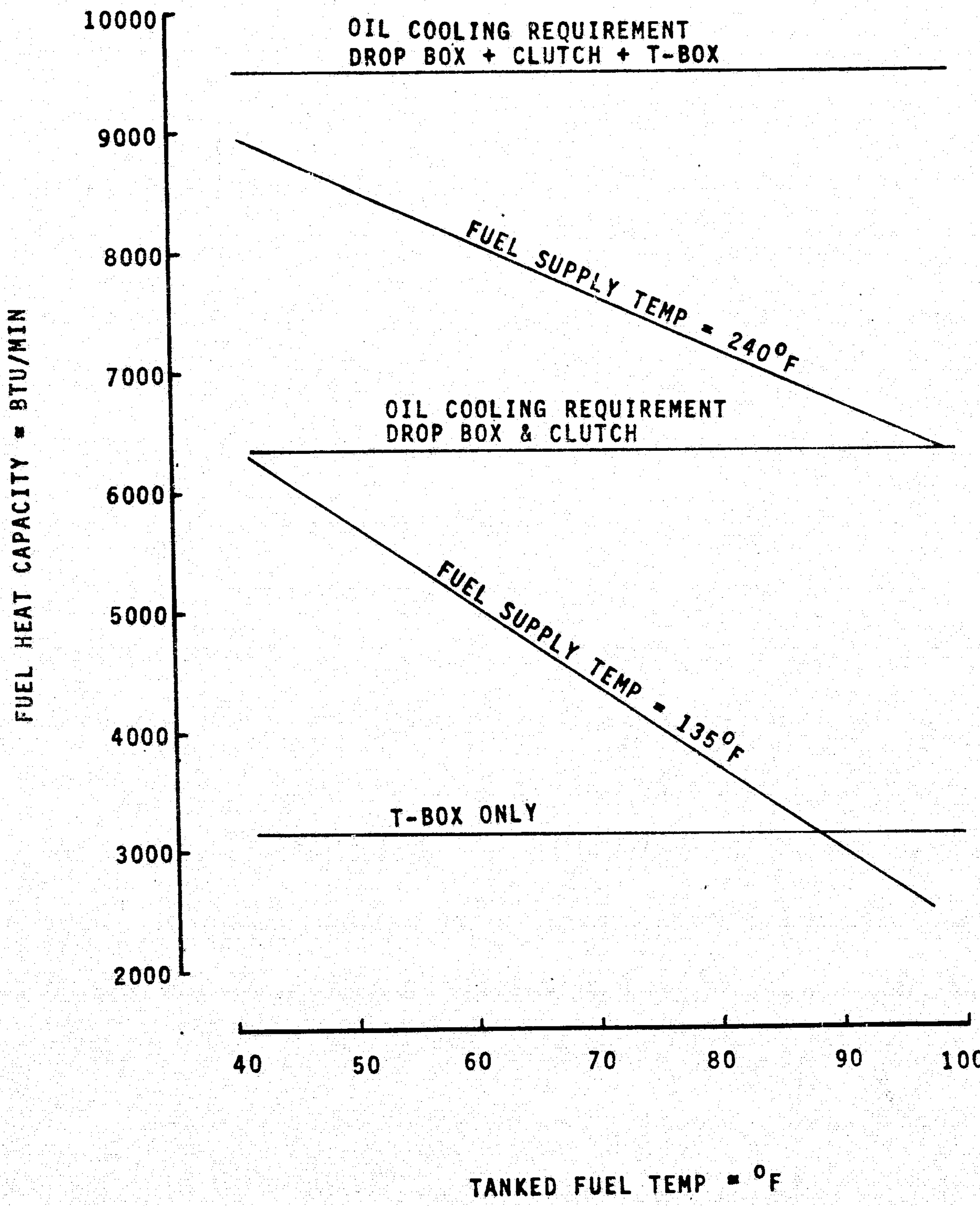


FIGURE 2.5-5 GEAR BOX OIL COOLING USING FUEL HEAT SINK

ings, and may also impede the return oil flow to the scavenge pickup points. These effects are particularly noticeable in a compact gearbox with a shallow oil collection area like the T box. Since the actual oil paths within the operating gearbox are experimentally determined a necessary development requirement is to provide baffles to minimize the windage effects, and to adjust jet size dna location to provide optimum lubrication to the working areas. Since the effects are primarily speed dependent, rather than load dependent, a considerable amount of development can be performed on simple test rigs that rotate the elements at full rotational speed but do not attempt to load the elements. The objective of these tests are to reduce windage and oil churning, and hence temperature rise through the box by developing effective baffle designs.

2.6.2 Gear Tooth Load Distribution

Spiral bevel gears, such as are used in the T box, are adjusted for optimum tooth load distribution by changes in mounting distance. The load distribution is affected also by the deflection of the gearbox and by the clearances of the bearings. Therefore, spiral bevel load distribution is commonly adjusted by loading the gears in the actual housing and observing the load distribution. The adjustment of load distribution in the initial development is by modification of gear tooth profile by regrinding. After this, subsequent gears are produced to the same profile and adjusted by shimming in or out of mesh. The actual distribution of load has, in the past, been estimated from visual observation of the load footprint. Our current practice is to place strain gages along the gear tooth and measure strains, and hence load distribution directly, rather than to infer the distribution from contact patterns. For this purpose a full torque test rig capable of slow turning through a fraction of a revolution is required.

Although helical gears are not adjustable in their mountings as bevel gears are, a very similar concern exists in load distribution. Due to torsional and other deflections, the load may vary along the face width, and between left and right helicals in a double helical gear. Adjustment is accomplished by grinding lead corrections along the gear tooth until the desired load distribution is achieved. The same type of test rig is required as for the bevel box, with suitable changes to accommodate shaft orientations.

Both types of gearing have been evaluated by Boeing in the recent past. Figure 2.6-1 illustrates a torsion fixture built and used for the HLH bevel gearing, and Figure 2.6-2 illustrates a fixture used for similar load distribution analysis of the PHM (Patrol-Hydrofoil-Missile) main propulsion gearbox, which used double helical gearing.

2.6.3 Resonant Frequency Control

A potentially destructive failure mode can originate in uncontrolled (undamped) resonance of gear rims when excited by gear tooth meshing frequencies. This is particularly likely to occur when the frequency of meshing is high, and when the gears are of moderate to large diameter. The possibilities of overlap between forcing frequencies and resonant frequencies is increased as compared to lower speed operation.

An analytical method is used as a predictive tool in the initial design of the gear blank. Given the gear geometry, this computerized method is able to define resonant frequencies within a few percent, and is also capable of defining mode shapes. The gear blank can thus be designed so that the resonant frequencies of significance do not fall on prime operating speeds. Necessarily, though there will be operating conditions where forcing frequencies coincide with resonances. To control the energy output at these points, damping rings are installed in grooves in the gear rim. These rings prevent destructive build-up of resonant energy.

The location and shape of the predicted resonant modes are verified by experimental methods. The gear is forced by electromagnetic or air blast methods and the response amplitude is measured. Figure 2.6-3 illustrates this method being applied to an HLH gear. The existing experimental equipment is sufficiently flexible to allow use of its application to the V/STOL gearing, and will be used for T and drop box verification.

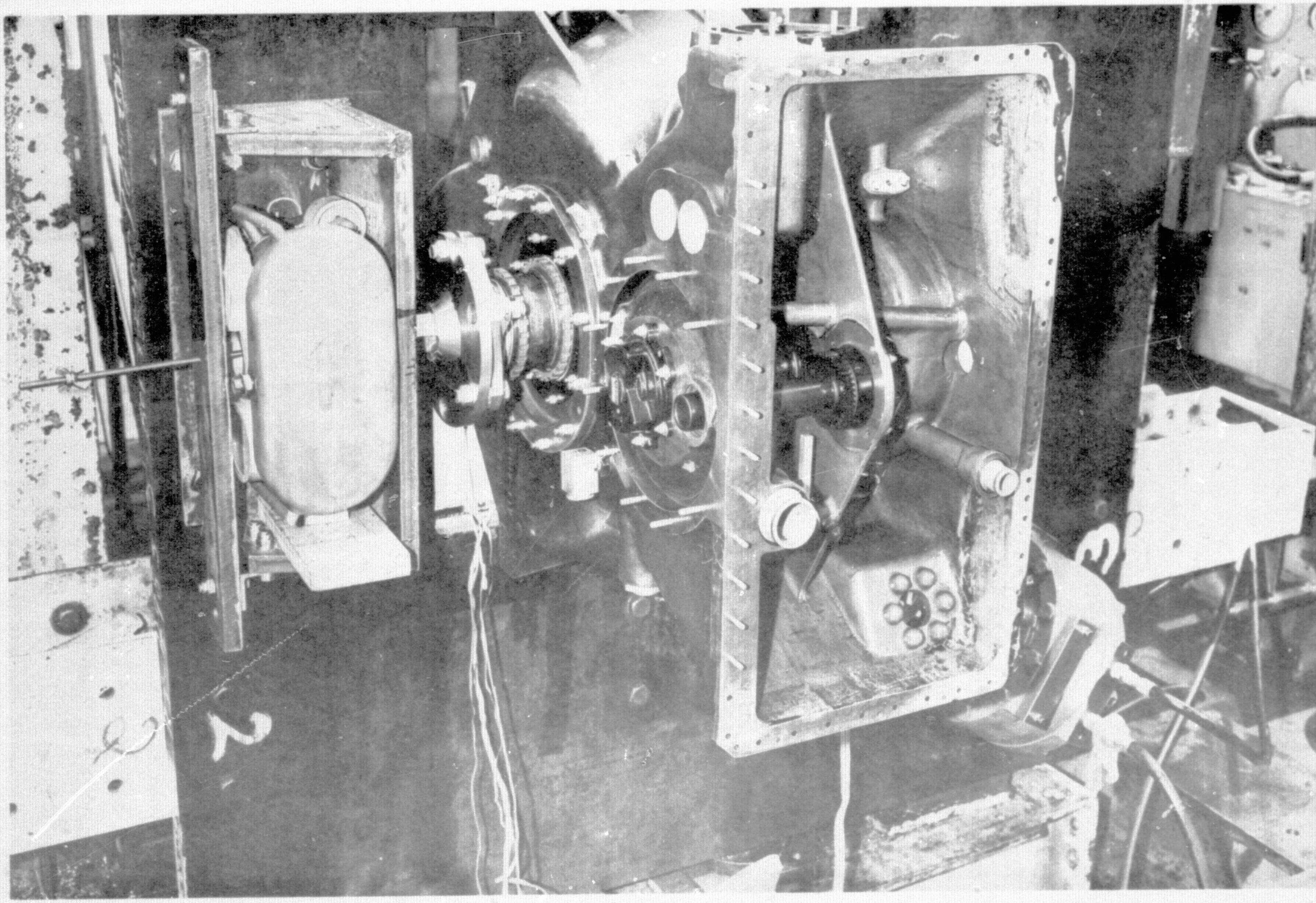


FIGURE 2.6-1 HLH BEVEL GEARING TORSIONAL FIXTURE

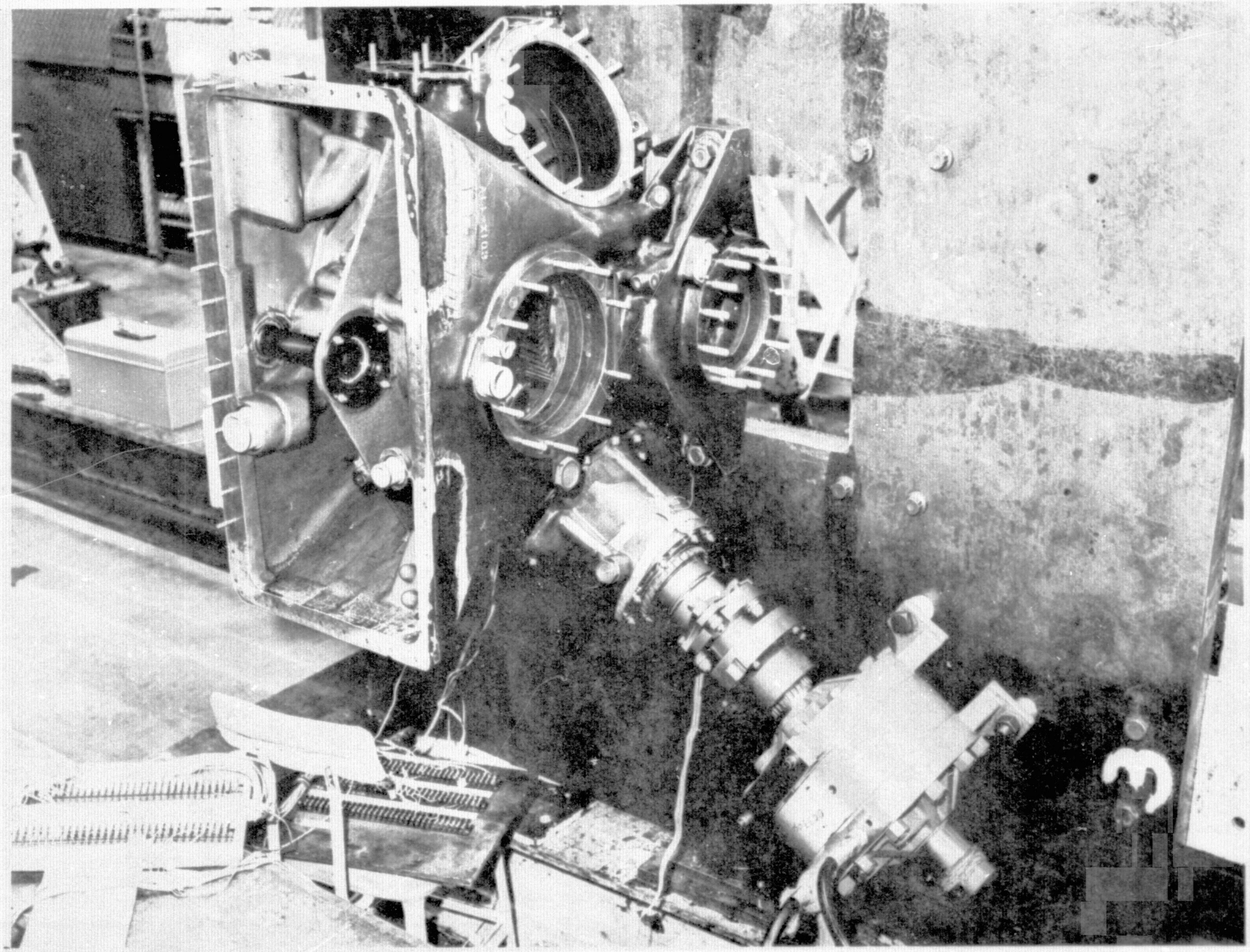


FIGURE 2.6-2 PHM BEVEL GEARING TORSIONAL FIXTURE

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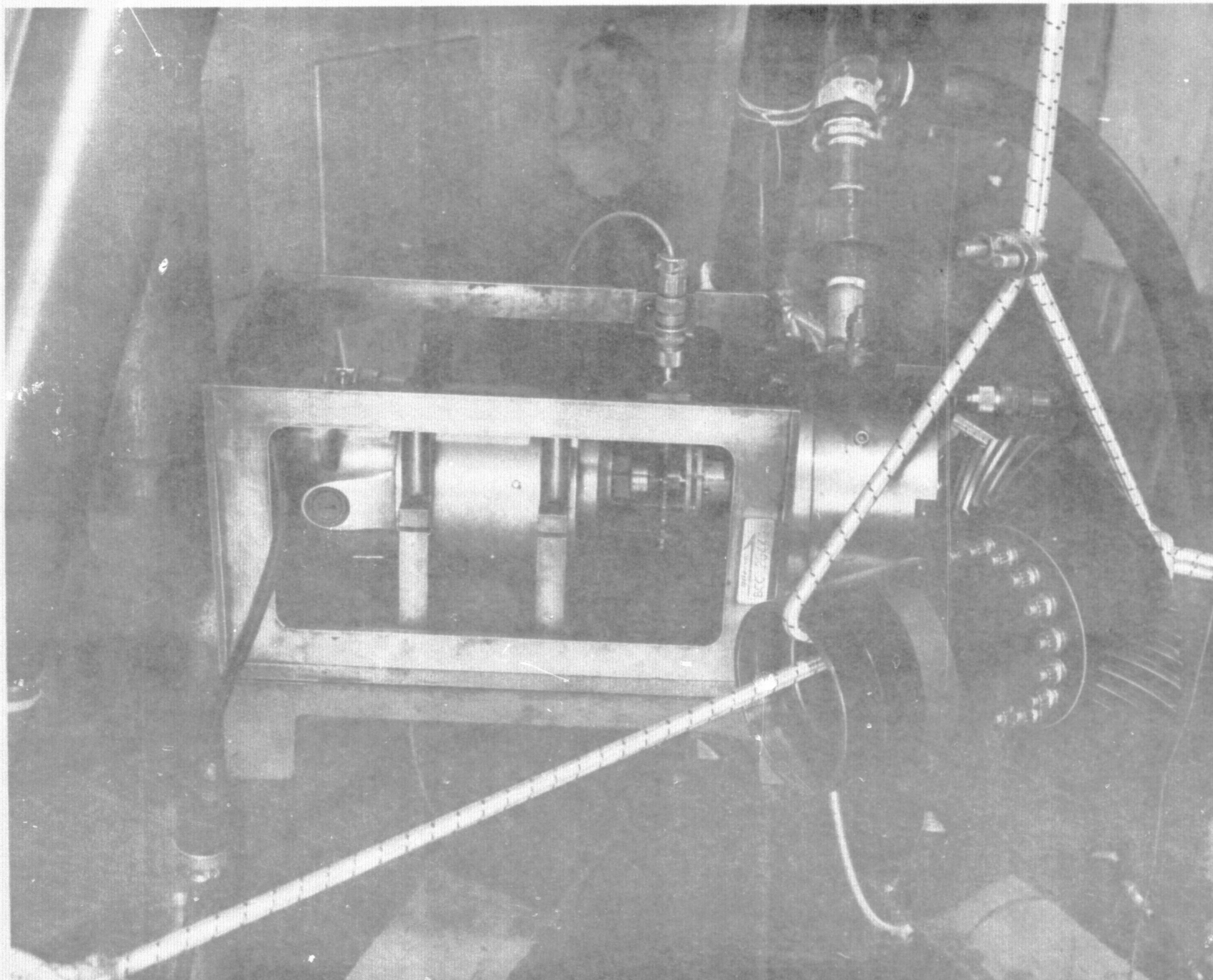

FIGURE 2.6-3 PREDICTED RESPONSE MODES TESTING

2.6.4 Verification of Design Integrity

The final state of development is realized as the gear box is run as a system, interfacing with the airframe, the propulsion system, drive shafting, lubrication and cooling and accessory drives. The fullest simulation of interfaces is desirable. Therefore, the V/STOL development plan includes construction of a full-scale test bed on which are mounted the complete power generation, transfer and absorption systems. The airframe structure is duplicated at all points where system loads are carried out. Full lengths of drive shafting connect the various components. Accessories are loaded to design powers. Oil cooler openings and ducts are duplicated to provide realistic airflows and component temperatures.

The particular types of tests that are run in this rig, and their objectives, are as follows:

> The gear tooth load distribution tests measure tooth stresses statically as they are rolled slowly through mesh. The effects of dynamic loading (centrifugal, tooth profile inaccuracy, resonant and thermally induced stresses) are determined by strain surveys of the gears as they operate at full speed and up to full torque in the box. The techniques for success have been developed by Boeing in the course of developing the HLH and UTTAS gear boxex. Necessary elements of this technique include gage placement to close limits, gage bonding to resist oil and centrifugal effects, readout from high-speed shafting by telemetry, and interpretation of the data. Figures 2.6-4 and -5 illustrate application of strain survey techniques to past Boeing programs. Figure 2.1-7 illustrates the instrumentation used in these surveys.
>
> Lubrication system evaluation to determine oil temperatures, pressures and flows in a realistic environment, with the transmissions connected to aircraft oil-air heat exchangers, is to verify the adequacy of the lubrication when the working surfaces are loaded to design levels. Tear-down inspections determine the condition of components; measuring wear, checking for overheating and surface finish.

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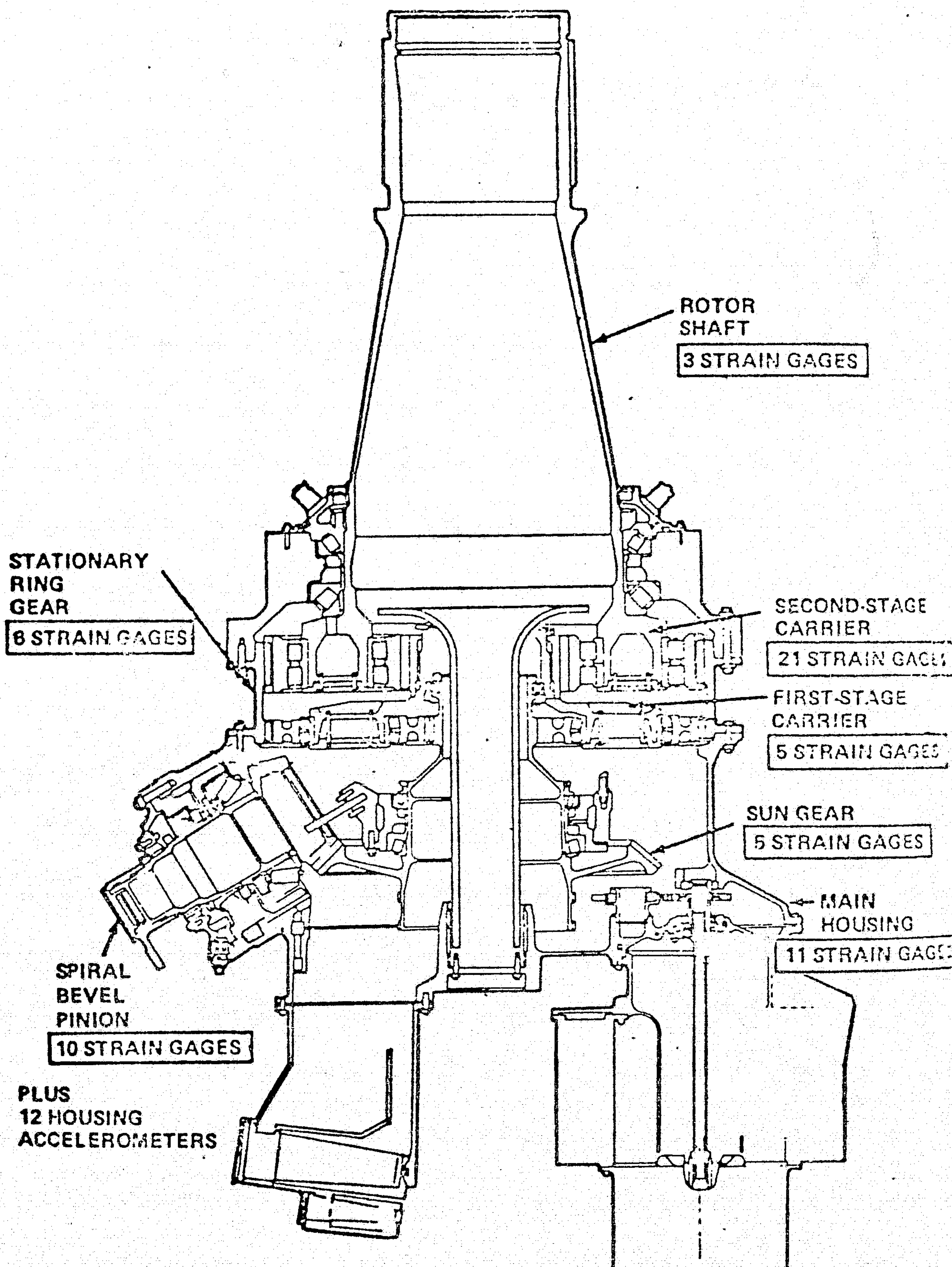


FIGURE 2.6-4 Aft Transmission Strain Survey Instrumentation.

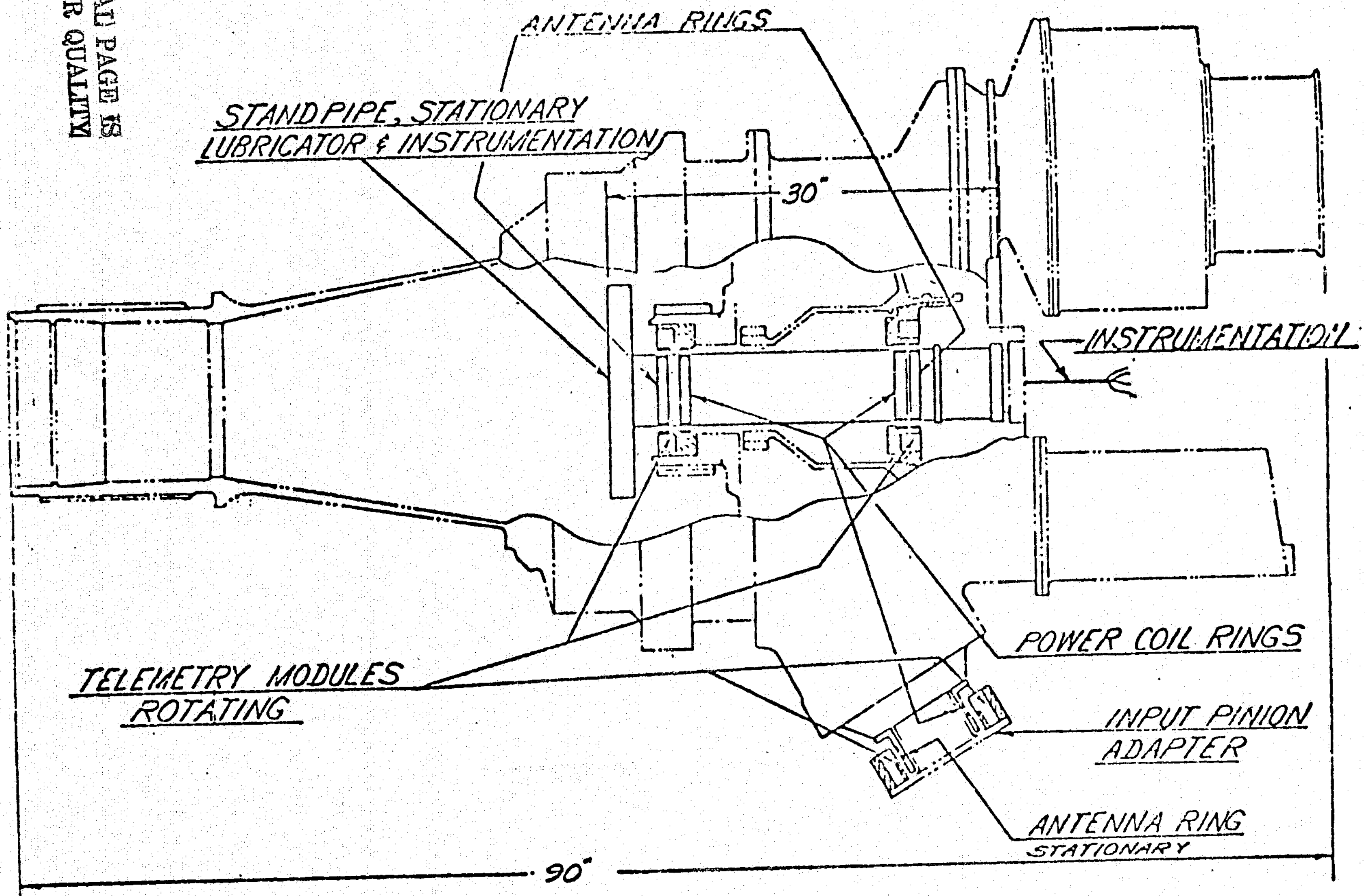


FIGURE 2.6-5 Aft Transmission Instrumentation Assembly.

Shaft critical speed evaluation will verify the absence of whirl modes within the operating spectrum. The system under test is operated at the maximum overspeed condition (usually at or above 110% of design operating speed) and acceleration at the transmission shaft inputs are recorded. Before this, a bang test of non-rotating exposed shafts is made to verify the calculated frequency.

Component Fatigue Testing is conducted at maximum conditions of power and speed through a number of cycles beyond the knee of the S-N curve to establish that components are operating below their endurance limit. Because the number of full-scale components that can be so tested is limited to a few specimens, added confidence is gained by testing at overloads. In the proposed test rig, the maximum overload capability per transmission will be determined by the power absorber, rather than engine, capability.

Determination of bearing life, as calculated by classical fatigue life calculations, is not an appropriate objective in a test that extends to only a small portion of the median design life. Unlike gears and shafting, whose design life is unlimited, bearings have a finite surface fatigue life. Bearings therefore operate above their endurance limit, and a valid test must extend for several calculated life times. Since surface fatigue damage is progressive, and is detectable by debris indicators, the possibility of such damage is generally accepted, and test programs for bearing life evaluation are reserved for reliability type testing of many thousands of hours.

Damage other than surface fatigue is not accepted, however, and is to be investigated during the proposed wear testing.

Wear determination is an objective of the test program. Test program times and loads are sized with the purpose of identifying and correcting sources of wear, which may include spline wear, fretting of faying surfaces (bearing race to shaft, gear mountings), roller end wear, cage wear, overrun clutch sprag wear during the overrun mode. The presence of these conditions are made known during intermediate teardown inspections thru the test cycle.

The length of the test program is basically sized by the need to develop wear life determinations sufficient to assure that the life of the transmission meets or exceeds the design goals.

Low-load testing also is included in the test agenda to investigate effect on transmission elements. Areas of sensitivity include bearings, where low loads may cause skidding damage to races and rollers, and drive shaft connections where low loads relax the centering forces and may cause whirl modes that are lower in speed than the calculated elastic body criticals. A further example of low-load testing is in the overrun mode, where sprag clutch wear is investigated.

REFERENCES

1. North American Aviation, Inc. Report NA-63-1347, "Actual Weight and Balance Report for a Twin Engine Turbojet Utility Trainer, NAA Designation NA-276, Air Force Designation T-39A, Contract AF33(657)-7272", dated October 30, 1963.

2. NASA CR-137749, Design Definition Study of a Lift/Cruise Fan/Technology V/STOL Airplane - Summary, Contract NAS2-6563, dated August 15, 1976

3. Ebers, R. S., and Sandlin, H. H., "V/STOL Landing Impact Criteria", AFFDL-TR-68-96, May 1969.

APPENDIX "A"

STATEMENT OF WORK

Introduction - The objectives of the follow-on studies are to refine the modified T-39 Sabreliner research and technology conceptual design for a truly low cost aircraft and to conduct sufficient design effort on the power transmission and control systems to identify the technical risks and to obtain a more detailed estimate of the design, fabrication, and testing costs. Some revisions to the guidelines and criteria have been made for clarification and to assist in obtaining a lower cost without substantially reducing the research and technology demonstration capability of the aircraft. The total emphasis of this study will be on two full mission modified T-39 aircraft. Therefore, component and system designs should not be based on what should be developed for an operational lift/cruise fan aircraft. Operational aircraft consideration need not be totally ignored, but obtaining a design that can be developed for minimum cost and perform the research and technology demonstrator missions with safety should be given first priority.

Some areas of the propulsion and control systems are considered possible high risk in that it is not known if the development of some components and assemblies are within the state-of-the-art when considering a low cost approach. Components with risk need to be identified and studies of the designs accomplished in sufficient depth to determine methods for development that are within the cost and schedule of the estimates.

The study will include three tasks;

A. Task I will include a refinement of the research and technology aircraft design for low cost and risk.

B. Task II will include design details, analysis, and development methods and costs for propulsion system components with high risk.

C. Task III will include an iteration of the Task I design to determine the reduction or change in research and technology demonstration capability of the aircraft when the cost of the Task I aircraft is reduced 20-percent.

D. R&QA provisions commensurate with the Statement of Work requirements for low cost, low risk, and minimum testing will be reflected in the design.

Statement of Work

The contractor shall provide all material, facilities, and services as required to complete the tasks outlined herein. The contractor shall be responsible for communicating with the appropriate engine manufacturers to obtain propulsion system data. These communications will be forwarded to the contract technical monitor.

Task I - Modified T-39 Conceptual Design Refinement

The contractor shall refine the modified T-39 research and technology aircraft conceptual design, 1041-135-2, that was prepared under Contract NAS2-6563 Mod 12 and reported in NASA Contractor Reports CR 137749 and CR 137750, to incorporate changes which will minimize cost and risk. The Design Guidelines and Criteria for Design Definition Study of a Lift Cruise Fan Technology V/STOL Aircraft, Attachment I to this Statement of Work, have been revised. Where the guidelines are too restrictive for obtaining a low cost, low risk aircraft design, the tradeoff possibilities shall be analyzed and documented. Areas that shall be considered for cost reduction shall include, but not be limited to the following.

1. Experimental shop approach shall be used in the design and fabrication of the technology aircraft.
2. Control System - The control system shall be designed with minimum sophistication for the initial flights assuming the VTOL and STOL missions in the guidelines and shipboard operation will be flown by highly qualified test pilots. The definition of an optimum control system for a lift cruise fan aircraft will not be a part of this study but will be the subject of future advanced research effort. The system shall incorporate, where possible, available off-the-shelf hardware using all or portions of developed systems and software.

3. Testing - Development testing shall be minimized assuming frequent inspections with minor repairs and total life commensurate with the low cost technology aircraft approach. Maximum use of system test in the aircraft may eliminate expensive test rigs.

4. Number of engines - In light of the revised guidelines with regard to the engine out hover capability and to achieve a low cost design, the contractor shall examine a three engine and a two engine configuration. For the shafted transmission designs the contractor shall investigate the use of two PD 370-16 engines, three XT-701 engines or two XT-701 engines plus one turbo-shaft engine that will produce a cost effective configuration. These engines will drive three variable pitch fans. If the guidelines performance cannot be met with the two engine configuration at a lower cost and with allowances made for reasonable weight growth which may occur in a minimum cost program, Task I shall be pursued with a three engine configuration and Task II shall be pursued if possible with the two engine configuration.

5. Maximum use of existing components - Examine the maximum use of the original airframe, cockpit, control panels, etc., as well as existing hardware components from other aircraft such as the F-101 empennage to determine if their use will provide a reduction in aircraft cost.

6. Avionics system - A minimum avionics system using existing components where suitable will be employed in the technology aircraft. Contractor's recommendation in this area will be documented.

7. Ejection seat - The selection of the ejection seat and the extent of testing will be examined for low cost. The use of cockpit and ejection seats from other aircraft will be examined.

Task II - Design Detail of Propulsion System Components

The contractor shall prepare design details and analysis of components of the power transmission system which are considered to have relatively high risk. The study shall include, but not be limited to the "T" gear box, bevel gear box, and engagement clutch. Components of the thrust vectoring system shall be considered where there is possible risk in development time, cost, or performance. The components that will be studied in detail shall be submitted to NASA for approval during the first month of this effort. The design detail effort and analysis shall be in sufficient depth to show that the components can be designed, fabricated, tested, and qualified for flight within the state-of-the-art assuming a low cost approach. The contractor shall prepare a design, fabrication, and test schedule and estimate the costs for each of the items addressed in this task.

Task III - Reduced Cost Aircraft

The contractor shall prepare a variation of the Task I conceptual design configuration which has a cost reduction of 20 percent below that estimated for the Task I aircraft design without an increase in risk. The reduction in any research and technology demonstration capability, performance limitations, design guidelines compromises, etc., to achieve the low cost shall be investigated. The Task I design changes and their associated costs shall be itemized .

ATTACHMENT 1: DESIGN GUIDELINES AND CRITERIA FOR DESIGN DEFINITION STUDY OF A LIFT/CRUISE FAN TECHNOLOGY V/STOL AIRCRAFT

The purpose of these guidelines is to provide a basis for comparing the conceptual designs of V/STOL Technology aircraft using the lift/cruise fan propulsion system. These guidelines will provide direction for only those items required for conceptual design considerations. This is not an attempt to provide criteria for either the preliminary or detail design of military aircraft.

Except where specific criteria are given, handling qualities shall be consistent with the intent of AGARD-R-577-70 and MIL-F-83300. Under MIL-F-83300, the aircraft will be considered in the Class II category. Two levels of operation will be considered. Level I is normal operation with no failures. Level 2 is operation with a single reasonable failure of the propulsion or control system.

Upon any reasonable failure of a power plant or in the control system, the aircraft shall be capable of completing a STOL flight mode takeoff and continuing sustained flight. With failure of the most critical power plant, Level 2 performance shall be achieved at sea level and at 90°F under the following conditions: (a) STOL Mode - capability for continuing flight on a flight path 1 1/2° above the horizontal at a weight which shall include 2500 lbs. payload and fuel sufficient for 11 STOL test missions; (b) VTOL Mode - capability for a thrust to weight ratio of 1.03 without altitude control at a weight which shall include 2500 lbs. payload and fuel sufficient for 2 VTOL test missions. Fan failure during low speed flight is not a design requirement (as similarly the case for rotor type or propeller-driven concepts), although consideration of a turbo-engine failure is a design requirement.

1.0 Flight Safety and Operating Criteria

1.1 Handling Qualities Criteria (low speed powered lift mode)

Definitions of the two levels are as follows:

Level 1: Flying qualities are satisfactory for research and technology demonstration missions when flown by and engineering test pilot.

Level 2: Flying qualities are adequate to continue flight and land. The pilot work load is increased but is still within the capabilities of an engineering test pilot.

1.1.1 Attitude Control Power (S.L., 90^{0}F).

Applicable for all aircraft weights and at any speed up to V_{con}. For purposes of this study, the VTOL values will apply near hover (0 to 40 kts); whereas the STOL values will apply when operating above 40 knots. The tables list minimum values, higher levels are desirable for research purposes.

Level 1: The low speed control power shall be sufficient to satisfy the most critical of the three following sets of conditions:

Conditions (a) -- to be satisfied simultaneously,

(1) Trim with the most critical CG position.

(2) In each control channel provide control power, for maneuver only, equal to the most critical of the requirements given in the following table.

Axis	$\frac{\text{Maximum Control Moment}}{\text{Inertia}}$		Attitude Angle in 1 sec after a Step Input	
	VTOL	STOL	VTOL	STOL
Roll	± 0.9 rad/sec^2	± 0.6 rad/sec^2	± 15 deg	$\pm$ 10 deg
Pitch	± 0.5 rad/sec^2	± 0.4 rad/sec^2	$\pm$ 8 deg	$\pm$ 6 deg
Yaw	$\pm$ 0.3 rad/sec^2	± 0.2 rad/sec^2	$\pm$ 5 deg	$\pm$ 3 deg

These maneuver control powers are applied so that 100% of the most critical and 30% of each of the remaining two need occur simultaneously.

Condition (b) -- At least 50% of the above control power shall be available for maneuvering, after the aircraft is trimmed in a 25 knot crosswind.

Condition (c) -- At least 90% of the control power specified in condition (a) shall be available after compensation of the gyroscopic moments due to the maneuvers specified in condition (a). This condition includes trim with the most critical CG position.

Level 2: The low speed control power shall be sufficient to satisfy, simultaneously, the following:

(1) With the most critical CG position trim after any reasonable single failure of power plant or control system.

(2) In each control channel, provide control power, for maneuver only, equal to at least the following:

Axis	$\frac{\text{Maximum Control Moment}}{\text{Inertia}}$		Attitude Angle in 1 sec after a Step Input	
	VTOL	STOL	VTOL	STOL
Roll	$\pm$ 0.4 rad/sec^2	$\pm$0.3 rad/sec^2	$\pm$ 7 deg	$\pm$ 5 deg
Pitch	$\pm$ 0.3 rad/sec^2	$\pm$0.3 rad/sec^2	$\pm$ 5 deg	$\pm$ 5 deg
Yaw	$\pm$ 0.2 rad/sec^2	$\pm$ 0.15 rad/sec^2	$\pm$ 3 deg	$\pm$ 2 deg

Simultaneous maneuver control power need not be greater than 100% - 30% - 30%.

1.1.2 Flight Path Control Power (SL to 1000 ft., 90^{0}F).

1.1.2.1 VTOL (0 - 40 kt TAS and zero rate of descent)

At applicable aircraft weights and at the conditions for 50% of the maximum attitude control power of critical axis specified in para. 1.1.1 it shall be possible to produce the following incremental accelerations for height control:

Level 1: (a) In free air $\pm$ 0.1 g

(b) With wheels just clear of the ground -0.10g, +0.05g

Level 2: (a) In free air -0.1g, +0.05g

(b) With wheels just clear of the ground -0.10g, +0.00g

It shall also be possible to produce the following horizontal incremental acceleration, but not simultaneously with height control.

Level 1: $\pm$ 0.15 g

Level 2: $\pm$ 0.10g

At applicable aircraft weights it shall be possible to produce the following stabilized thrust-weight ratios without attitude control inputs.

Level 1: $\frac{F}{W}$ = 1.05 in free air (Takeoff power rating)

Level 2: $\frac{F}{W}$ = 1.03 in free air (Emergency power rating)

With the most critical engine failed, Level 2 performance shall be achieved at a weight which shall include 2500 lbs. payload and fuel sufficient for 2 VTOL test missions (Figure 1a).

1.1.2.2 VTOL and STOL Approach (40 kts. to V_{CON})

At the applicable landing weight the aircraft shall be capable of making an approach at 1000 FPM rate of descent while simultaneously decelerating at 0.08g along the flight path.

It shall be possible to produce the following incremental manual acceleratio by rotation alone (angle of attach change and constant thrust) in less than 1.5 seconds at the STOL landing approach airspeed where reasonable rotation (angle of attack changes) will produce at least 0.15 g's.

Level 1: $\pm 0.1g$

Level 2: $\pm 0.05g$

It shall be possible to produce the following normal accelerations in at least 0.5 seconds for flight path, flare, or touchdown control by either thrust changes or combined thrust changes and rotation at STOL landing approach speeds below which 0.15g's can be produced by reasonable rotation alone.

Level 1: $\pm 0.1g$

Level 2: $\pm 0.05g$

1.1.3 VTOL and STOL Low Speed Control System Lags (S.L. to 1000 ft. 90^0).

The effective time constant (time to 63% of the final value) for attitude control moments and for flight path control forces shall not exceed the levels given in the following table.

	Level 1	Level 2
Attitude Control Moments	0.2 sec	0.3 sec
Flight Path Control Forces	0.3 sec	0.5 sec

With a step-type input at the pilot's control the commanded control moment or force shall be applied within the following:

Level 1: 0.3 seconds for 0.5 inches of pilot's control
0.5 seconds for full pilot's control

Level 2: 0.5 seconds for full pilot's control

1.1.4 Stability (S.L. to 1000 ft., 90^0F)

1.1.4.1 Hovering

The frequency and damping of the airframe/control system dynamics, in the hovering condition, shall be within the following limits for the three rotary axes:

Level 1: Optimum damping and frequency zone established from the Ames six-degree-of-freedom moving base simulator (Figure 2).

Level 2: The zone given in Figure 2. The boundary of this zone corresponds to a damping factor of 0.166 for values of above 1 rad sec.

1.1.4.2 Low Speed

Level 1: The dominant oscillatory modes shall be maintained as close as possible to the optimum zone specified in section 1.1.4.1 while maintaining other oscillatory modes damped. Aperiodic modes, if unstable, shall have a time to double amplitude of greater than 20 sec.

Level 2: The dominant oscillatory modes shall be maintained within the Level 2 zone given in Figure 2. Other oscillatory modes may be unstable provided their frequency is less than 0.84 rad/sec and their time to double amplitude greater than 12 sec. Aperiodic modes, if unstable, shall have a time to double amplitude of greater than 12 sec.

1.1.4.3 Cruise

The aircraft as configured for cruise flight shall be statically stable at all gross weights with a stability margin of 0.05 at the critical center of gravity without stability augmentation.

1.2 STOL Takeoff Performance

The climbout gradient in the takeoff configuration, at takeoff gross weight, with gear down and most critical power plant failed at lift off shall be positive and the aircraft will continue to accelerate.

During takeoff wing lift shall not exceed 0.8 CL_{MAX}.
No catapults or arresting gear will be utilized. The rolling coefficient of friction will be 0.03. (For calculations).

1.3 Conversion Requirements (STOL and VTOL)

It must be possible to stop and reverse the conversion procedure quickly and safely without undue complicated operation of the powered lift controls.

The maximum speed in the powered-lift configuration shall be at least 20% greater than the power-off stall speed in the converted configuration for level 1 operation and the speed in the powered lift configuration shall be at least 10% greater than the power off stall speed for the level 2 operation.

2.0 Mission

2.1 Mission Summary

2.1.1 Land Operation -- The VTOL and STOL test missions are described in Figure 1.

- Minimum Mission Time - Level 1

VTOL Missions	1/2 hour
STOL Missions	1 hour
Cruise/Endurance Mission	2 hours

- Payload (not including crew) 2500 lbs (minimum)
  50 cu. ft.

2.1.2 Shipboard Operation -- The aircraft shall be capable of operating from the deck of a Naval aircraft carrier.

2.2 Minimum Cruise Speed

- 300 KEAS at sea level and 0.7 at 25,000 ft.

3.0 General Design Guidelines

3.1 Austerity is to be stressed but not by compromising safety.

3.2 The limit load factor will be no less than +2.5g, -0.5g at design gross weight.

3.3 Sufficient attitude control power will be available to perform research on control requirements. The contractor shall indicate those axes where greater control power than required in section 1.0 would be made available for research purposes.

3.4 New aircraft components will be designed for approximately 500 flight hours.

3.5 Additional Information

- Crew	2 pilots (flyable by one pilot only, or by either pilot)
- Sink rate at touchdown	12 fps at max landing weight, 15 fps desired

- Pressurized cockpit is desired but not required
- Oxygen required
- Cockpit Environmental System — Minimum
- Pilot's Primary Flight Controls — Stick and Pedals
- Ejection System for both pilots
- Maximum possible visibility

3.6 The Contractor shall furnish as a minimum:

A. Conceptual design aircraft layout drawings.

B. Mil Std. 1374 Part 1 shall be used to show the empty weight breakdown into the usual structural and system group including additions and deletions to the original aircraft.

C. Low speed performance envelope at design gross weight.

D. Conceptual definition of proposed aircraft low speed control and stabilization system.

E. Control moment coefficients and control power about each axis with all gas generators operating and with most critical gas generator failed.

F. Engine and fan data which were used to calculate mission performance in all flight modes.

4.0 Summary of costing information required for the Research and Technology Aircraft

The Cost Breakdown is for a two airplane buy. The Cost Breakdown shall be stated in five pricing elements;

(1) Engineering labor

(2) Manufacturing labor

(3) Materials and Purchased Items

(4) Other direct costs

(5) Spares (if any). A listing of Government Furnished Equipment (GFE) assumed in the costing shall be included. It is intended that the costing information shall be complete in that the total costs of the subitems listed in paragraphs 4.1 thru 4.8 shall equal the total costs of the aircraft excluding the GFE items.

4.1 Airframe Design and Modification including:

- Landing Gear
- Subsystem and conventional controls
- Cockpit
- Ejection seats
- Wings
- Fuselage
- Empennage
- Miscellaneous

4.2 Propulsion system including:

- Components in 5.0
- Transmission components
- Transmission subsystem
- Thrust vectoring
- Miscellaneous

4.3 Control System including:

- Fly-by-wire controls
- Augmentation systems
- Miscellaneous

4.4 Propulsion System Testing including:

- o Components in 5.0
- o Transmission components
- o Thrust vectoring
- o Qualification tests
- o Aircraft ground tests
- o Miscellaneous

4.5 Control System Aircraft Testing including:

- o Component tests
- o System integration
- o Aircraft ground tests

4.6 Aircraft Ground Tests

- o Excluding aircraft ground tests in sections 4.4 and 4.5

4.7 Ejection Seat Tests

4.8 Flight Tests

- o Contractor Flight Test

4.9 Government Furnished Equipment including:

- o NA265-40 basic airframe
- o Airframe components
- o Fans
- o Engines
- o Research instrumentation
- o Miscellaneous

5.0 Summary of the Costing Information required for the high risk propulsion components

The costs for each component shall be stated in four pricing elements; engineering labor, manufacturing labor, material and purchased items, and other direct costs. For each of the pricing elements, the component costs shall be stated for the following categories: data base requirements (effort required to accumulate required data before detail design including data search, analysis, tests, etc.), design and manufacture, component testing, and unit qualification testing. Thus each component costs shall be stated in a four by four matrix.

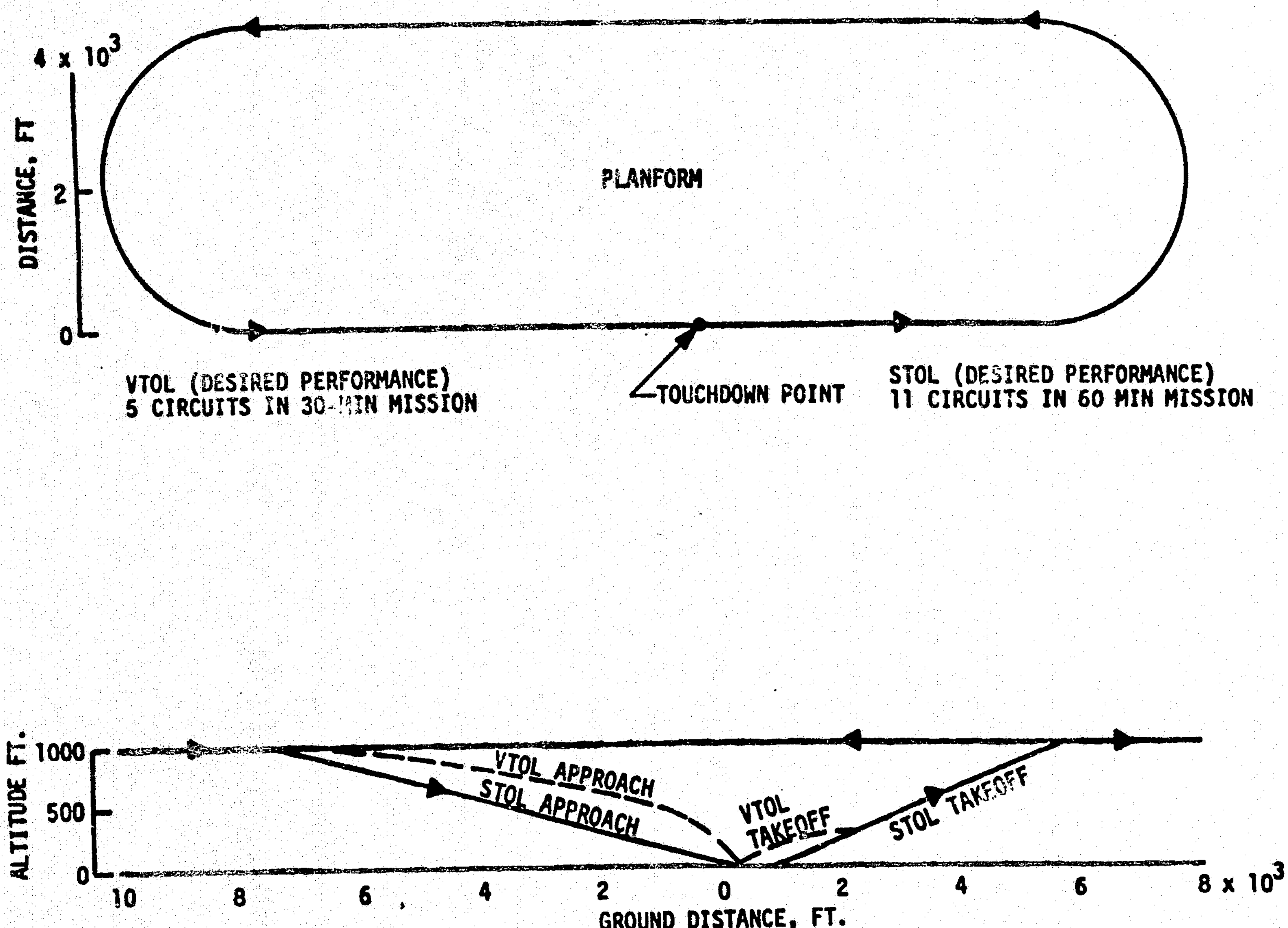


FIGURE 1 LCFA TECHNOLOGY AIRPLANES, TYPICAL TERMINAL AREA TEST MISSIONS

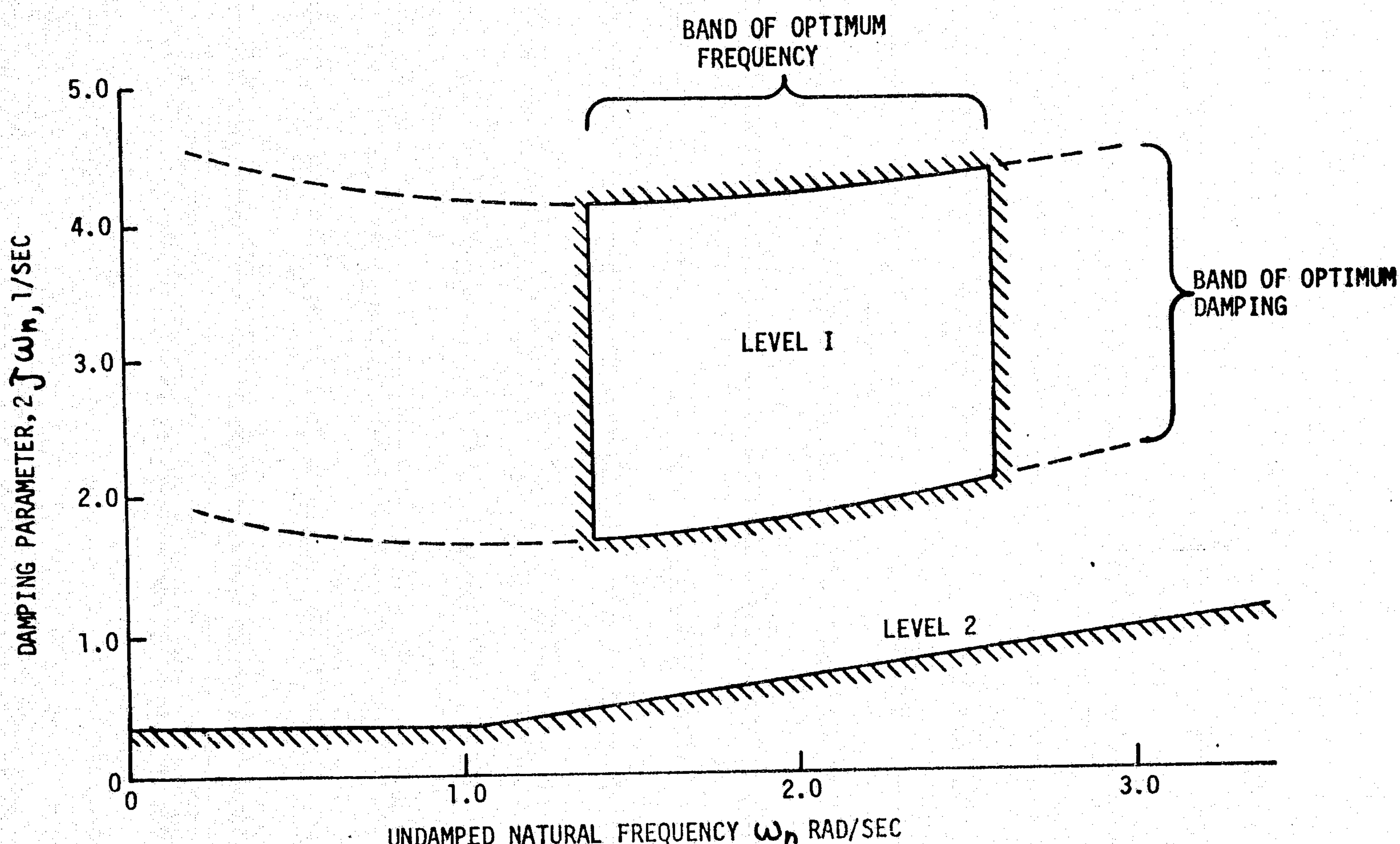


FIGURE 2: DYNAMIC STABILITY CRITERIA

APPENDIX B SCHEDULES

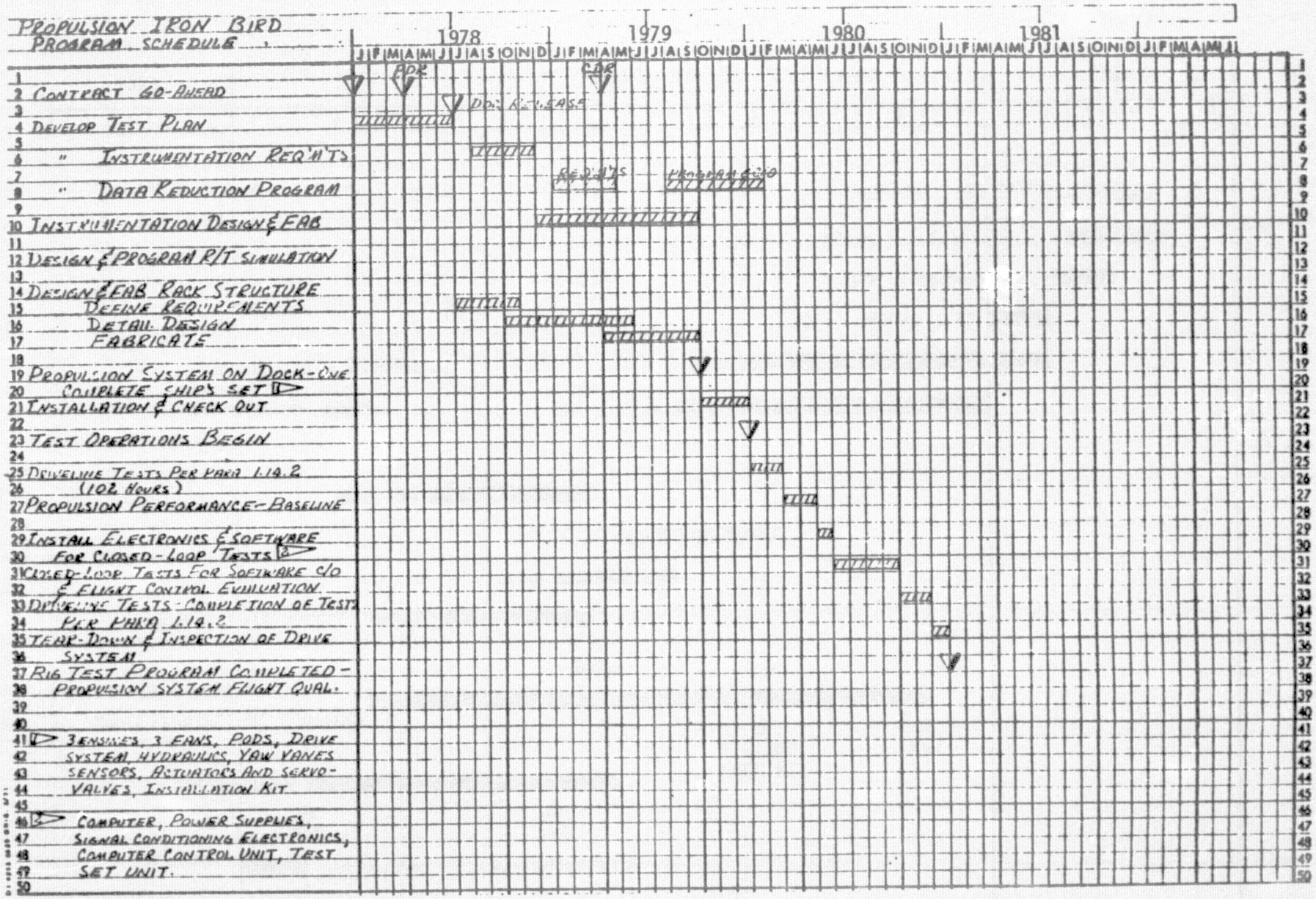
Overall Program Schedule

Figure B-1 shows the schedule for design, fabrication and flight test for two V/STOL Technology Demonstrator airplanes. The schedule assumes a contract award early in calendar year 1978 allowing flight test of the first vehicle to begin in the second quarter of 1981.

Schedule of Integrated System Testing (Iron Bird)

A comprehensive system-level test program conducted with a full-scale propulsion test rig (iron-bird) is considered essential to the timely development of a safe, reliable flight vehicle. Experience has shown that low risk, step-by-step system tests, with complexity increasing at each step, is a cost-effective way to develop flight systems that consistently meet their cost, schedule, and performance objectives.

Figure B-2 shows the propulsion Iron Bird Program Schedule to be consistent with the overall program plan.



PROPULSION IRON BIRD TEST PROGRAM SCHEDULE
FIGURE B-1

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TECHNOLOGY DEMONSTRATOR PROGRAM
MODEL 1041-135-2A

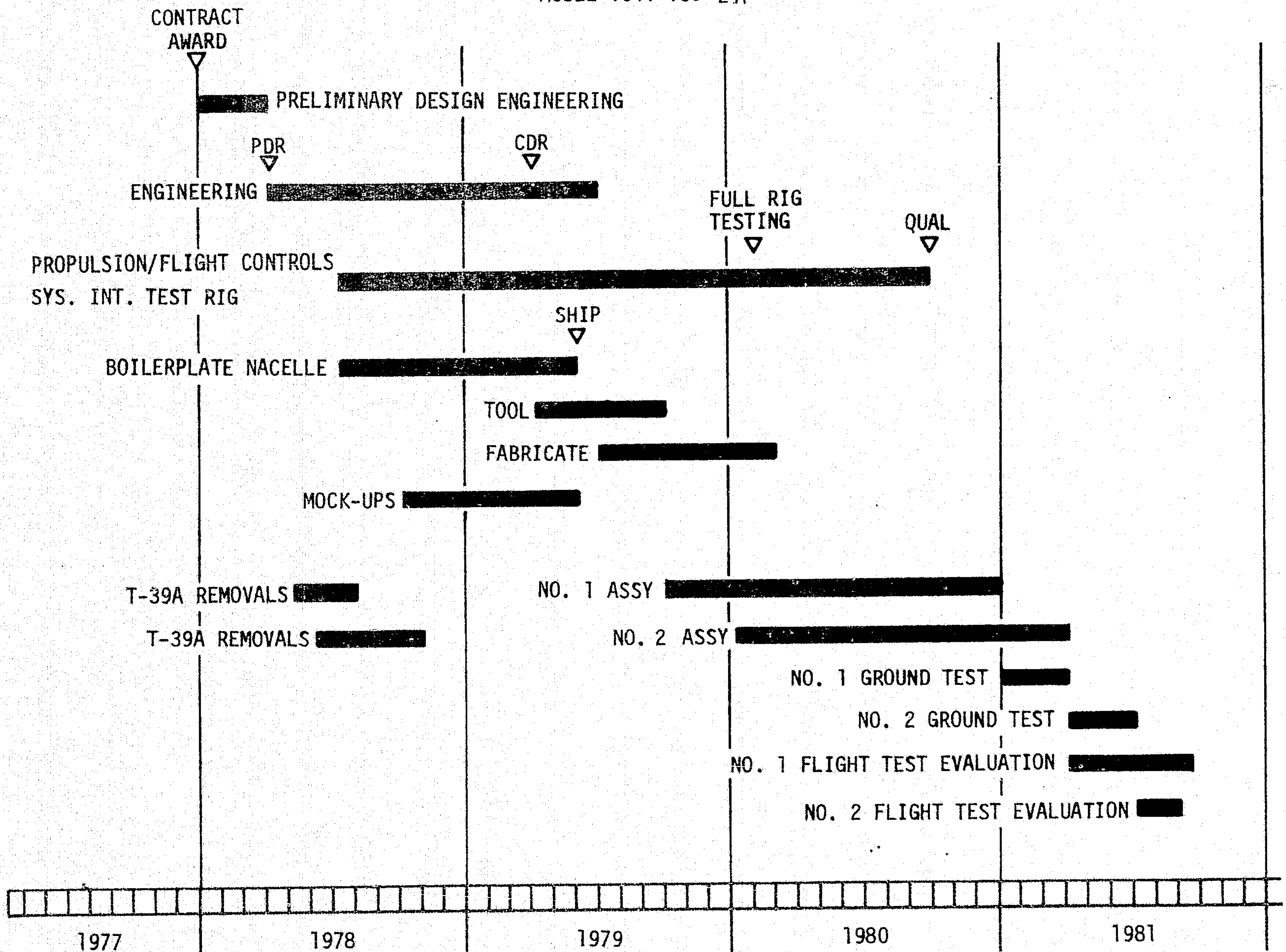

FIGURE C-2

APPENDIX "C" WEIGHTS

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WING	WEIGHT	HORIZONTAL	
		ARM	MOMENT
BASIC T-39A WING WEIGHT	1661	265.9	
ADD LANDING GEAR BEAM	+ 42	295.0	
ADD LANDING GEAR CUTOUT, DOOR & MECHANISM	+ 42	282.0	
REVISE INBOARD SECTION OF REAR SPAR FOR INCREASE IN LANDING LOADS	+ 12	270.0	
REVISE INBOARD TRAILING EDGE TO ACCOMMODATE LANDING GEAR & POD ROTATION. INCREASE THICKNESS	NEGLI.	-	
ADD FAIRING TO INBOARD PORTION OF STRUCTURAL BOX	+ 30	280.0	
BEEF-UP BASIC BOX & RIBS (INBD) FOR INCREASE LANDING LOADS	+ 48	243.0	
MISC. JTS, SPLICES, ETC.	+ 11	265.0	
ROUND-OFF	+ 4	266.0	
MODEL 1041-135-2R WING WT	1850	266.6	
REMOVE 30" WING TIP SECTION & OUTBOARD SECTION OF AILERON	- 70	341.0	
MODEL 1041-135-2A WING WT.	1780	263.7	
NO CHANGE	0	-	
MODEL 1041-135-2B WING WT.	1780	263.7	

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HORIZONTAL TAIL	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A HTAIL WEIGHT	220	472.6	
REMOVE EXISTING HORIZONTAL TAIL	- 220	472.6	
ADD NEW SLAB TAIL TO TIP OF VERTICAL DESIGN FOR NEW LOADS & FACTORS	+ 350	497.0	
MODEL 1041-135-2R HORIZ. TAIL WEIGHT	350	497.0	
NO CHANGE	0	-	
MODEL 1041-135-2A HORIZ. TAIL WEIGHT	350	497.0	
NO CHANGE	0	-	
MODEL 1041-135-2B HORIZ. TAIL WEIGHT	350	497.0	

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VERTICAL TAIL	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A VERTICAL TAIL WEIGHT	91	471.6	
NEW VERTICAL TAIL FOR TEE CONFIGURATION & NEW DESIGN LOADS	+ 99	465.0	
MODEL 1041-135-2R VERTICAL TAIL WEIGHT	190	467.9	
NO CHANGE	0	-	
MODEL 1041-135-2A VERTICAL TAIL WEIGHT	190	467.9	
NO CHANGE	0	-	
MODEL 1041-135-2B VERTICAL TAIL WEIGHT	190	467.9	

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BODY	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A BODY WEIGHT	1780	217.3	
REVISE NOSE SECTION FORWARD OF COCKPIT FOR INSTALLATION OF NOSE FAN & FLT. TEST EQUIPMENT	+ 140	35.0	
REMOVE SPEED BRAKES & SUPPORTS	- 22	169.0	
REPLACE PRESENT CANOPY WITH LIGHT FRANGIBLE ACRYLIC	- 50	114.0	
ADD DRIVE SYSTEM SUPPORT & NOISE PROTECTION	+ 224	170.0	
REVISE BODY BASIC STRUCTURE FOR INCREASES IN INERTIA LOADS, TAIL LOADS & LANDING LOADS	+ 200	240.0	
ROUND-OFF	- 2	206.0	
MODEL 1041-135-2R BODY WEIGHT	2270	206.1	
REMOVE BS 334 PRESSURE BULKHEAD	- 41	334.0	
ADD 3RD ENGINE SUPPORT BULKHEAD	+ 21	334.0	
ADD ENTRANCE DOOR TO BULKHEAD 143	+ 20	143.0	
ADD 3RD ENGINE AIR INLET DOORS, DUCTING/PLENUM & FLEX SEAL	+ 50	284.0	
REPLACE (1) BODY WINDOW WITH ENGINE ROOM VENTILATION INLET	NEGLI.	-	
NOSE BOOM WT. INCLUDED IN FLIGHT TEST EQUIPMENT	-	-	
MODEL 1041-135-2A BODY WEIGHT	2320	206.1	
NO CHANGE	0	-	
MODEL 1014-135-2B BODY WEIGHT	2320	206.1	

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MAIN LANDING GEAR	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A MAIN GEAR WEIGHT	467	254.0	
REMOVE EXISTING MAIN GEAR	- 467	254.0	
INSTALL A-4 MAIN GEAR WITH SLIGHT MODIFICATIONS	+ 680	284.0	
. USE A-4 ROLLING GEAR (+235 LB/APL)			
. USE A-4 SHOCK & DRAG STRUTS WITH MODIFIED			
METERING PIN (+320 LB/APL)			
. INSTALL SHOCK STRUT IN A NEW DESIGN SUPPORT			
JACKET & ATTACHMENTS (+70 LB/APL)			
. REMOVE A-4 SWIVELING MECHANISM (-9 LB/APL)			
. USE A-4 CONTROLS (+64 LB/APL)			
MODEL 1041-135-2R MAIN GEAR WEIGHT	680	284.0	
NO CHANGE			
MODEL 1041-135-2A MAIN GEAR WEIGHT	680	284.0	
NO CHANGE			
MODEL 1041-135-2B MAIN GEAR WEIGHT	680	284.0	

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NOSE LANDING GEAR	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A NOSE GEAR WEIGHT	147	98.0	
REMOVE EXISTING NOSE GEAR	- 147	98.0	
INSTALL A-4 NOSE GEAR WITH SOME MODIFICATIONS	+ 210	80.0	
. USE A-4 ROLLING GEAR (+33 LB/APL)			
. USE A-4 STRUCTURE WITH MODIFIED METERING PIN & KNUCKLED DRAG BRACES FOR AFT RETRACTION CAPABILITY (+143 LB/APL)			
. USE A-4 CONTROLS (+34 LB/APL)			
MODEL 1041-135-2R NOSE GEAR WEIGHT	210	80.0	
NO CHANGE	0	-	
MODEL 1041-135-2A NOSE GEAR WEIGHT	210	80.0	
NO CHANGE			
MODEL 1041-135-2B NOSE GEAR WEIGHT	210	80.0	

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NACELLE	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A NACELLE WEIGHT	372	334.0	
REMOVE EXISTING ENGINE PODS	- 372	334.0	
ADD NOSE FAN CASE & FRAME	+ 257	37.0	
ADD NOSE FAN INLET DOORS	+ 79	32.5	
ADD L/C NOSE COWL, INLET & FAN COWL	530	313.0	
ADD ENGINE COWL plus Eng. Mounts	245	392.0	
ADD L/C FAN FRAME & CASE	831	332.0	
ADD STRUTS	351	343.0	
ROUND OFF	- 3	290.0	
MODEL 1041-135-2R NACELLE WEIGHT	2270	290.2	
ADD 3RD ENGINE MOUNTING & PROVISIONS			
. BULKHEADS & INLET PLENUM INCLUDED IN BODY WT			
. EXHAUST TUBES INCLUDED IN "EXHAUST & DEFLECTOR" WEIGHT	+ 40	335.0	
ADD 3RD ENGINE FIREWALL	+ 40	332.0	
MODEL 1041-135-2A NACELLE WEIGHT	2350	291.7	
REDUCE ENGINE FIREWALL DIA., ADD LENGTH	NEGLI.	-	
MODEL 1041-135-2B NACELLE WEIGHT	2350	291.7	

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WING SKID	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A WING SKID WEIGHT	15	223.0	
REMOVE WING SKID	- 15	223.0	
MODEL 1041-135-2R WING SKID WEIGHT	0	-	
NO CHANGE	0	-	
MODEL 1041-135-2A WING SKID WEIGHT	0	-	
NO CHANGE	0	-	
MODEL 1041-135-2B WING SKID WEIGHT	0	-	

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ENGINE	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39 ENGINE WEIGHT	980	336.0	
DELETE (2) P&W J60-P-3A ENGINES	- 980	336.0	
ADD (2) DDA XT701-AD-700 TURBO SHAFT ENGINES	+2270	391.0	
MODEL 1041-135-2R ENGINE WEIGHT	2270	391.0	
ADD (1) DDA XT701-AD-700 TURBO SHAFT ENGINE IN THE BODY ON BODY C_L	+1135	332.0	
ROUND OFF	+ 5	371.0	
MODEL 1041-135-2A ENGINE WEIGHT	3410	371.3	
REMOVE (1) DDA XT701-AD-700 TURBOSHAFT ENGINE IN THE BODY ON BODY C_L	-1135	332.0	
ADD (1) DDA T56-A-14 TURBOSHAFT ENGINE IN THE BODY ON BODY C_L	+1535	342.0	
MODEL 1041-135-2B ENGINE WEIGHT	3610	372.8	

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MODEL 1041-135

ENGINE ACCESSORIES	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A ENGINE ACCESSORIES WEIGHT	17	330.2	
DELETE T-39 COOLING, DRAIN & LUBE SYSTEMS	- 17	330.2	
ENGINE ACCESSORIES FOR (2) XT701-AD-700 ENGINES ARE INCLUDED IN ENGINE WT.	0	-	
MODEL 1041-135-2R ENGINE ACCESSORIES WEIGHT	0	-	
NO CHANGE	0	-	
MODEL 1041-135-2A ENGINGE ACCESSORIES WEIGHT	0	-	
NO CHANGE	0	-	
MODEL 1041-135-2B ENGINE ACCESSORIES WEIGHT	0	-	

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ENGINE CONTROLS	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A ENGINE CONTROLS WEIGHT	30	194.0	
REMOVE EXISTING CONTROLS	- 30	194.0	
ADD FLY-BY-WIRE CONTROLS FOR L/C ENGINES	+ 30	218.0	
MODEL 1041-135-2R ENGINE CONTROLS WEIGHT	30	218.0	
ADD 3RD ENGINE THROTTLE & CONTROLS	+ 15	305.0	
MODEL 1041-135-2A ENGINE CONTROLS WEIGHT	45	247.0	
NO CHANGE	0	-	
MODEL 1041-135-2B ENGINE CONTROLS WEIGHT	45	247.0	

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STARTING SYSTEM	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A STARTING SYSTEM WEIGHT	22	269.0	
REMOVE EXISTING STARTING SYSTEM	- 22	269.0	
ADD GROUND CART STARTING PROVISIONS	+ 20	368.0	
MODEL 1041-135-2R STARTING SYSTEM WEIGHT	20	368.0	
ADD GROUND CART STARTING FOR 3RD ENGINE	+ 15	305.0	
MODEL 1041-135-2A STARTING SYSTEM WEIGHT	35	341.0	
NO CHANGE	0	-	
MODEL 1041-135-2B STARTING SYSTEM WEIGHT	35	341.0	

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FUEL SYSTEM	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A FUEL SYSTEM WEIGHT	195	292.0	
REMOVE AFT FUSELAGE BLADDER TANK & SUPPORTS (183 GAL.)	- 35	343.0	
REMOVE EXISTING PUMPS & DISTRIBUTION SYSTEM	- 53	348.9	
ADD NEW PUMPS TO SUPPLY THE (2) XT701-AD-700 ENGINES	+ 26	400.0	
REVISE DISTRIBUTION SYSTEM LINE SIZES & LOCATIONS FOR NEW PUMP CAPACITIES & ENGINE PLACEMENT	+ 67	270.0	
REMOVE VENT, DRAIN & FILLING PROVISIONS FOR AFT BODY BLADDER TANK	- 20	343.0	
MODEL 1041-135-2R FUEL SYSTEM WEIGHT	180	267.1	
ADD PUMP FOR 3RD ENGINE	+ 13	320.0	
ADD DISTRIBUTION, TRANSFER & CONTROL SYSTEMS FOR 3RD ENGINE	+ 35	280.0	
ROUND OFF	+ 2	272.0	
MODEL 1041-135-2A FUEL SYSTEM WEIGHT	230	272.1	
REVISE 3RD ENGINE PUMPS & PLUMBING	NEGLI.	-	
MODEL 1041-135-2B FUEL SYSTEM WEIGHT	230	272.1	

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FANS	WEIGHT	HORIZONTAL	
		ARM	MOMENT
BASIC T-39A FAN WEIGHT	0	–	
ADD FWD 62" DIAM. HAMILTON STANDARD FANS WITH ADV. COMPOSITE BLADES	+ 321	37.0	
ADD (2) 62" L/C FANS	+ 642	332.0	
ROUND OFF	– 3	235.0	
MODEL 1041-135-2R FAN WEIGHT	960	233.7	
NO CHANGE	0	–	
MODEL 1041-135-2A FAN WEIGHT	960	233.7	
NO CHANGE	0	–	
MODEL 1041-135-2B FAN WEIGHT	960	233.7	

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DRIVE SYSTEM	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A DRIVE SYSTEM WEIGHT	0	-	
ADD DRIVE SYSTEM FOR (2) XT701-AD-700 ENGINES DRIVING (3) 62" HAMILTON STANDARD FANS. WATER INJECTION USED FOR AUGMENTATION.			
ADD NOSE FAN GEARBOX (1)	+ 316	75.0	
ADD L/C FAN GEARBOXES (2)	+ 806	343.0	
ADD SYCHRO CLUTCH (1)	+ 80	332.0	
ADD OVERRUNNING CLUTCHES (2)	+ 70	358.0	
ADD TEEBOX (1)	+ 380	354.0	
ADD LONGITUDINAL DRIVE SHAFTING & BEARINGS	+ 199	202.0	
ADD CROSS SHAFTING BETWEEN L/C PODS	+ 68	354.0	
ADD CONTROLS	+ 34	230.0	
ADD COOLERS, BLOWER & LUBE PUMPS	+ 114	330.0	
ROUND OFF	+ 3	288.0	
MODEL 1041-135-2R DRIVE SYSTEM WEIGHT	2070	288.3	
ADD OVERRUN CLUTCH (1)	+ 35	299.0	
ADD DROP BOX	+ 489	258.0	
REVISE LONGITUDINAL DRIVE SHAFT & ADD LENGTH FROM 3RD ENGINE TO DROP BOX	+ 29	286.0	
CHANGE LOCATION OF SYCHRO CLUTCH	+ 80	273.0	
	- 80	334.0	
ROUND OFF	- 3	281.0	
MODEL 1041-135-2A DRIVE SYSTEM WEIGHT	2620	280.9	
NO CHANGE	0	-	
MODEL 1041-135-2B DRIVE SYSTEM WEIGHT	2620	280.9	

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EXHAUST & DEFLECTORS	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A EXHAUST WEIGHT	28	384.0	
ADD NOSE FAN EXHAUST LOUVERS	+ 220	43.0	
REMOVE EXISTING EXHAUST WEIGHT	- 28	384.0	
ADD TWO-POSITION FAN NOZZLE & YAW VANES	+ 410	376.0	
ADD L/C POD ROTATION PROVISIONS	+ 350	354.0	
MODEL 1041-135-2R EXHAUST & DEFLECTORS WEIGHT	980	293.4	
ADD 3RD ENGINE EXHAUST TUBES, STIFFENING & INSULATION	+ 80	382.0	
MODEL 1041-135-2A EXHAUST & DEFLECTORS WEIGHT	1060	300.1	
REMOVE XT701-AD-700 (3RD ENGINE) EXHAUST TUBES, STIFFENING & INSULATION	- 80	382.0	
ADD T56-4-14 EXHAUST TUBE, STIFFENING & INSULATION	+ 70	406.0	
MODEL 1041-135-28 EXHAUST & DEFLECTORS WEIGHT	1050	300.9	

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H_2O INJECTION SYSTEM	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A H_2O INJECTION WEIGHT	0	-	
ADD WATER INJECTION SYSTEM	+ 150	375.0	
MODEL 1041-135-2R H_2O INJECTION WEIGHT	150	375.0	
REMOVE WATER INJECTION SYSTEM	- 150	375.0	
MODEL 1041-135-2A H_2O INJECTION WEIGHT	0	-	
NO CHANGE	0	-	
MODEL 1041-135-2B H_2O INJECTION WEIGHT	0	-	

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FLIGHT CONTROLS	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A FLIGHT CONTROLS WEIGHT	326	236.1	
REMOVE SPEED BRAKE CONTROLS	- 28	184.0	
REVISE, BEEF-UP &/OR REDESIGN FLIGHT CONTROLS FOR HIGHER GROSS WEIGHTS, INERTIAS, ETC.	+360	302.0	
EMPLOY FLY-BY-WIRE	- 130	306.0	
ROUND OFF	+ 2	267.0	
MODEL 1041-135-2R FLIGHT CONTROLS WEIGHT	530	266.6	
NO CHANGE	0	-	
MODEL 1041-135-2A FLIGHT CONTROLS WEIGHT	530	266.6	
NO CHANGE	0	-	
MODEL 1041-135-2B FLIGHT CONTROLS WEIGHT	530	266.6	

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INSTRUMENTS	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A INSTRUMENTS WEIGHT	166	197.9	
REMOVE FUEL QUANTITY INSTRUMENTS TO AFT BODY BLADDER FUEL TANK	- 7	235.0	
REMOVE SEXTANT & CASE (STOWED)	- 13	196.0	
REMOVE NAVIGATOR STATION INSTRUMENTS	- 25	271.0	
ADD FAN INSTRUMENTS	+ 60	200.0	
ADD DRIVE SYSTEM INSTRUMENTS	+ 30	200.0	
ADD ADDITIONAL INSTRUMENTS FOR V/STOL FLIGHT MONITORING	+ 30	123.0	
ROUND OFF	- 1	181.0	
MODEL 1041-135-2R INSTRUMENT WEIGHT	240	180.8	
ADD 3RD ENGINE INSTRUMENTS	+ 30	130.0	
MODEL 1041-135-2A INSTRUMENT WEIGHT	270	175.2	
NO CHANGE	0	-	
MODEL 1041-135-2B INSTRUMENT WEIGHT	270	175.2	

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HYDRAULICS	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A HYDRAULICS WEIGHT	145	341.3	
REMOVE EXISTING HYDRAULICS SYSTEM	- 145	341.3	
ADD NEW DESIGN HYDRAULIC SYSTEM (3000 PSI)	+ 220	355.0	
MODEL 1041-135-2R HYDRAULIC WEIGHT	220	355.0	
NO CHANGE	0	-	
MODEL 1041-135-2A HYDRAULIC WEIGHT	220	355.0	
NO CHANGE	0	-	
MODEL 1041-135-2B HYDRAULIC WEIGHT	220	355.0	

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ELECTRICAL	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A ELECTRICAL WEIGHT	924	262.4	
REMOVE PASSENGER COMPARTMENT LIGHTING SYSTEM	- 20	175.0	
REDESIGN THE ELECTRICAL SYSTEM FOR (2) 20 KVA GENERATORS AND A SMALL BATTERY			
. REDUCE POWER SUPPLY WEIGHT (-80 LB/APL)			
. REDUCE POWER CONVERSION WEIGHT (-60 LB/APL)			
. REDUCE DISTRIBUTION & CONTROL & TRANSFER SYSTEM WEIGHT (-300 LB/APL)	- 440	252.0	
ROUND OFF	- 4	276.0	
MODEL 1041-135-2R ELECTRICAL WEIGHT	460	276.0	
NO CHANGE	0	-	
MODEL 1041-135-2A ELECTRICAL WEIGHT	460	276.0	
NO CHANGE	0	-	
MODEL 1041-135-2B ELECTRICAL WEIGHT	460	276.0	

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AVIONICS	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A AVIONICS WEIGHT	464	97.5	
REMOVE EQUIPMENT NOT REQUIRED FOR DEMONSTRATION FLIGHTS	-204	97.5	
MODEL 1041-135-2R AVIONICS WEIGHT	260	97.5	
NO CHANGE	0	-	
MODEL 1041-135-2A AVIONICS WEIGHT	260	97.5	
NO CHANGE	0	-	
MODEL 1041-135-2B AVIONICS WEIGHT	260	97.5	

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FURNISHINGS	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A FURNISHINGS WEIGHT	877	202.3	
REMOVE EXISTING SEATS	- 182	192.9	
ADD (2) SIIIS EJECTION SYSTEMS	+ 260	128.0	
REMOVE EXISTING OXYGEN INSTALLATION	- 74	140.0	
ADD A SMALL CAPACITY (5 LITERS) O_2 SYSTEM	+ 37	140.0	
REMOVE CONSOLES, CHARTBOARD & LEG & CONTROL STANDS	- 90	181.0	
REMOVE FLOOR COVERING, SOUNDPROOFING, THERMAL INSULATION, INTERIOR TRIM, CURTAINS, SCREENS, PARTITIONS & BAGGAGE PROVISIONS	- 392	215.0	
ADD COCKPIT SOUNDPROOFING & INSULATION	+ 60	106.0	
REMOVE PORTABLE FIRE EXTINGUISHER, AXE & FIRST AID KIT	- 13	240.0	
ROUND OFF	- 3	151.0	
MODEL 1041-135-2R FURNISHINGS WEIGHT	480	151.3	
ADD FIRE EXTINGUISHING FOR THE 3RD ENGINE. USE THE EXISTING FIRE DETECTION SYSTEM IN THE PASSENGER CABIN FOR 3RD ENGINE FIRE DETECTION	+ 27	332.0	
ROUND OFF	+ 3	161.0	
MODEL 1041-135-2A FURNISHINGS WEIGHT	510	160.9	
NO CHANGE	0	-	
MODEL 1041-135-2B FURNISHINGS WEIGHT	510	160.9	

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AIR CONDITIONING	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A AIR CONDITIONING WEIGHT	279	257.2	
REMOVE AIR-CONDITIONING & PRESSURIZATION SYSTEM FOR EXISTING PASSENGER COMPARTMENT	- 47	182.8	
ADD EQUIPMENT COOLING PROVISIONS FOR FLIGHT TEST EQUIPMENT	+ 100	53.0	
ROUND OFF	- 2	206.0	
MODEL 1041-135-2R AIR CONDITIONING WEIGHT	330	206.2	
ADD 3RD ENGINE ROOM VENTILATION PROVISIONS	+ 30	300.0	
MODEL 1041-135-2A AIR CONDITIONING WEIGHT	360	214.0	
NO CHANGE	0	-	
MODEL 1041-135-2B AIR CONDITIONING WEIGHT	360	214.0	

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ANTI-ICING	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A ANTI-ICING WEIGHT	54	237.0	
REMOVE WING ANTI-ICING	- 16	267.0	
REVISE AIR INLET ANTI-ICING TO ACCOMMODATE NEW NACELLE SIZE	+ 12	290.0	
MODEL 1041-135-2R ANTI-ICING WEIGHT	50	240.1	
ADD 3RD ENGINE INLET ANTI-ICING	+ 10	281.0	
MODEL 1041-135-2A ANTI-ICING WEIGHT	60	246.9	
NO CHANGE	0	-	
MODEL 1041-135-2B ANTI-ICING WEIGHT	60	246.9	

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LOAD & HANDLING	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A LOAD & HANDLING WEIGHT	3	210.3	
INCREASE JACKING & HOISTING PROVISIONS DUE TO HIGHER GROSS WEIGHTS	+ 7	210.3	
MODEL 1041-135-2R LOAD & HANDLING WEIGHT	10	210.3	
NO CHANGE	0	-	
MODEL 1041-135-2A LOAD & HANDLING WEIGHT	10	210.3	
NO CHANGE	0	-	
MODEL 1041-135-2B LOAD & HANDLING WEIGHT	10	210.3	

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CREW	WEIGHT	HORIZONTAL	
		ARM	MOMENT
BASIC T-39A CREW WEIGHT	340	115.0	
USE NAVY CREW WEIGHT	+ 20	115.0	
MODEL 1041-135-2R CREW WEIGHT	360	115.0	
NO CHANGE	0	-	
MODEL 1041-135-2A CREW WEIGHT	360	115.0	
NO CHANGE	0	-	
MODEL 1041-135-2B CREW WEIGHT	360	115.0	

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UNUSABLE FUEL	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A UNUSABLE FUEL WEIGHT	170	248.4	
REMOVE UNUSABLE FUEL IN AFT BODY BLADDER TANK	- 30	345.0	
MODEL 1041-135-2R UNUSABLE FUEL WEIGHT	140	227.7	
NO CHANGE	0	-	
MODEL 1041-135-2A UNUSABLE FUEL WEIGHT	140	227.7	
NO CHANGE	0	-	
MODEL 1041-135-2B UNUSABLE FUEL WEIGHT	140	227.7	

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OIL & TRAPPED OIL	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A OIL & TRAPPED OIL WEIGHT	24	324.0	
REMOVE EXISTING OIL & TRAPPED OIL	- 24	324.0	
ADD OIL & TRAPPED OIL FOR (2) XT701-AD-700 ENGINES & (3) 62" FANS. DRIVE SYSTEM OIL IS INCLUDED IN DRIVE SYSTEM WEIGHT	+ 110	299.0	
MODEL 1041-135-2R OIL & TRAPPED WEIGHT	110	299.0	
ADD OIL FOR 3RD ENGINE	+ 20	310.0	
MODEL 1041-135-2A OIL & TRAPPED WEIGHT	130	300.7	
NO CHANGE	0	-	
MODEL 1041-135-2B OIL & TRAPPED WEIGHT	130	300.7	

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CREW EQUIPMENT	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A CREW EQUIPMENT WEIGHT	0	-	
NONE REQUIRED			
MODEL 1041-135-2R CREW EQUIPMENT WEIGHT	0	-	
NO CHANGE	0	-	
MODEL 1041-135-2A CREW EQUIPMENT WEIGHT	0	-	
NO CHANGE	0	-	
MODEL 1041-135-2B CREW EQUIPMENT WEIGHT	0	-	

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INTERNAL WING FUEL	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A INT. WING FUEL CAPACITY WEIGHT	5818	256.0	
CHANGE FUEL UNIT WEIGHT TO 6.8 LBS/GAL.	+ 272	256.0	
MODEL 1041-135-2R WING FUEL WEIGHT	6090	256.0	
NO CHANGE	0	-	
MODEL 1041-135-2A WING FUEL WEIGHT	6090	256.0	
NO CHANGE	0	-	
MODEL 1041-135-2B WING FUEL WEIGHT	6090	256.0	

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H_2O & ALCOHOL	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A H_2O & ALCOHOL WEIGHT		-	
ADD H_2O & ALCOHOL FOR AUGMENTATION	+ 240	375.0	
MODEL 1041-135-2R H_2O & ALCOHOL WEIGHT	240	375.0	
REMOVE H_2O + ALCOHOL	- 240	375.0	
MODEL 1041-135-2A H_2O + ALCOHOL WEIGHT	0	-	
NO CHANGE	0	-	
MODEL 1041-135-2B H_2O + ALCOHOL WEIGHT	0	-	

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PAYLOAD	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A PAYLOAD WEIGHT	1100	234.6	
REMOVE EXISTING PAYLOAD	-1100	234.6	
ADD NOSE BOOM	+ 100	- 70.0	
ADD FLIGHT TEST EQUIPMENT IN NOSE SECTION FORWARD OF NOSE FAN	+1200	- 10.0	
ADD FLIGHT TEST EQUIPMENT IN COMPARTMENT JUST AFT OF THE COCKPIT	+1200	140.0	
MODEL 1041-135-2R PAYLOAD WEIGHT	2500	59.6	
NO CHANGE	0	-	
MODEL 1041-135-2A PAYLOAD WEIGHT	2500	59.6	
NO CHANGE	0	-	
MODEL 1041-135-2B PAYLOAD WEIGHT	2500	59.6	

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CONTINGENCY	WEIGHT	HORIZONTAL ARM	HORIZONTAL MOMENT
BASIC T-39A CONTINGENCY WEIGHT	0	-	
ADD 10.5% OF OPERATING WEIGHT FOR CONTINGENCY ALLOWANCE TO COVER UNKNOWN DESIGN CONDITIONS & ASSOCIATED WEIGHT GROWTH	+1860		
MODEL 1041-135-2R CONTINGENCY WEIGHT	1860	223.6	
ADD ADDITIONAL CONTINGENCY @ +10.5% OF △ O.W. REQUIRED FOR THIRD ENGINE INSTALLATION & ASSOCIATED CHANGES	+ 190	223.6	
MODEL 1041-135-2A CONTINGENCY WEIGHT	2050	223.6	
ADD ADDITIONAL CONTINGENCY @ +10.5% OF △ O.W.	+ 20	223.6	
MODEL 1041-135-2B CONTINGENCY WEIGHT	2070	223.6	

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