CR-114682

ADVANCEMENT OF PROPROTOR TECHNOLOGY

TASK I – DESIGN STUDY SUMMARY (NASA CONTRACT NAS 2-5386)

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CONTENTS

CONTENTS - Continued

CONTENTS - Continued

LIST OF FIGURES

LIST OF FIGURES - Continued

LIST OF FIGURES - Continued

LIST OF FIGURES - Continued

LIST OF FIGURES - Continued

LIST OF DRAWINGS

LIST OF DRAWINGS - Continued

LIST OF TABLES

LIST OF TABLES - Continued

I. SUMMARY

A tilt-proprotor proof-of-concept aircraft design study has been conducted under NASA Contract NAS2-5386. The results are presented in this report. The objective of the contract is to advance the state of proprotor technology through design studies and full-scale wind-tunnel tests. The specific objective of Task I, which is now complete, is to conduct preliminary design studies to define a minimum-size tilt-proprotor research aircraft that can perform proof-of-concept flight research. The studies, which were based on prior work done under Bell Helicopter Company Independent Research and Development include aircraft layouts, weight estimations, performance calculations and dynamic analyses.

The aircraft that results from these studies (Figure I-1), designated the Bell Model 300, is a twin-engine, high-wing aircraft with 25-foot, three-bladed tilt proprotors mounted on pylons at the wingtips. Each pylon houses a Pratt and Whitney PT6C-40 engine with a takeoff rating of 1150 horsepower. Empty weight is estimated at 6876 pounds. The normal gross weight is 9500 pounds, and the maximum gross weight is 12,400 pounds.

The maximum level-flight speed of the Model 300 is 312 knots at 15,000 feet. The aircraft can hover out of ground effect at 6,400 feet and 95°F at the normal gross weight. The 4000 foot, 95°F hovering ceiling occurs at a weight of 10,300 pounds. Its suitability for flight research and simulated civil and military missions is analyzed and found to be more than adequate to demonstrate proof of the concept.

Performance, weight and dynamic analyses of the Model 300 are based on the layout drawings in Section X. The results are substantiated statistically and by model-test data obtained from previous Bell IR&D programs. Drag estimates are based on one-fifth-scale model wind-tunnel tests. The method of estimating proprotor performance is correlated with NASA tests of a 13-foot-diameter proprotor. The proprotor-pylon stability analyses shows good correlation with a one-seventh-scale aeroelastic model tested in the NASA-Langley 16-foot Transonic Tunnel, and with a one-fifth-scale semispan model of the Model 300 tested in a low-speed wind tunnel. High torsional stiffness of the wing provides a speed margin for proprotor-pylon stability of over 70 percent of the limit dive speed--far in excess of the 20-percent flutter-free margin required. The methods used to predict the important characteristics of the Model 300 appear to be valid, but they will be further substantiated under Task II of the contract, which will include the collection of full-scale test data in the NASA 40-by-80-foot tunnel, using a 25-foot proprotor of the same design as that for the Model 300 proof-of-concept aircraft.

As a part of the contract, a long-range proof-of-concept program has been developed. The proposed program involves three phases:

design and fabrication of the aircraft, flightworthiness tests, and proof-of-concept flight research. It is directed toward establishing proof of concept in three areas: technical, economic, and environmental (noise, downwash, etc.). The schedule calls for the first inflight conversion in the third quarter of 1972, in order to permit proof-of-concept flight research to start early in CY 1973. Alternatives to the basic program, including the folding-proprotor concept, are discussed.

It appears that the urgent need for civil and military VTOL transportation can be met by tilt-proprotor aircraft. Implementation of a proof-of-concept flight-research program is the next logical step toward meeting that need.

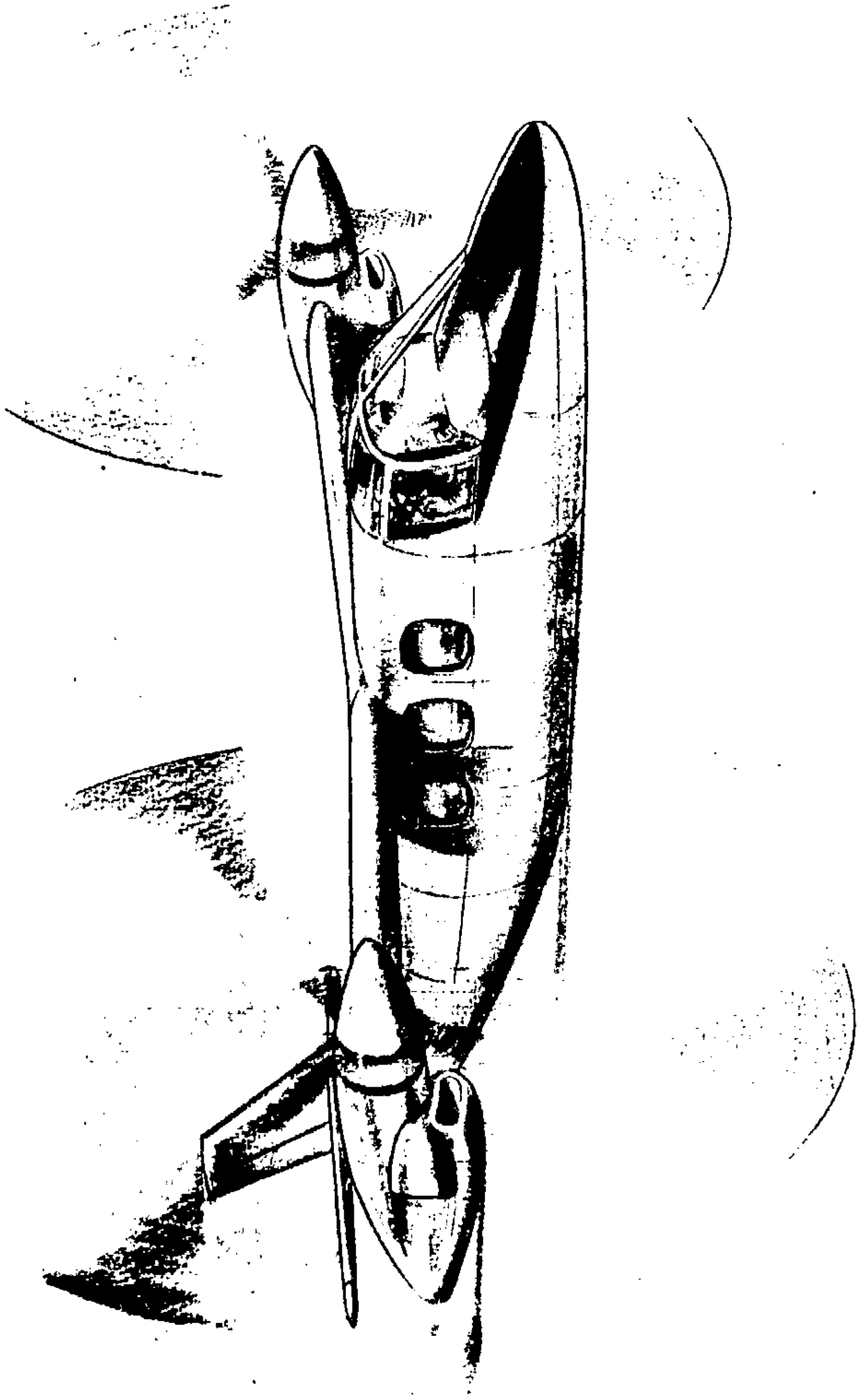


Figure I-1. Bell Model 300 Proof-Of-Concept Tilt-Proprotor Aircraft.

II. INTRODUCTION

Military and civil planners are becoming aware of the need for aircraft with VTOL capability in a variety of military missions and for civil transportation. The following paragraphs discuss this need and appraise the reasons for the failure of past efforts to meet the need. A NASA proof-of-concept program which could provide the first step for the logical and expedient attainment of operational VTOL aircraft is outlined. The qualifications of the tilt-proprotor VTOL aircraft for early operational application are discussed and the role of the proof-of-concept flight research aircraft is defined.

A. The Need for VTOL

The need for practicable VTOL aircraft in both military and civil transportation grows continually more urgent (References 1 through 13). On the military side, VTOL is essential to the requirements for fast reaction, operational flexibility, and economy of effort in such missions as airborne assault, local-area defense, antisubmarine warfare, and tactical logistics. The "Six-Day War" of 1967 showed how vulnerable an air-defense system can be if it is dependent on large airfields. Viet Nam has made one point clear: the ability of our armed forces to operate efficiently in the underdeveloped regions of the world must be retained and enhanced. Initially at least, airfields will be few or nonexistent; VTOL aircraft may be the only means of access to vital zones of combat. Even in highly developed nations, VTOL would be essential to counter an enemy who had incapacitated existing airfields and ground lines of transport.

In Viet Nam the helicopter (a slow, short-range VTOL) has proved its value in airborne assault, ground support, surveillance, medical evacuation, and rescue. Its remarkable usefulness is a good indicator of the greater VTOL potential that could and should be developed. Larger, faster, and longer-range VTOL aircraft can realize the full capability attainable for these and other missions.

On the civil side, commercial aviation can be expected to continue its rapid expansion. Airline operators are preparing for the introduction of faster and larger aircraft. The supersonic transport will cut flying time for transcontinental and intercontinental flight in half. Giant transport aircraft with capabilities of nearly 500 passengers will be entering service in 1970. Yet the air traveler may receive poorer service in terms of travel time to his ultimate destination. Air traffic has already overloaded our airports, not only in their ability to handle the airplanes and load and unload passengers and baggage, but also in moving passengers out to their final destinations. Decreasing flight times will no longer significantly shorten total trip time. VTOL aircraft can help to avoid this stagnation in civil air transport, and open new opportunities to benefit from the rapid progress of aeronautics.

A VTOL transportation system can relieve congestion at major airports by carrying passengers between cities up to 500 miles apart, using convenient VTOL ports near their points of origin and destination. VTOL transportation would also be available to carry passengers from the major regional airports into nearby cities.

The development of such a system will require the cooperation of local and federal planners in integrating the necessary facilities and services, and formulating regulations for the operation of the system. Most important, VTOL aircraft that can provide the desired services at reasonable cost, with low noise, ground disturbance, and environmental pollution, and with high standards of safety must be developed.

B. The Failure to Meet the VTOL Need

Despite the urgent need for VTOL aircraft, their development has been painfully slow. In 1968, the Director of Defense Research and Engineering told Congress that the US had built 17 V/STOL aircraft during the preceding 10 years, and had spent more than half a billion dollars without real results. The best design approach for VTOL is not yet obvious. A great variety of VTOL concepts have been proposed, and many have been flight-tested. Some have demonstrated their technical feasibility, but none has been selected for production and operation in the United States. The various concepts have suffered technical problems, undesirable operational characteristics, or economic shortcomings, or combinations of these faults.

The "requirements/technology dilemma" has also impeded the development of military VTOL capabilities. The Department of Defense has been slow to establish requirements for VTOL aircraft because of a lack of demonstrated VTOL mission capabilities. Conversely, industry has been reluctant to develop the needed technology without the military requirements. The dilemma even more strongly affects the development of civil VTOL aircraft.

Although VTOL technology is still not adequate, concerted national goals can and should be formulated to guide its development.

C. The NASA Proof-Of-Concept Program

It is in the national interest to proceed with VTOL development with all possible expediency. The NASA can readily perform two of the most essential steps in accomplishment of this objective: (1) it can develop and evaluate VTOL technology. (2) It can determine and demonstrate the capabilities and limitations of the promising VTOL types, thereby permitting operational specifications and requirements to be realistically prepared. In short, the requirements/technology dilemma is resolved. Once these steps have been taken, industry, military, and civil planners can move swiftly to develop the needed operational aircraft for introduction into VTOL transportation systems.

NASA efforts can be most effectively focused in a proof-of-concept program. A proof-of-concept program should be conducted for each of the most promising VTOL concepts to provide and verify technology and to determine the suitability of the concept to fill the role of the civil and military VTOL aircraft. A proof-of-concept flight research aircraft should be tested to obtain the necessary data to determine the suitability of the concept for future development and service.

The program should include testing to establish proof of concept in three specific areas:

- <u>Technical Proof of Concept</u>

  The aircraft's performance, flight characteristics and problem areas would be investigated and evaluated to determine if technology is in hand for the successful development of an operational aircraft with the desired technical characteristics.

- <u>Economic Proof of Concept</u>

  The ability of the aircraft to perform a variety of possible missions would be investigated. Payload/lift capability, range, endurance and fuel consumption would be measured. These data along with operation analysis inputs would be used to determine economic feasibility of the concept. The cost effectiveness of the concept and its competitive position with alternate means for accomplishing specific missions could then be established.

- <u>Environmental Proof of Concept</u>

  The desirability of using the concept for particular civil and military missions would be evaluated to determine if the aircraft is a "good neighbor" and to determine its suitability for operation in battlefield environments. This would provide data on flight safety and safety to ground personnel, internal and external noise levels, and the effects of downwash and recirculation on engine injestion, visibility and dust signature. Approach and landing procedures would be evaluated for operation from airports and heliports as well as undeveloped areas to provide data for the development of navigational systems and to determine real estate requirements for future VTOL ports and to permit definition of low-speed maneuver requirements of military aircraft.

The proof-of-concept approach offers a way for this nation to make up for lost time and acquire the VTOL transportation systems that are so urgently needed.

D. The Tilt-Proprotor Aircraft - A Promising VTOL Concept

Of all of the VTOL concepts that have been investigated, one appears to offer great promise for an effective transportation system. This concept, the tilt-proprotor, has the characteristics necessary for economic feasibility and social acceptability in both civil and military roles. Economic and operational effectiveness require a high-payload vertical-lift capability, as well as high-speed cruise efficiency. Attempts to achieve these objectives with configurations that rely on high-disc-loading devices for lift have been generally unsuccessful. Recently, interest in low-disc-loading VTOL has been renewed, largely because of continued technological progress and the demonstrated operational usefulness of the helicopter.

Studies have shown that rotor-lifted (low-disc-loading) VTOL aircraft can have good cost effectiveness. Of the several concepts for low-disc-loading VTOL, the tilt-proprotor has been shown to hold the most promise for transport missions and missions requiring extended hovering. Comparative operations analyses conducted by Bell Helicopter Company, Reference 14, have shown the significant advantages inherent in the tilt-proprotor aircraft. In other studies, Bell, Westland, Lockheed, Sikorsky, Boeing, and the Marine Corps have reached essentially the same conclusion (References 2, 10 and 15 through 20).

The proprotor is an efficient lifting device that makes it possible to control hovering position precisely and to maneuver at low speed in confined areas. In high-speed cruise it functions efficiently as an airplane propeller. It has the added advantage over some other concepts of requiring only one propulsion system for both flight modes. Since there is no weight penalty for duplicate powerplants or propulsion devices, the ratio of payload to gross weight can be high. Consequently, the proprotor aircraft can realize high levels of mission effectiveness over distances much longer than helicopters or compound helicopters can fly. Arranging the proprotors side-by-side makes the overall span of the lift system large. In forward flight in the helicopter mode, this feature keeps power requirements low, and provides exceptionally good STOL characteristics. This latter capability could be used to advantage in many missions where the initial takeoff can be made from an airfield.

The low disc loading of the tilt proprotors will minimize noise and dust; the low-downwash velocities will contribute to the safety of ground personnel. The low noise level should make the proprotor aircraft one of the most acceptable VTOL aircraft for operation over and in populated areas.

Flight safety is enhanced by the simple conversion process which may be stopped or reversed at any point. The aircraft may be flown continuously with the proprotors at any conversion angle from vertical to horizontal. Its control characteristics make it safe to maneuver, climb, or descend during conversion or reconversion. Because its rotor-lifted speed range overlaps its wing-lifted speed range, the conversion corridor is wide, and such parameters as airspeed or power need not be scheduled with conversion angle. Steep descents are possible: there are no restrictions on the approach angle. In the event of complete loss of power, a conventional helicopter-like autorotational flare and landing is possible. The power-off reconversion capability, which was demonstrated with the XV-3 Convertiplane, makes it possible to enter autorotation from any flight mode. At airspeeds of 150 knots and above, power-off reconversions can be performed without loss of altitude.

The technology is sufficiently well developed that the tilt-proprotor aircraft's capability to meet the requirements for civil and military transportation can be demonstrated.

While proprotor technology has recently been advancing rapidly, the concept is not new; its technological development has been under way for more than twenty years. Efforts to develop the technology date back to the 1940s and this early work led to the initiation of the joint Air Force-Army XV-3 Convertiplane program in 1951 (Figure II-1). The flight evaluation of the XV-3 by Army, Air Force, and NASA pilots demonstrated the soundness and safety of the conversion principle and showed that a proprotor could be used equally well for lift and propulsion. These tests also defined dynamic stability problems that required further analysis and correction.

The evaluation tests were completed in 1961 and reported in References 21 to 23. The program included more than 375 hours of wind-tunnel and ground-run time, and more than 250 test flights in 125 hours of flight time. The test aircraft was flown by ten Government test pilots and two Bell pilots, who made a total of more than 110 full conversions. Five of the Government test pilots made power-off reconversions from cruise to helicopter autorotation after simulated engine failure.

Much of the recent proprotor work has been under government sponsorship as part of the Army's Composite Aircraft Program. These efforts have been concentrated on resolving the technical problems uncovered during the XV-3 tests, and on providing a technological base for the design and development of modern high-performance tilt-proprotor aircraft. The Composite Aircraft Exploratory Definition phase was completed in September

1967, with the design of the 28,000-pound, 360-knot, Bell Model 266 (Figure II-2). This work has been reported in detail in Reference 24 and in summary in Reference 25. The program to construct and test the demonstrator aircraft was not implemented, primarily because R&D funds were not available.

During the Composite Aircraft Program, design solutions for all the known proprotor problem areas were found, and the technical risk in a full-scale development appeared to have been minimized. Some uncertainty remains, however, since the design was substantiated only by analysis and the results of small-scale model tests. Large-scale model testing and/or a demonstrator aircraft program can finally prove the adequacy of proprotor technology for application to the design and development of such VTOL aircraft. Bell Helicopter Company has designed and is fabricating, as part of its IR&D program, a flightworthy 25-foot-diameter proprotor suitable for full-scale wind-tunnel testing.

The NASA-Ames Research Center has contracted with Bell for a wind-tunnel test program in which the 25-foot proprotor will be used to obtain data on performance and blade loads and to investigate dynamic stability of the rotor-pylon-wing system in airplane flight. These tests, which are scheduled for 1970, are illustrated in Figures II-3 and II-4. The first phase of this contracted work includes a design study of a proof-of-concept aircraft. This report presents the results of that study.

Under separate NASA Contract, Bell is conducting a design study of a more advanced tilt-proprotor concept, the folding proprotor (Figure II-5), which promises to extend operating speeds into the 400-to-500-knot range. The Air Force has also initiated a program (with several study contracts) to develop the technology for this advanced concept. The Bell-NASA folding-proprotor contract also calls for the design and fabrication of a 25-foot-diameter folding proprotor for full-scale wind-tunnel testing.

E. The Tilt-Proprotor Proof-Of-Concept Aircraft Evaluation Program

The proof-of-concept aircraft must be capable of undergoing a flight research program which will provide the verification as to whether or not the tilt-proprotor VTOL can meet the demands for future civil and military transportation. This means that the test aircraft and its test program must provide the data necessary to evaluate the economic and social acceptance of the tilt-proprotor aircraft as well as its technical characteristics. These proof-of-concept requirements are met in the tilt-proprotor aircraft design presented herein. It is believed that this aircraft, the Bell Model 300, can positively demonstrate the technical, operational, economic, and environmental suitability of the tilt-proprotor VTOL to fill the role of civil and military transports. The mockup shown in Figure II-6 is representative of a civil VTOL of the size and design of the Model 300 proof-of-concept aircraft.

The relatively small size of the aircraft permits it to be tested in the NASA-Ames Full-Scale Wind Tunnel and is a result of the desire to make the proof-of-concept program as economical as possible without scarificing any program objectives. In this respect the aircraft is a minimum size; however, test results and conclusions will be applicable to the largest VTOL transports envisioned at this time. The aircraft can operate in level flight to speeds of 300 knots and dive to higher speed to investigate aircraft dynamics. The Model 300 aircraft has been designed efficiently and configured so that it can show economic proof of concept by demonstrating the capability to perform a variety of civil and military missions.

The three view of the aircraft in Section X shows the passenger accommodations which could be provided for such missions. Typical missions which could be simulated by this aircraft are depicted in Figures II-7 and II-8.

The aircraft can investigate the effects of noise and downwash during takeoff and landings at a disc loading from 7 to 12-1/2 pounds per square foot. The low noise signature of tilt-proprotor aircraft will be typified by the Model 300. Section VII of this report presents the predicted noise level.

A noise pressure level of 90 PNdb would be experienced 300 feet from the hovering aircraft. This compares with a noise level range of 80-100 PNdb for automobile and truck traffic noise at 50 feet from a busy downtown street.

Takeoff, approach and landing evaluations can be conducted in undeveloped areas, heliports and airports to evaluate the effect of flight procedure on the noise and dust generated and to determine the terminal navigation and traffic control system requirements for VTOLs. Because the Model 300 has good STOL performance capability, operations in this mode can also be investigated. A proof-of-concept flight research program with the Model 300 tilt-proprotor aircraft will define requirements and establish technology, thereby paving the way toward realizing the VTOL transportation systems so urgently needed.

Figure II-1. XV-3 Convertiplane.

Figure II-2. Operational Version of Model 266 Composite Aircraft.

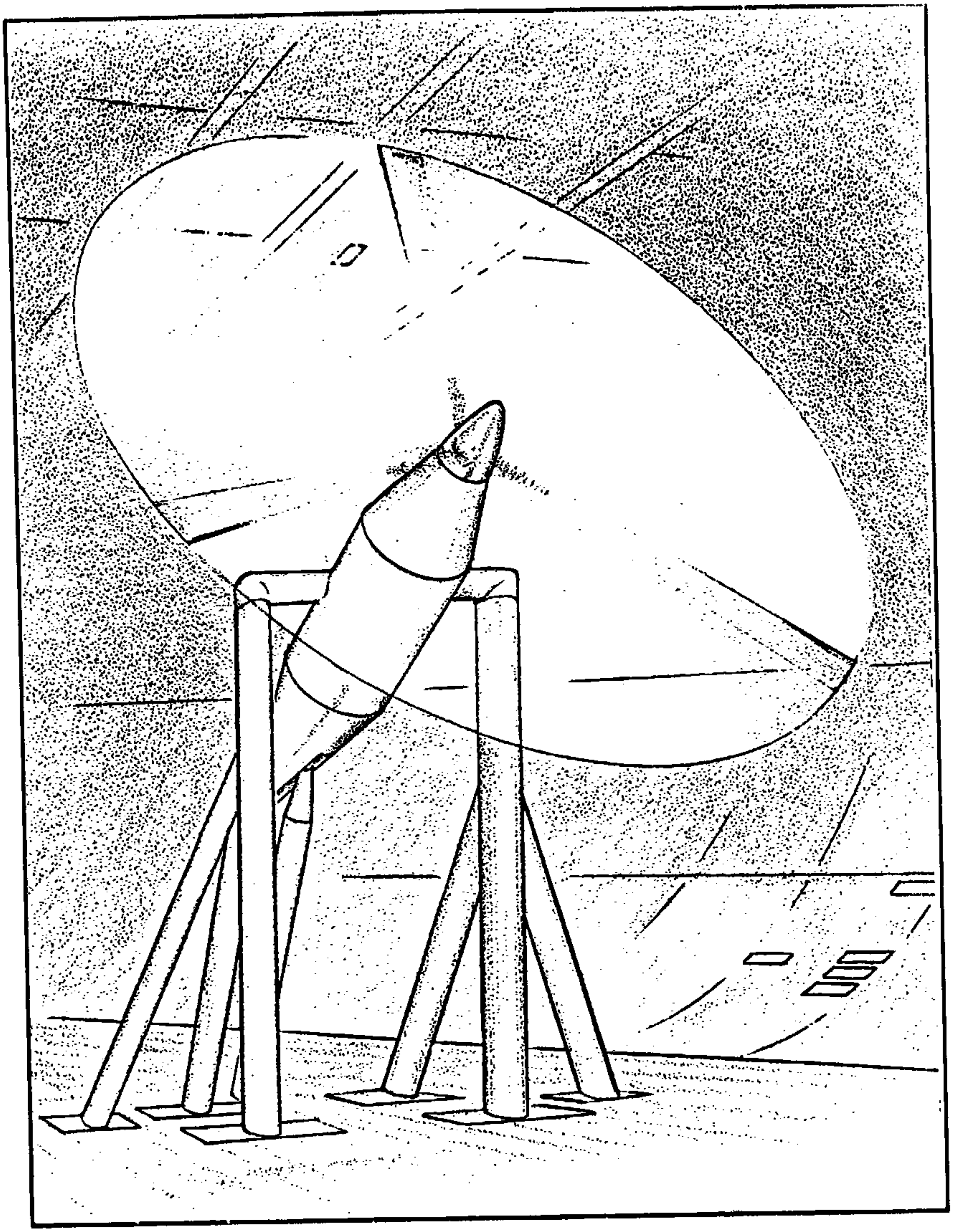


Figure II-3. 25-Foot Proprotor Performance Test in the NASA-Ames 40-by-80-Foot Full-Scale Tunnel.

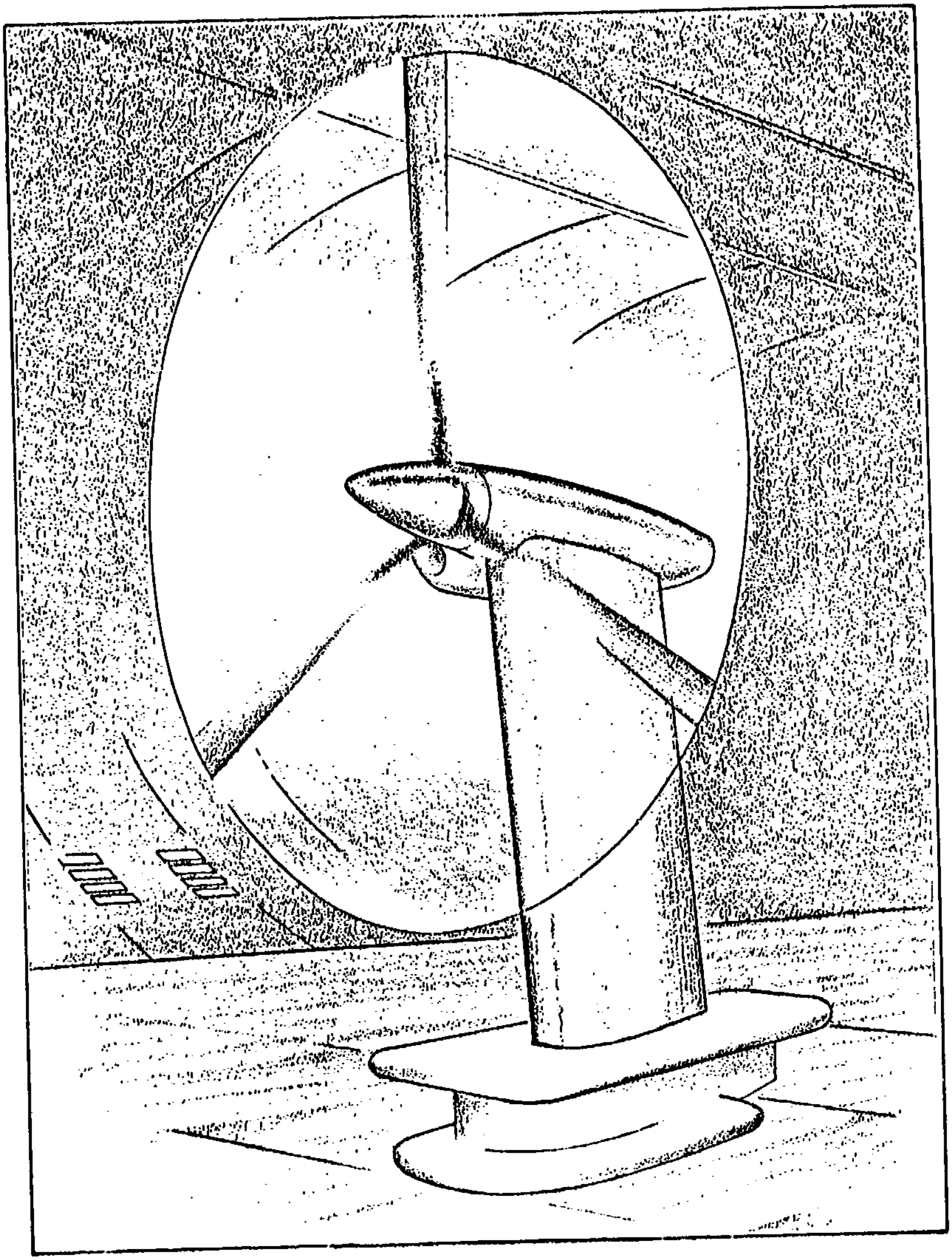


Figure II-4. 25-Foot Proprotor Dynamic Test on Simulated Wing in the NASA 40-by-80-Foot Full-Scale Tunnel.

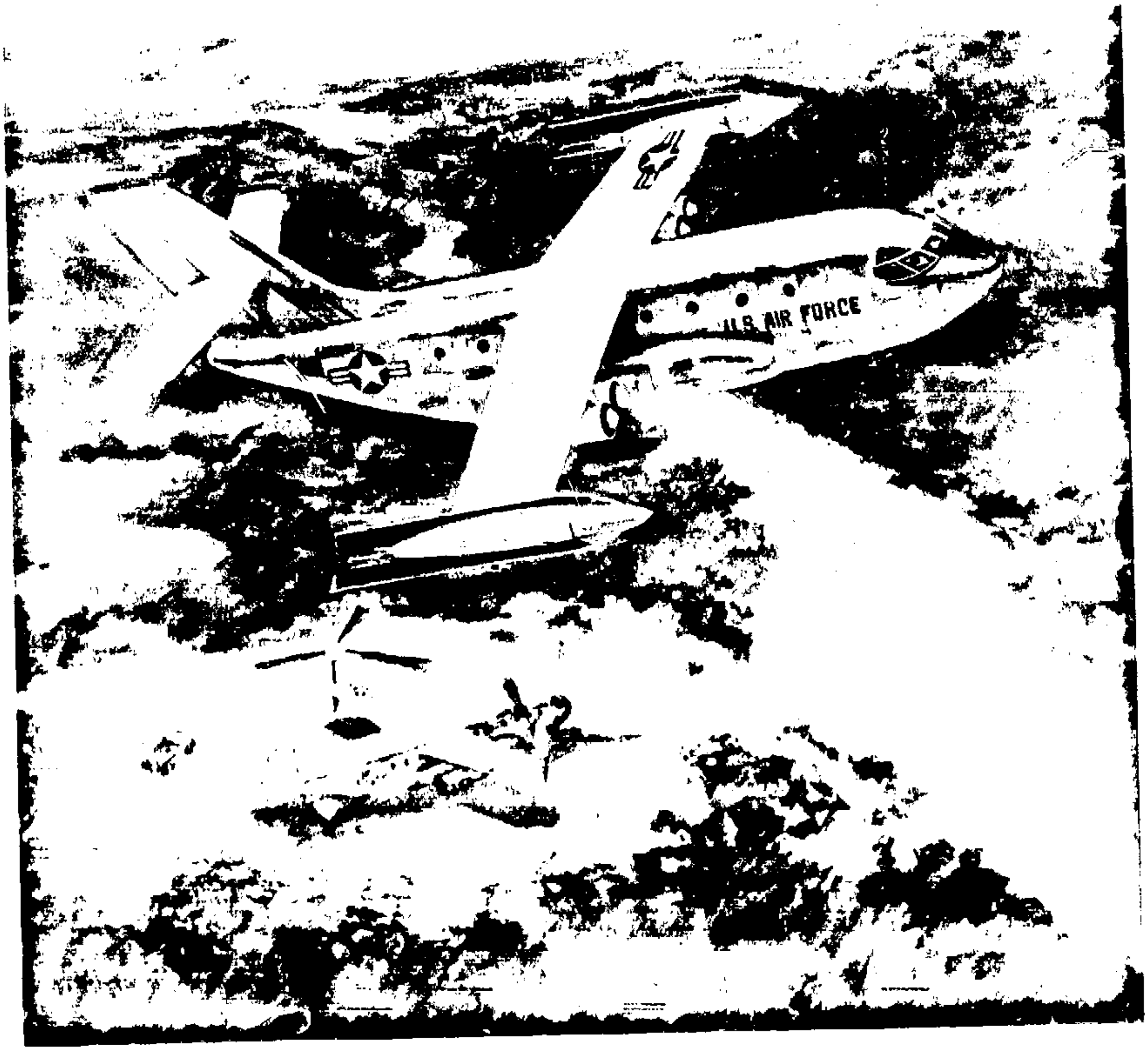


Figure II-5. Folding Proprotor VTOL Aircraft.

Figure II-6. Mockup of Representative Civil Tilt-Proprotor Aircraft.

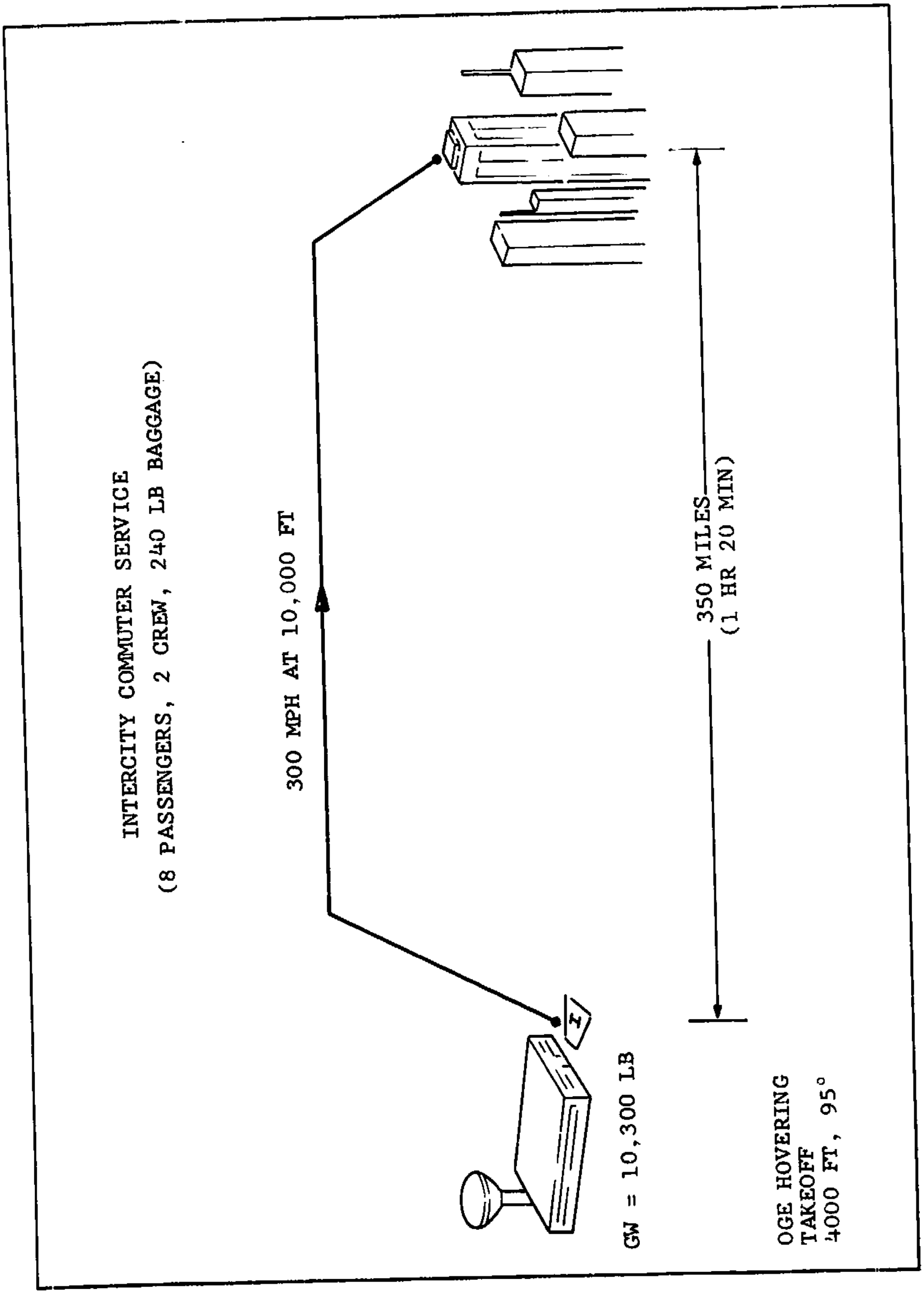


Figure II-7. Civil VTOL Aircraft Mission.

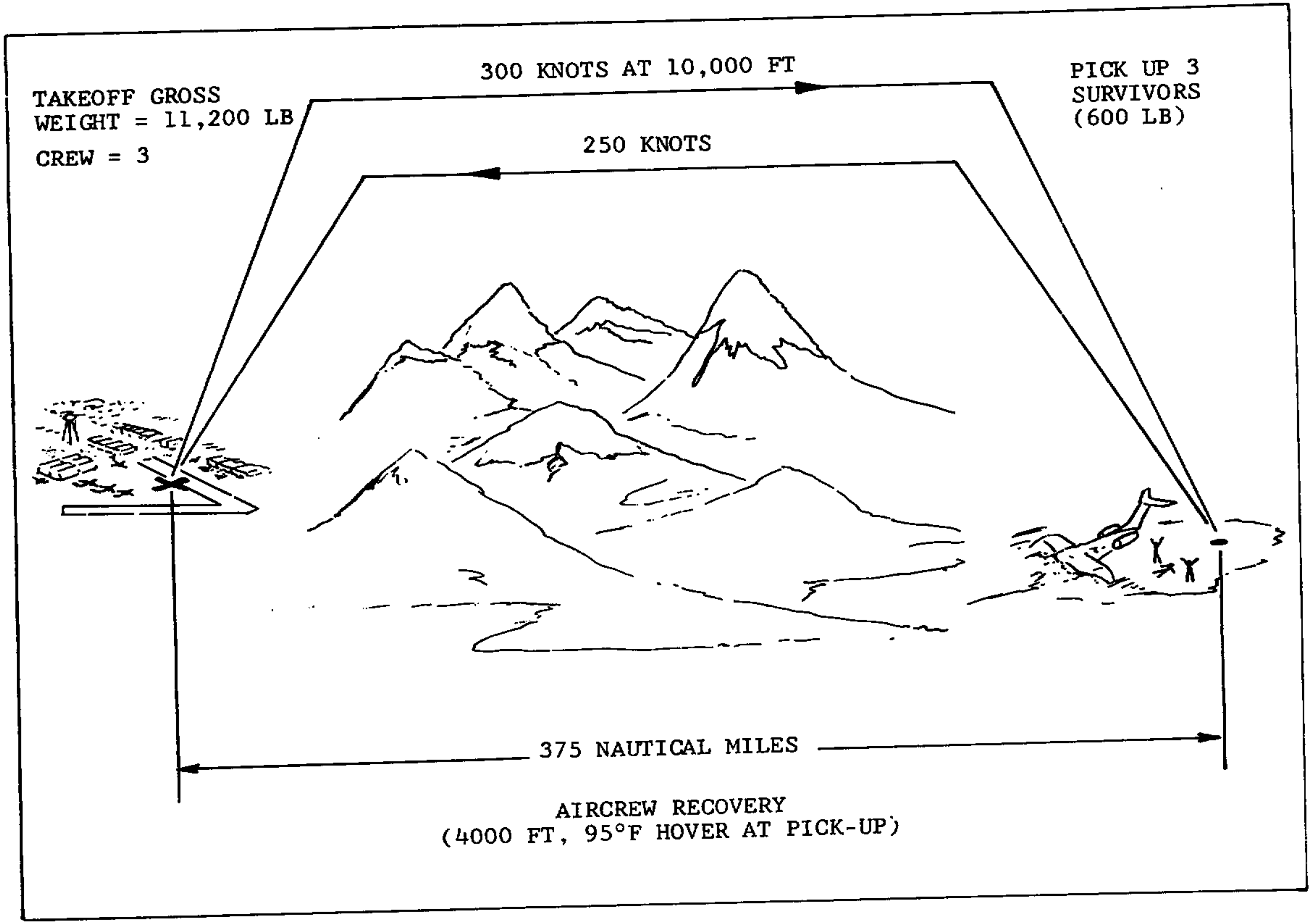


Figure II-8. Military VTOL Aircraft Mission.

III. DESIGN DESCRIPTION

A. General

The Model 300 tilt-proprotor proof-of-concept aircraft has twin proprotors at the tips of a forward swept high wing. The proprotors are mounted with gearboxes and turboshaft engines in self-contained propulsion system pods. The aircraft uses two 1150-horsepower Pratt and Whitney PT6C-40 direct-drive engines. General arrangement of the aircraft is shown in the three view in Section X.

The three-bladed 25-foot-diameter proprotors are gimbal mounted with hub springs to increase longitudinal control power in helicopter mode. The proprotors are identical in design to the proprotor to be tested early in 1970 in the NASA-Ames full-scale tunnel under Task II of this program. Disc loading is 9.7 pounds per square foot at the normal gross weight of 9500 pounds and 12.6 at the maximum gross weight of 12,400 pounds. At normal weight the wing loading is 54 pounds per square foot.

The cockpit is arranged for a crew of two and can be flown from either seat. Conversion and power management procedures are simple and straightforward and permit the aircraft to be flown by a single pilot. Power is controlled in the helicopter mode by a collective stick and twist grip throttles and in airplane mode by throttle levers and a proprotor governor. Conversion is controlled by fore and aft movement of switches on the pilot and copilot cyclic control sticks.

The canopy and forward fuselage are designed for installation of Douglas Escapac 1-D ejection seats for the research flight tests. The cabin is large enough to accommodate eight passengers in commercial seating or twelve troops in a high-density seating arrangement.

The aircraft is designed for a 2.0 g load factor in helicopter and conversion mode and 3.5 g's in airplane mode. Design limit dive speed is 350 knots. Basic design criteria are summarized in Section B. Basic data are summarized in Table III-1, dimensional data in Table III-2 and control travels in Table III-3.

The following design layouts are included in Section X of this report. Throughout the text of this section, where reference is made to these drawings, drawing number will be shown in parentheses.

300-960-001	Three View
300-960-002	Proprotor and Controls
300-010-001	Blade Assembly
300-010-100	Proprotor Assembly
300-960-003	Inboard Profile - Nacelle
300-960-004	Main Transmission
300-960-005	Fixed Controls - Wing

300-960-006	Fixed Controls - Fuselage
300-960-007	Wing Assembly
300-960-008	Fuselage and Empennage
300-960-009	Crew Station and General Arrangement

TABLE III-1

BASIC DATA

Aircraft Weight	
Normal Gross Weight	9500 lb
Maximum Gross Weight	12400 lb
Empty Weight	6876 lb
Design Landing Weight	9500 lb
Engine (Two)	
Manufacturer	Pratt and Whitney
Model	PT6C-40
30-Minute Rating (2 x 1150)	2300 hp
Maximum Continuous Rating (2 x 995)	1990 hp
Power Loading at Normal Gross Weight	4.1 lb/hp
Power Loading at Maximum Gross Weight	5.4 lb/hp
Proprotor (Two)	
Diameter	25 ft
Number of Blades Per Rotor	3
Solidity	0.089
Disc Loading at Normal Gross Weight	9.67 lb/sq ft
Disc Loading at Maximum Gross Weight	12.63 lb/sq ft
Wing	
Span	34.2 ft
Area	176 sq ft
Aspect Ratio	6.6
Wing Loading at Maximum Gross Weight	70.5 lb/sq ft

TABLE III-2

DIMENSIONAL DATA

Aircraft Dimensions	
Overall Length (41.8 feet)	501.0 in
Overall Width (Proprotor Turning) (57.2 feet)	686.0 in
Overall Width (Proprotors Removed) (36.4 feet)	436.0 in
Overall Height Pylons Vertical (Top of Spinner - From Static GL) at NGW (15.32 feet)	185.0 in
Overall Height (Top of Fin - From Static GL) at NGW (15.7 feet)	188.0 in
Span Between Proprotor Centerlines at Conversion Pivot Points (32.2 feet)	386.0 in
Static Ground Line Reference at WL	11.0
Height of Conversion Pivot Point Above Static GL at NGW (7.42 feet)	89.0 in
Conversion Axis Location, Percent Wing MAC	39.0
Distance from Conversion Pivot Point	
To Horizontal Tail 1/4-Chord of MAC (20.5 feet)	247.1 in
To Vertical Tail 1/4-Chord of MAC (19.3 feet)	231.7 in
Distance from Wing 1/4-Chord of MAC	
To Conversion Axis	8.7 in
To Horizontal Tail 1/4-Chord of MAC (21.3 feet)	255.8 in
To Vertical Tail 1/4-Chord of MAC (20.03 feet)	240.4 in
Ground Clearance at NGW (GL at WL 11.0)	12.0 in
Main Gear Tread Width	110.0 in
Distance from Nose-Wheel Axle to Main-Gear Axles	214.0 in
Engine	
30-Minute Rating	
Horsepower	1150 shp
RPM (Output Shaft)	30000 in-lb
Torque	2420 in-lb

TABLE III-2 - Continued

Maximum Continuous Rating	
Horsepower	995 shp
RPM (Output)	30000
Torque	2090 in-lb
Dry Weight	325 lb
Drive System Gear Ratios	
Engine to Proprotor	53.1:1
Engine to Interconnect Shaft	4.63:1
Proprotor	
Number of Blades per Proprotor	3
Diameter	25.0 ft
Disc Area per Proprotor	491 sq ft
Blade Chord	14 in. basic blade 17 in. cuff root at 0.0875R Tapering to 14 in. at 0.25R
Blade Area (3 blades)	43.75 sq ft
Solidity	0.089
Blade Airfoil Section	
Root (C_L Mast)	NACA 64-935 a = 0.3
Tip	NACA 64-208 a = 0.3
Blade Twist (See Figure III-1 for Distribution)	-45.0 deg
Hub Precone Angle	+2.5 deg
δ_3	-15.0 deg
Underslinging	0 deg
Mast Moment Spring (per Rotor)	2700 in lb/deg
Flapping Design Clearance	±12.0 deg
Blade Flapping Inertia (per Blade)	105 Slug ft^2
Blade Lock Number	3.83
Direction of Rotation, Inboard Tip Motion	Aft/Up
Pylon and Conversion Actuator	
Point of Intersection of Mast and Conversion Axes	
FS	300.0
WL	100.0
BL	193.0
Conversion Axis Wing Chord Location	39.0 percent MAC
Conversion Axis Forward Sweep	5.5 deg
Conversion Axis Dihedral (Up)	3 deg

TABLE III-2 - Continued

Angle of Mast Axis to Conversion Axis	95.5 deg
Angle of Outboard Tilt of Mast Axis	
Helicopter Mode	2.5 deg
Airplane Mode	0 deg
Distance Rotor Flapping Axis to Conversion Axis	56.0 in
Conversion Range (Pylon Vertical = 0)	-5.0 to + 90 deg
Actuator Length	
Extended	39.39 in
Retracted	10.00 in
Travel	29.39 in
Distance Engine C_L from Mast C_L	17.0 in
Wing	
Span (34.2 feet)	410.0 in
Span Between Conversion Axis Pivot Points	386.0 in
Area (Total)	176.0 sq ft
Root Chord (BL 28.0)	62.0 in
Tip Chord (BL 205.0)	62.0 in
Mean Aerodynamic Chord	
Chord (BL 102.75)	62.0 in
Leading Edge at FS	275.8
1/4 Chord at FS	291.3
Airfoil Section (Constant)	NACA 64A223 Modified
Aspect Ratio	6.63
Forward Sweep	6.5 deg
Dihedral	2.167 deg
Angle of Incidence	3.0 deg
Wing Twist	0 deg
Aileron	
Area/Side (Aft of Hinge Line)	10.4 sq ft
Span (Along Hinge Line) (8.04 feet)	96.5 in
Chord/Wing Chord	0.25
Flap	
Area/Side (Aft of Hinge Line)	5.5 sq ft
Span (Along Hinge Line)(4.25 feet)	51 in
Chord/Wing Chord	0.25
Wing Loading	
Normal Gross Weight	54 lb/sq ft
Maximum Gross Weight	70.5 lb/sq ft

TABLE III-2 - Continued

Fuselage	
Length (38.1 feet)	457.5 in
Maximum Breadth	66.0 in
Maximum Depth	74 in
Cabin Length (Cockpit Plus Cargo Compartment) (15.25 feet)	183 in
Cargo Compartment Length (9.16 feet)	110 in
Cargo Compartment Width	
Maximum	60.0 in
Floor Line	48.0 in
Cargo Compartment Height	
Ahead of Wing	60.0 in
Under Wing	54.0 in
Cargo Floor Space (9.16 feet x 4 feet)	36.6 sq ft
Cargo Compartment Volume (9.16 feet 4 feet x 4.75 feet)	174 cu ft
Vertical Tail	
Span (10.33 feet)	124.0 in
Total Area	57.8 sq ft
Rudder Area (Aft of Hinge)	7.6 sq ft
Rudder Chord/Total Chord	0.15
Aspect Ratio	1.84
Sweep of 1/4 Chord	36.83 deg
Root Chord at WL 75.0	97.0 in
Airfoil Section	NACA 64A015
Tip Chord at WL 199.0	37.34 in
Airfoil Section	NACA 64A015
MAC Chord (WL 127.8)	71.6 in
MAC Leading Edge at FS	513.8
MAC 1/4-Chord at FS	531.7
Horizontal Tail	
Total Area	62.5 sq ft
Span (16.0 feet)	192.0 in
Aspect Ratio	4.1
Angle of Incidence	0 deg
Elevator Area (Aft of Hinge)	17.6 sq ft
Elevator Chord/Total Chord	0.30
Root Chord (BL 0)	56.0 in
Airfoil Section	NACA 64A015
Tip Chord (BL 96.0)	37.75 in
Airfoil Section	NACA 64A015
MAC Chord (BL 44.88)	47.46
MAC Leading Edge at FS	535.2
MAC 1/4-Chord at FS	547.1
Sweep of 1/4-Chordline	22.72 deg

TABLE III-2 - Continued

Main Gear	
Number of Wheels per Side	1
Tire Size, Type and Ply Rating	8.50 x 10, Type III, 10-ply
Inflation Pressure	70 psi
Nominal Outside Diameter	25.2 in
Load Rating (Helicopter)	9200 lb
Flat-Tire Radius	7.0
Maximum Ground Speed	80 kt
Oleo Strut Stroke (Total)	10.0 in
Nose Gear	
Number of Wheels	2
Wheel Spacing (Dual)	9.5 in
Tire Size, Type and Ply Rating	5.00 x 5, Type III, 6-ply
Inflation Pressure	49 psi
Nominal Outside Diameter	13.9 in
Load Rating (Helicopter)	2100 lb
Flat Tire Radius	3.8 in
Maximum Ground Speed	80 kt
Oleo Strut Stroke (Total)	9.0 in

TABLE III-3

CONTROL TRAVELS

Cockpit Controls	
Cyclic Stick Fore and Aft	±6.0 in
Cyclic Stick Lateral	±6.0 in
Collective Stick	12.0 in
Rudder Pedals	±2.5 in
Fedal Adjustment	±2.0 in
Proprotor Controls	
Collective Pitch at 0.75R	
Helicopter	-2, +18 deg
Conversion	See Figure III-7
Airplane	+18, +50 deg
Differential Collective Pitch (Lateral Cyclic Stick)	
Helicopter	±3.0 deg
Conversion	See Figure III-8
Airplane	±0.3 deg

TABLE III-3 - Continued

Collective Pitch Trim	
Helicopter	±0.5 deg
Conversion	±0.5 deg
Airplane	±0.5 deg
Cyclic Pitch Total	±4.0 deg
Fore and Aft Cyclic Pitch	
Helicopter	±10.0 deg
Conversion	See Figure III-9
Airplane	0
Differential Cyclic Pitch (Rudder Pedals)	
Helicopter (0-60 kt EAS)	±4.0 deg
(60-100 kt EAS)	See Figure III-11
(100 kt EAS +)	±1.0 deg
Conversion	See Figure III-10
Airplane	0 deg
Fixed Surfaces	
Aileron	
Flaps Up 10 deg	+15.0 -15.0 deg
Flaps at 0 deg	+21.6 -15.2 deg
Flaps down 30 deg	+18.0 -12.5 deg
Flaps down 60 deg	-14.2 - 5.0 deg
Elevator	±20.0 deg
Elevator Trim Tab	±20.0 deg
Rudder	±20.0 deg

B. Design Criteria

Criteria have been established to provide a safe and efficiently designed flight research aircraft. Basic criteria comply with the Federal Aviation Regulations. Design limits, load factors, and conditions have been established in accordance with the requirements of the FAA, "Tentative Airworthiness Standards for Verticraft/Powered Lift Transport Category Aircraft - Part XX," dated July 1968. In the areas where this document fails to provide adequate definition, the applicable requirements of the Federal Aviation Regulations for rotorcraft and airplanes were used as a guide. In the areas not covered by any of the regulations and/or where exceptions have been customarily granted, Bell design practice for helicopters has been used. The basic design criteria and design parameters for the Model 300 are given in Table III-4.

TABLE III-4

BASIC DESIGN CRITERIA

Normal Gross Weight	9500 lb	
Maximum Gross Weight	12400 lb	
Empty Weight	6876 lb	
Design Operating Speed, EAS		
Helicopter	140 kt	
Conversion	140-170 kt	
Airplane	260 kt	
Design Limit Speed, EAS		
Helicopter	156 kt	
Conversion	189 kt	
Airplane	350 kt	
Proprotor Maximum Operating Speed and RPM		
	Tip Speed (fps)	(rpm)
Helicopter	740	565
Conversion	700	534
Airplane	600	458
Limit Load Factors at 9500 Pounds		
Helicopter	2.0	
Conversion	2.0	
Airplane	3.5	
Transmission Design Power		
Helicopter	1060 hp	
Airplane	860 hp	
Conversion	1000 hp	
Single Engine	1150 hp	

C. Proprotor

The 25-foot-diameter proprotor which is currently being fabricated for full-scale wind-tunnel testing, is designed to flight-worthy standards and is appropriately sized for use on the Model 300 aircraft. Aerodynamic design parameters have been selected for efficient cruise in the 200- to 300-knot speed range. The same requirements for reliability, service life and maintenance as an operational helicopter were met in the detail design of this proprotor. Based on wind-tunnel test results, necessary design changes would be made prior to fabricating the proprotors for the research aircraft. The proprotor blades and hub are described below.

1. Blades

The blades (300-010-001) use type 17-7PH stainless steel as the basic blade material as a result of a design study in which the relative merits of aluminum, titanium and several types of stainless steel were considered. Results indicate a substantial weight savings for both steel and titanium compared with aluminum blade designs. The 17-7PH steel blade provided the desired natural frequencies and strength for minimum weight.

Thickness, taper, twist and camber distributions were selected to meet the varying structural and aerodynamic requirements for helicopter and airplane flight. NASA 64-series airfoils are used with a 64-208 at the tip and a 64-935 at the theoretical root (blade Station 0). The thick blade root section is required to provide adequate blade strength when the blade is at high pitch in airplane flight where torque and inplane gust loading cause high bending moments about the airfoil chord line.

The basic chord of the blade is 14 inches. Chord, twist, lift coefficient and thickness distributions are shown in Figure III-1. Blade stiffness and mass distribution are shown in Figure III-2.

2. Hub

The hub (300-010-100) consists of a titanium yoke with three spindles and a universal joint assembly that is splined to the mast. A nonrotating, elastometric hub-moment spring is attached to the yoke through a bearing. The lower end of the hub-moment spring is attached to the transmission case by studs.

The universal joint assembly consists of a steel cross with bearings mounted in aluminum pillow blocks on two opposing spindles and a steel fork with bearings on the other two spindles. These four roller bearings are not provided with inner races, but roll on the case-hardened journals of the steel cross member. A common oil reservoir is created by oil passages drilled within the cross member. Oil level sight gages are installed on the pillow block housings. The bearing housings contain thrust bearings to carry the proprotor H-forces and seals to retain the oil.

The inboard and outboard pitch change roller bearings assemble in the blade's integral root fitting. The inner race of these bearings assemble on the spindles of the yoke. A stainless steel liner is bonded to the spindle to prevent fretting between the inner race and the titanium spindle. The pitch-change bearings are oil lubricated from a reservoir located in the pitch horn.

The three wire-wound blade retention straps have an integral steel fitting which seats at the inboard end of each spindle of the yoke. The outboard fitting, of the retention strap, is attached to the blade by a steel bolt through the blade root fitting, spar and doublers.

D. Drive System

The drive system consists of a main transmission assembly (300-960-004) at each wingtip, a system of drive shafting through the wings connecting the two main transmissions, and a center gearbox mounted inside the fuselage (300-960-007). The PT6C-40 engine attaches directly to the transmission pylon case. Each transmission is attached to a steel spindle which is supported by the two outboard wing ribs. Hydraulically-powered and mechanically-interconnected Acme screw actuators support, power, and control the conversion of the pylon assembly about the transmission-spindle axis. In the airplane mode the actuators drive the pylon into a down stop supported by the tip rib.

In normal operation, each transmission delivers power to its proprotor from its own engine. The interconnecting shafts in the wings operate unloaded, except during maneuvers, single-engine operation, or asymmetrical loading conditions, where the interconnect driveshaft distributes power as required.

Design power for the transmission is shown in Table III-4 and is based on the same design torque of each mode of flight with both engines operating. To permit the use of maximum power from the remaining engine in the event of an engine failure, the engine output shafting and herringbone gear stage are designed for the maximum engine output power of 1150 horsepower. Several factors are multiplied by the design power and torque to arrive at limit and ultimate torques for the various stages. A distribution factor of 1.10 is applied to obtain the maximum steady power which allows for an uneven distribution of power between the proprotors. A transient torque factor of 1.67 is then applied to obtain limit torque during asymmetric maneuvers. Ultimate torque is 1.5 times limit torque.

The main transmission assembly supports all pylon components. The structural parts of the assembly consists of a spindle, pylon case, intermediate case, and a top (mast) case. The engine and pylon cowlings are also supported by the transmission.

Power is transmitted from the engine by an adapter shaft which picks up the female spline of the PT6C-40 power turbine shaft, then through a combination power and torquemeter shaft which is splined to a herringbone pinion. The herringbone stage of reduction gears transmits the power through a one-way clutch to the two planetary reduction units. Power is supplied to the rotor masts by the planet carrier of the upper planetary stage.

The interconnect power train, linked to the main proprotor drive side of the one-way clutch, consists of a spur gear set, an intermediate shaft (with torquemeter shaft), and a spiral bevel gear set.

The accessory gears provided for:

- Hydraulic pump
- Transmission oil pump
- AC generator
- N_{II} governor

The center gearbox, with splash lubricated bevel gears, is mounted on the rear spar of the wing at the centerline of the fuselage. This gearbox accommodates the change in interconnect shaft angle due to wing sweep. A magnetic sensor is installed to provide a signal to the rotor tach indicator and proprotor governor.

E. Powerplant

1. Engine

The Model 300 is powered by the Pratt and Whitney PT6C-40 free turbine turboshaft engine. This engine is an advanced version of the PT6 turboprop engine widely used in executive, third-level airline and utility turboprop aircraft. The PT6C-40 is a direct drive engine with an output speed of 30,000 rpm. The turboprop gearbox is removed and the lubrication system modified for vertical operation. The engine has takeoff and 30-minute ratings of 1150 horsepower and a maximum continuous power of 995 horsepower. The engine is rigidly mounted to the proprotor transmission case. Engine torque is read by a Simmonds magnetic torquemeter located on the main transmission.

2. Induction System

The engine induction system (300-960-003) is designed to give maximum total pressure at the engine inlet screen, to provide anti-icing, and to protect the engine from dust and sand ingestion. Air enters through the nacelle inlet and diffuser duct. A 90-degree turn into the engine plenum provides an effective inertial particle separator to remove dust and sand. A screen is provided which ices over during icing conditions to increase the separator efficiency.

Moisture and debris are removed from the primary engine air in the by-pass duct and ejector. By-pass air spills out of the by-pass duct to a plenum, then moves across the engine and transmission oil coolers to the engine accessory section. Air passes over the accessories from the coolers, through a bell-mouth and blower to provide a power source for the ejector. The ejector then emits the by-pass air, induction air contamination, and heat from oil coolers and engine accessory section.

3. Oil System

The engine is supplied with oil from a 2.3-gallon tank that is an integral part of the compressor inlet case on the engine. Oil flows from the tank to the accessory reduction gears, engine bearings and filter. Scavenge oil is directed through the oil cooler located behind (airplane mode) or below (helicopter mode) the accessory gear case. The oil cooler is equipped with a thermostatically regulated bypass to prevent high surge pressures during starts under cold weather conditions. Air for the coolers is provided by a mechanically driven blower. A shaft from the accessory drive pad on the engine provides power for the blower. The oil leaves the coolers and is returned to the tank forming a "cold tank system". The tank is vented overboard. Continuous indication of system operation is provided by oil temperature and pressure instruments in the cockpit. Warning lights are also provided to indicate low oil pressure and high oil temperature.

4. Fuel System

Fuel is supplied by two separate systems, one for each engine. Each system is composed of two cells interconnected to form a single tank in each wing, with a total fuel capacity of 1600 pounds. The cells are constructed of a flexible rip-resistant material. Continuous support for each cell is provided by the structural honeycomb panels of the wing. Gravity refueling is accomplished through filler caps in each of the inboard cells. One dc fuel-booster pump is provided at each inboard cell.

Engine fuel passes from the booster-pump discharge through a check valve, fuel filter, firewall shutoff and conversion-swivel fitting before entering the fuel control. A pressure gage in the cockpit indicates the discharge pressure of the booster pump. An interconnect between the two discharge lines permits one pump to supply both engines if a pump fails. Opening the tank interconnect valve will allow inter-tank gravity transfer of fuel to the operative pump.

F. Airframe

1. Wing

The following specific objectives were established for the wing design (300-960-007).

- Place the elastic axis far forward to minimize the torsional deflections resulting from coupled proprotor/pylon/wing motions.
- Provide high torsional stiffness without undue weight penalty.
- Provide the maximum possible flap and aileron area, and design them to deflect to a large angle, in order to minimize the projected wing area and hence the aircraft download, during hover.
- Provide an unobstructed passageway to route controls and the transmission interconnect.
- Provide fuel space

These objectives are accomplished by a forward location of the structural box and sweeping the wing forward 6.5-degrees to obtain the desired relationship between the wing center of pressure and the conversion axis. Fuel cells are located inside the structural box; the controls and the transmission interconnect shaft are located aft of the rear spar.

A lightweight, torsionally efficient structure is obtained by using sandwich construction for the skins of the structural box. All of the skin is effective in both bending and torsion, whereas with a plate-stringer combination the skins may be in a buckled state under load, and the stringers provide no torsional stiffness.

A significant factor in obtaining high torsional rigidity is the high wing-thickness ratio which provides a large wing-box cross-sectional area. Since the rigidity varies with the square of the area, high torsional stiffness results. In addition to contributing to the torsional rigidity, the high thickness ratio contributes to the wing bending stiffness. This in turn results in the low structural weight required to carry the high bending moments resulting from the lift being concentrated at the wing tips during vertical flight.

The aluminum alloy front and rear spars are designed partially by stiffness requirements and partially by structural requirements. The front spar is a shear-resistant web. The rear spar is honeycomb sandwich. The outboard two ribs of the wing support the conversion spindle and the tip rib supports the conversion-actuator spindle and pylon down stop. The intermediate ribs form the bulkheads for the extremities of the fuel cells and provide for redistribution of loads from the ailerons and flap hinge ribs.

The leading-edge structure is honeycomb sandwich, hinged for access to the conversion interconnect shaft.

EI's and GJ distribution and panel-point weights for the wing are shown in Figure III-3.

2. Fuselage

The fuselage (300-960-008) is a conventional nonpressurized, semi-monocoque structure of 2024 and 7075 aluminum alloy. The four main longerons run continuously above and below the cutouts required for doors and the landing gear. Stringers break up the skin panels to the required size. Major bulkheads are provided for the ejection seat rails at both sides of the entrance door, front and rear wing spars, at both ends of the landing gear bay, and for the vertical stabilizer spars. Nose landing gear support beams, at BL 7.75 extend between Stations 131 and 237.

The cabin extends between the canted bulkheads at Stations 219.8 and 347. The inside cross-sectional dimensions are 60-inches wide, 127-inches long and 60-inches high (54 inches under the wing). The entrance door opening, located at the forward left side, is 28-inches wide and 52-inches high. Emergency exits are provided on each side of the cockpit and in the cabin on the right side between Stations 319 and 347. The floor is an aluminum honeycomb sandwich with a rigidized upper surface.

EI's and GJ distribution and panel point weights for the fuselage are shown in Figure III-4.

3. Empennage

Three spars of the vertical stabilizer (300-960-008) attach to canted bulkheads of the fuselage. The horizontal stabilizer is located at approximately the lower third span of the vertical stabilizer and attach as to the front and center spar. Ribs and chordwise stiffeners to break up the skin panel are located between the three spars.

The structural box of the horizontal stabilizer (300-960-008) is a single-cell configuration consisting of two spars and surface coverings of honeycomb sandwich construction. Bulkheads are provided at elevator hinge points and at the intersection with the fin.

EI's, GJ, and mass distribution for the vertical and horizontal tail are shown in Figures III-5 and III-6.

4. Landing Gear

A fuselage mounted main gear was chosen because of the high-wing configuration. The gear retracts into the sides of the fuselage. Flush doors are provided between the bulkheads at Station 347 and 410. The gear geometry was developed to permit the gear to clear the lower longerons when it retracts. A dual-wheel nose gear retracts into the compartment between Stations 131 and 169. Shock-absorption system is a conventional air-oil oleo.

G. Aircraft Systems

1. Conversion System

The conversion system provides controlled rotation of the propulsion pod from the vertical to the horizontal position and return. It can safely lock the pylon in either extreme, or in any intermediate position. The system also serves as a reference for the control system by providing a phasing control motion as a function of flight regime. The conversion actuator holds the pylon against the down stop on the tip rib in airplane mode.

Conversion is controlled by a switch on the control stick grips. Forward movement of the switch rotates the pylons forward from helicopter to airplane flight position, and rearward switch movement returns them to the helicopter position. The conversion of the pylons may be stopped or reversed at any position. The normal conversion time is approximately nine seconds.

Should one conversion actuator fail to function due to hydraulic or electrical failures, that unit is driven by the actuator motor on the opposite wingtip through the mechanical interconnect shaft. In the event of a complete dc power failure, a mechanical backup system, operated by pulling the emergency reconversion T-handle located in the cockpit,-positions the hydraulic valves to cause the actuators to move the pylons to the helicopter position.

The major components (300-960-007) of the conversion system include the double screw conversion actuators with hydraulic motors and electrically powered servo-valves, the interconnect shafting and a control phasing gearbox located on the forward side of the front spar of the wing. The hydraulic motors that power the conversion actuators are controlled by dual, pilot-activated, three-position servo-valves which receive feedback information through a small gearbox on the interconnect shaft near each wingtip.

The control phasing gearbox, provides a linear output, proportional to the pylon angle, that phases the various fixed controls during conversion. This unit also provides the signal for the pylon-conversion-angle indicator in the cockpit. A conversion brake assembly is incorporated in the phasing gearbox which locks the pylon when the aircraft is on the ground with hydraulic power off.

2. Hydraulic System

The hydraulic system is a MIL-H-5440 Type II (-65°F to +275°F) system utilizing MIL-H-5606 fluid at an operating pressure of 1500 psi. It has two independent transmission-driven hydraulic pumps. Since the pumps are driven by the transmission, they operate whenever the proprotors are turning, and they are independent of engine power.

The primary hydraulic system, connected to one side of the dual flight-control actuators, is powered by the hydraulic pump in the left pylon. The utility system is powered by the pump in the right pylon. After retraction, the landing gear portion of the utility system is separated from the flight-control portion by an isolation valve. The system powered by the right transmission pump, isolated from any utility function, then becomes a second primary system for one side of the dual flight-control actuators. Dual or single power is provided to the hydraulically operated components as shown below:

Primary (Left Pump)	Utility (Right Pump)	Function
x	x	Cyclic
x	x	Collective
x	x	Ailerons
x	x	Proprotor Governor
x		Conversion Actuator Left
	x	Conversion Actuator Right
x	x	SCAS
	x	Proprotor Trim
	x	Landing gear

3. Electrical System

The electrical system consists of two 200-ampere, 28-volt dc starter generators, and two 13-ampere-hour batteries, providing primary dc power. Two 250-va, 115/200-volt, single-phase 400-Hertz ac inverters provide the ac power. Two essential dc busses are connected in parallel through a bus-tie relay. Each 200-ampere starter generator supplies power to one of the essential dc busses. Each essential dc bus has a 250-va inverter with its essential ac bus. There are two essential ac and dc busses, and one nonessential dc bus.

The electrical system is designed to provide complete dual ac and dc power sources. These sources and their essential and nonessential busses are designed for complete isolation of the sources and their busses in the event of any failure.

II. Aircraft Controls

1. Proprotor Controls

Proprotor controls (300-960-002) consist of a rise-and-fall collective head assembly above the proprotor and a monocyclic (fore and aft) swashplate below the proprotor.

The collective head is attached to the proprotor mast. A non-rotating tube, extending inside the mast to the collective boost cylinder, gives vertical motion to the rotating collective head. A collective lever is attached to each of the three trunnions of the collective head. A control tube extends from one end of each collective lever to a pitch horn. At the other end of each collective lever a tube goes to the rotating swashplate.

The rotating swashplate (outer) is driven by the lower ring of the proprotor spinner. The nonrotating swashplate is attached to the top case of the transmission and is free to tilt about only one axis. The cyclic cylinder is attached to the non-rotating swashplate (300-960-003).

Collective control inputs, which increase or decrease the pitch of all blades at the same time, are introduced by means of a tandem hydraulic cylinder which is attached to the transmission case below the mast (300-960-003). The servo-valve linkage of the collective cylinder receives its input from the pilot through a swivel joint, on the conversion axis, which connects to the fixed controls in the wing (300-960-007). The input motion is introduced along the conversion axis so that the collective system functions in the same way in both airplane and helicopter modes of operation, though with different ranges of collective pitch.

The cyclic control cylinder tilts the swashplate, which causes one-per-rev variations in blade pitch. The servo valve of the cyclic cylinder is actuated by the pilot through a linkage (300-960-003) which is automatically phased out as the pylon converts from vertical to horizontal. This phase out is accomplished by having the fixed controls in the wing (300-960-007) impart vertical motion to the end of the cyclic input tube that is located on the conversion axis. Axial motion in introduced when the input tube is vertical (helicopter mode). No axial motion occurs when the tube is horizontal (airplane mode)

The design of the control linkage boost cylinder permits the pilot to control the proprotors manually in the event of a double hydraulic system failure.

2. Flight Control

The flight-control system combines the basic elements of conventional helicopter and airplane control systems. The cockpit controls for the proprotors and control surfaces are arranged so that a single pilot can maintain full control in all flight regimes, including conversion. Each of the two crew stations (300-960-009) has complete controls for pitch, roll, yaw, and thrust in all modes of flight. They consist of control sticks, rudder pedals, and collective levers, for both the pilot and copilot; a single set of power-management controls and a flap control on the center pedestal, and a rotor trim control on each cyclic stick. Dual-twist grips on the collective levers are interconnected with the power-management controls on the center pedestal.

In helicopter mode, the controls apply blade-pitch changes to produce powerful control moments and forces. Fore-and-aft cyclic pitch provides longitudinal control, while differential-cyclic pitch produces directional control. Collective pitch is used for vertical flight and differential-collective pitch controls roll.

In airplane mode, the controls actuate conventional control surfaces which provide the control response characteristics of a conventional airplane. These control surfaces are also actuated in the helicopter flight mode, but they have minimal effectiveness because of the low dynamic pressures and high control moment capability of the proprotors.

Conversion or reconversion can be made within a wide range of variables such as airspeed, conversion angle, and fuselage attitude. Mechanical phasing of the proprotor control authority minimizes the need for control inputs during conversion. To provide the proper control authority during conversion (or reconversion), some controls are phased out, others are phased in, and the authority of others is altered. The automatic changes in controls as the pylon is converted from helicopter mode (-5 degrees to +15 degrees conversion angle) to airplane mode (+90 degrees conversion angle) is shown on Figures III-7 through III-11.

3. Stability and Control Augmentation System

A stabilization system is used to enhance the flying qualities in helicopter and conversion modes. The three-axis stability and control augmentation system (SCAS), uses rate gyros to sense pitch, roll and yaw. Electrical signals from pilots control motions as well as signals from the rate gyros are input into appropriate shaping networks of the SCAS. The result is an aircraft that is stable and well damped for external turbulences, yet is highly responsive to pilot control inputs. As a normal procedure, the pilot will engage the SCAS prior to takeoff and fly the aircraft in the helicopter and conversion modes where SCAS is phased out as the helicopter controls are phased out.

The SCAS actuates the rotor controls through servo actuators located on the collective and cyclic hydraulic cylinders. Redundancy is provided by dual actuators which are operated individually by the two hydraulic systems.

4. Proprotor Governor System

The proprotor governor system is used to simplify power management and rpm control and to prevent engine-power adjustments and external disturbances from changing proprotor rpm. The system is a closed-loop control system that maintains a pilot-selected proprotor rpm by controlling collective blade-pitch in the airplane mode.

The proprotor governor system detects any error between the command rpm and the actual proprotor rpm. This error signal is amplified and used to drive a hydraulic actuator in the collective control system. With a constant proprotor rpm setting, increasing power with the power management levers will increase the collective blade-pitch to hold a constant rpm. This will result in increased aircraft velocity without changing proprotor rpm. Decreasing power will decrease the collective blade pitch and reduce aircraft velocity.

The proprotor governor is a fail-operate type system. Sufficient redundancy and monitoring circuitry is included so that a single failure will not result in loss of the proprotor governor. If a failure occurs, a warning light will be illuminated. If a second failure occurs, the proprotor governor system will automatically shut off and the pilot will control proprotor rpm manually with the collective lever.

5. Power Management

Power management is simple and is designed for straightforward cockpit procedures. Power control is provided by two control systems. For helicopter flight, the engine power-turbine governors maintain selected proprotor rpm by increasing or decreasing power as manual changes are made in collective pitch. In airplane flight, the proprotor-pitch governor maintains selected rpm by increasing or decreasing collective pitch as manual power changes are made. Thus, the Model 300 may be flown in helicopter mode in the same manner as a conventional helicopter, and in airplane mode in the same manner as a conventional turboprop airplane.

Throttle and proprotor governor rpm select levers are mounted on the pedestal (300-960-009), convenient to both the pilot and copilot. The pilot and copilot collective sticks have dual twist grip throttles and turbine governor rpm beeper switches.

Conventional helicopter rpm droop compensation is provided for each engine. The collective stick is connected through the droop-compensator linkage to a droop cam on each engine. The droop cams position the load-signal shafts on the engine fuel control, which in turn schedule limited rpm changes to compensate for the engine droop characteristics.

Engine output power for both helicopter and airplane flight is regulated by the fuel control on each engine. Movement of the power-control shaft on each engine controls fuel flow. This shaft is positioned by the throttle levers. The throttle levers are controlled remotely by the twist grips during helicopter flight.

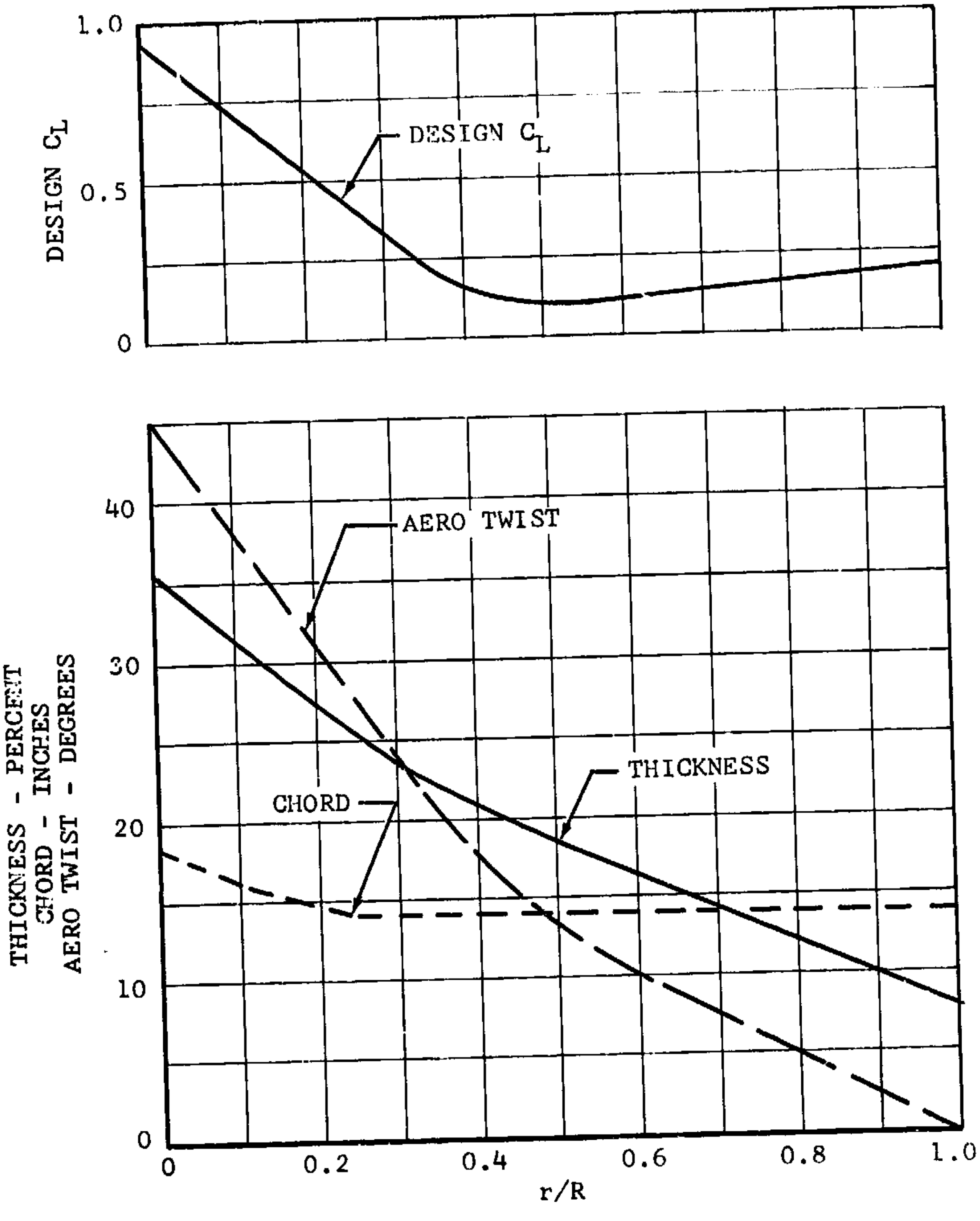


Figure III-1. 25-Foot Proprotor Parameters.

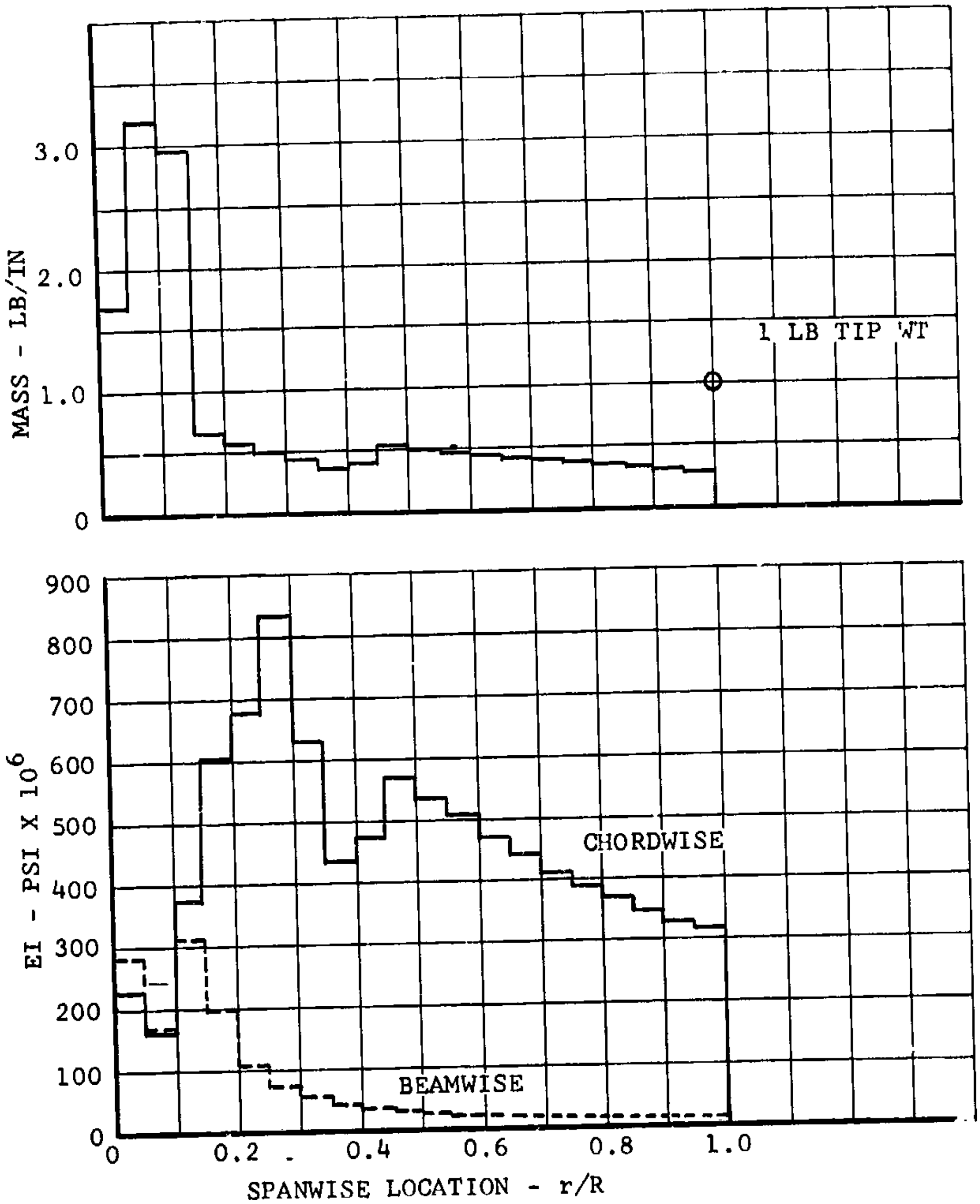


Figure III-2. Proprotor Stiffness and Mass Distribution.

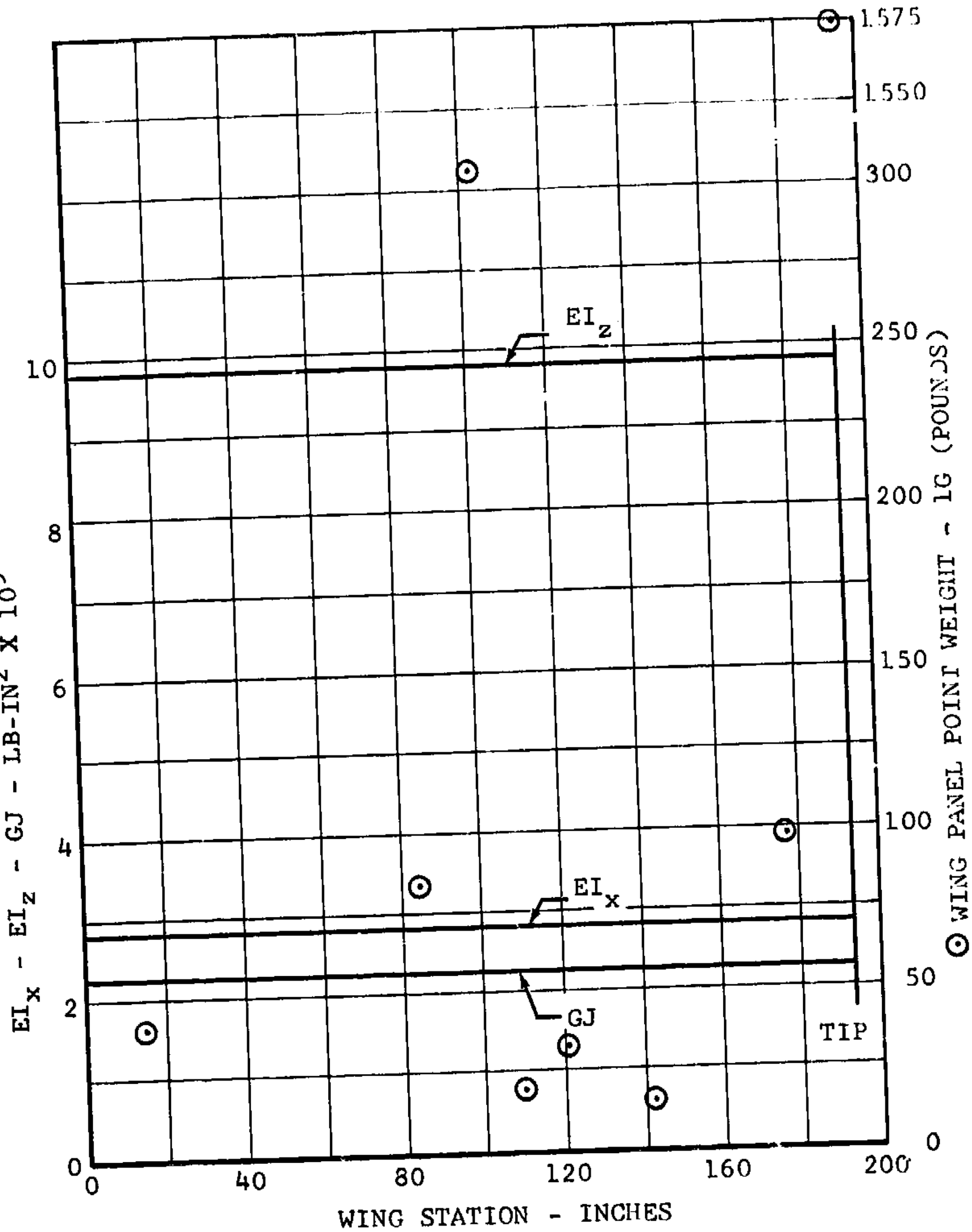


Figure III-3. Wing Stiffness and Panel Point Weight.

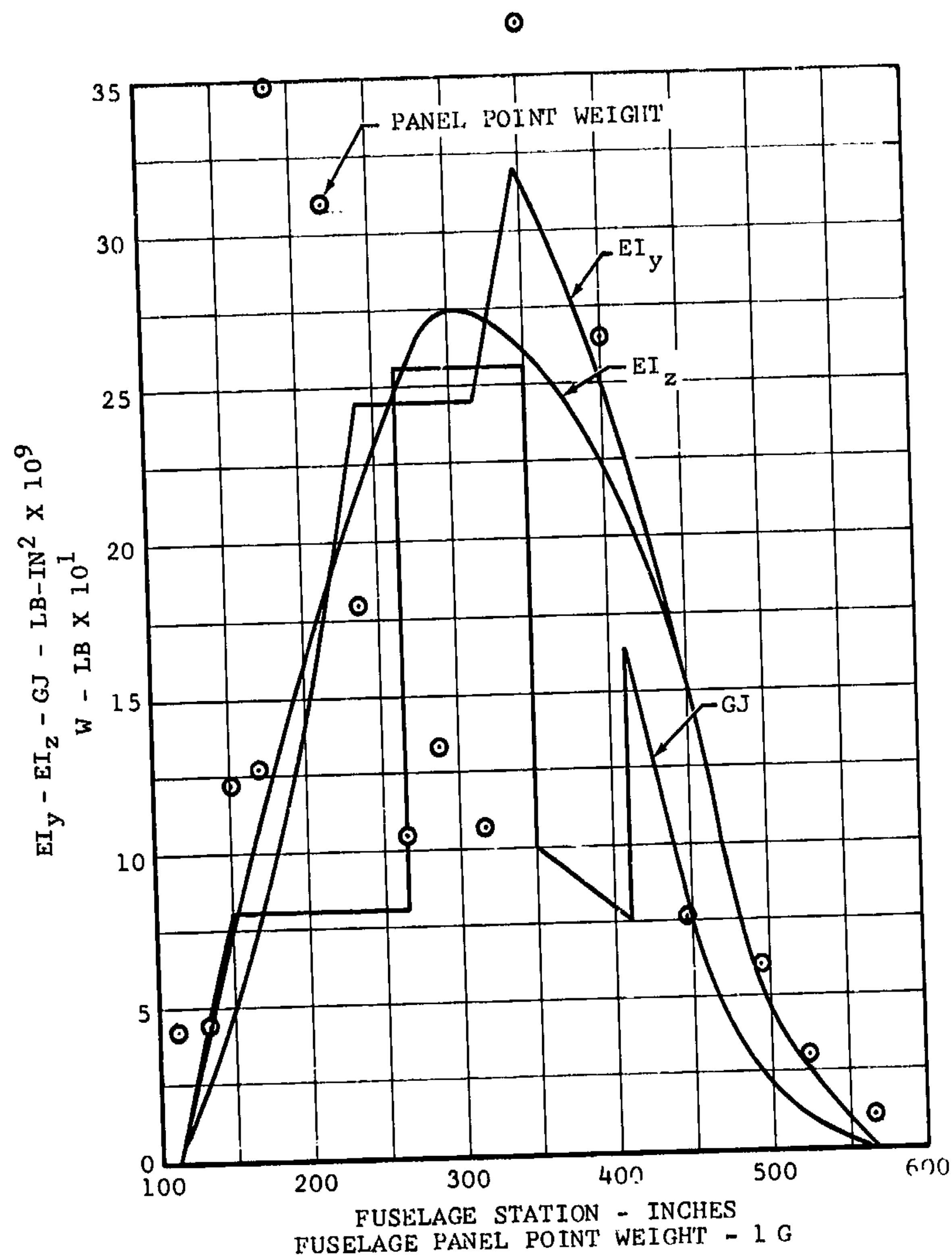


Figure III-4. Fuselage Stiffness and Panel Point Weight.

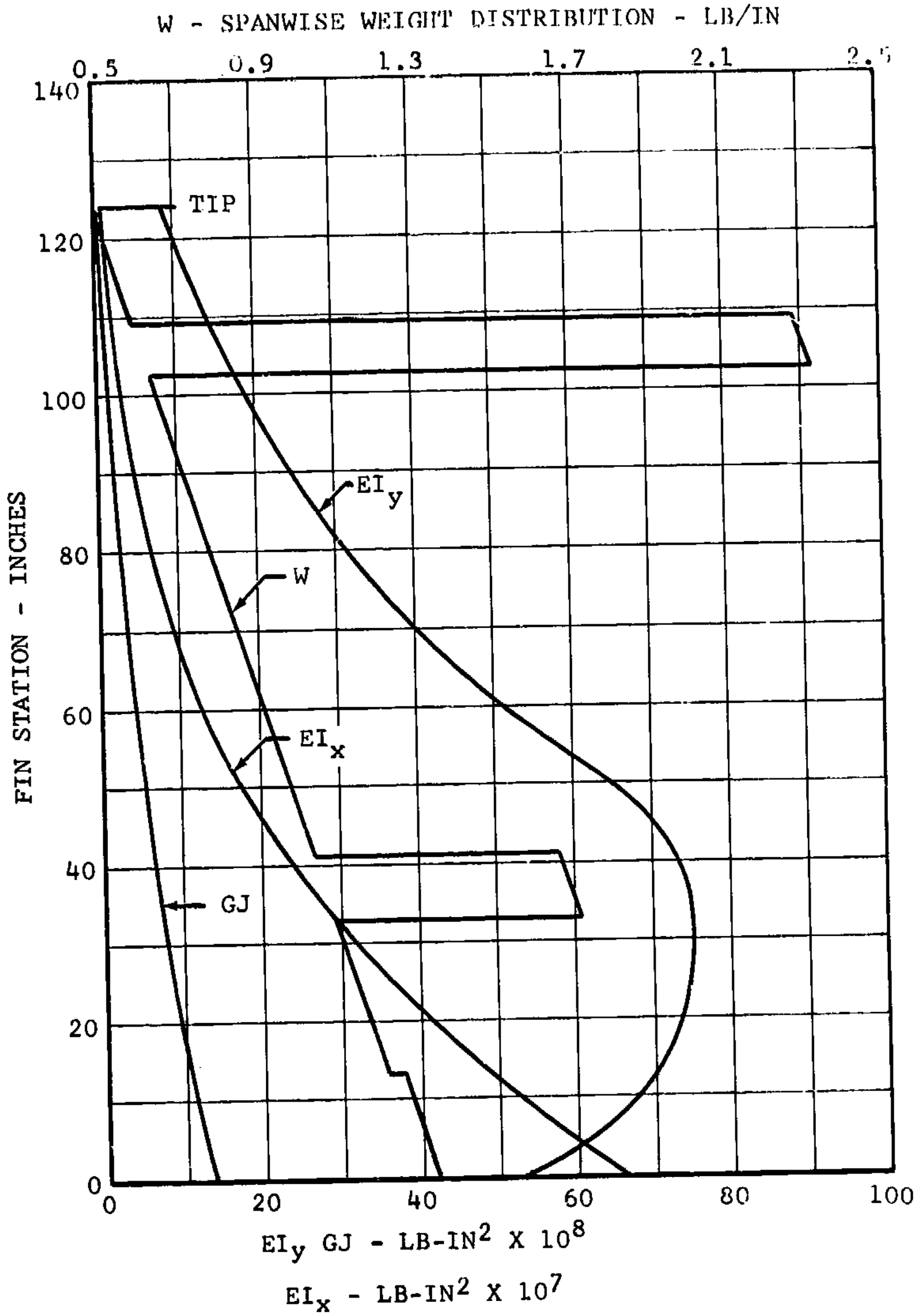


Figure III-5. Vertical Tail Stiffness and Weight Distribution.

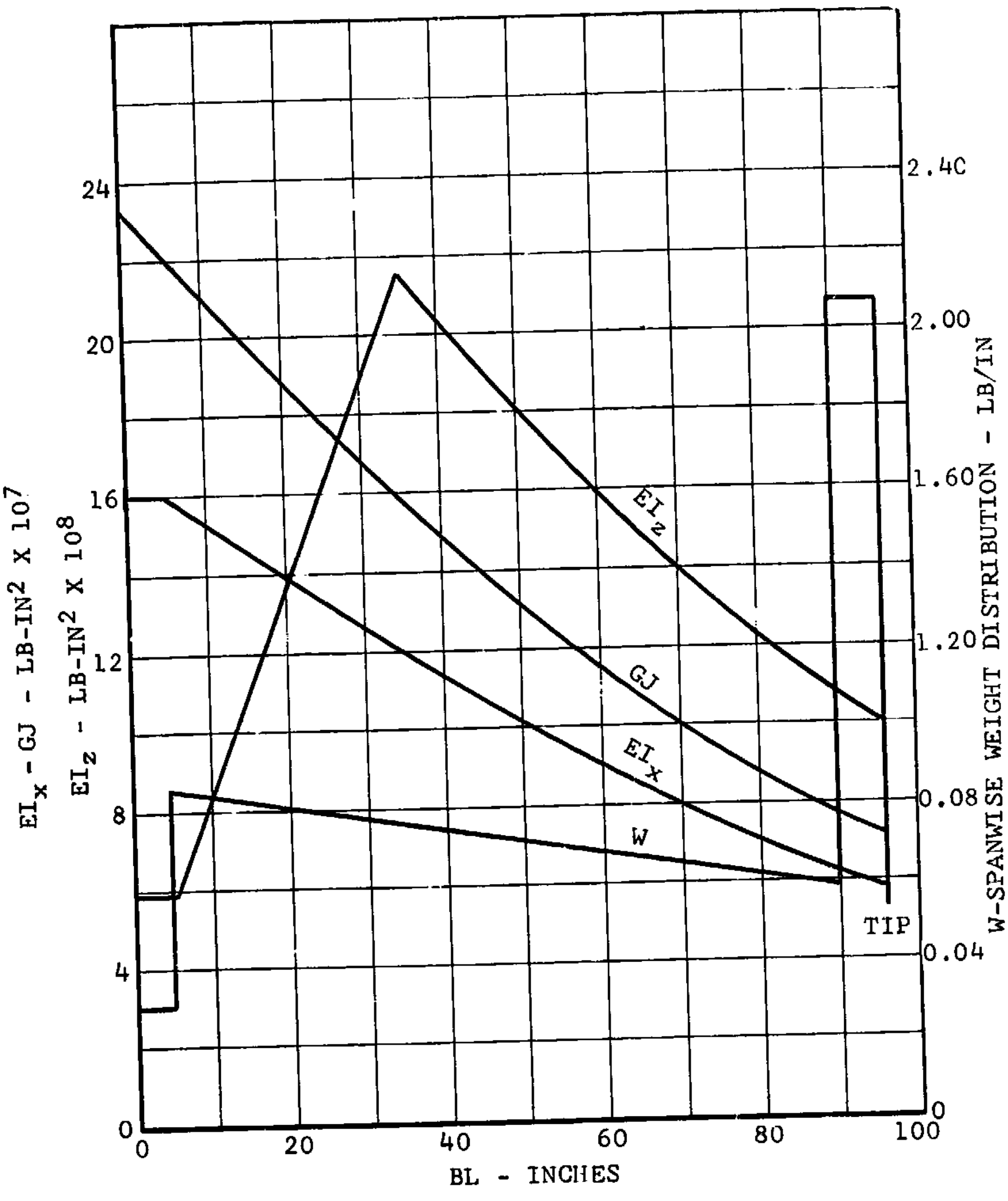


Figure III-6. Horizontal Tail Stiffness and Weight Distribution.

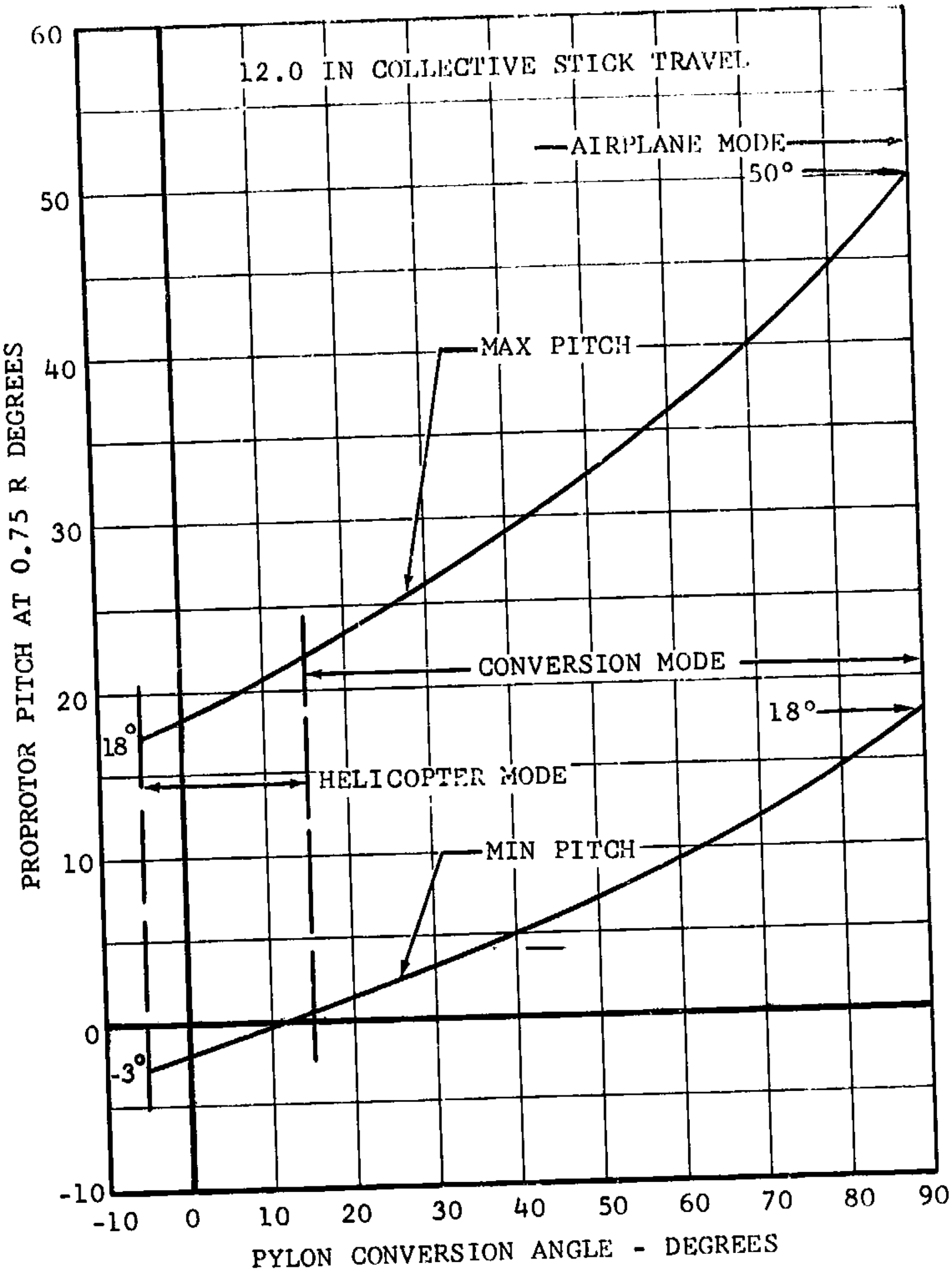


Figure III-7. Collective Pitch Versus Conversion Angle.

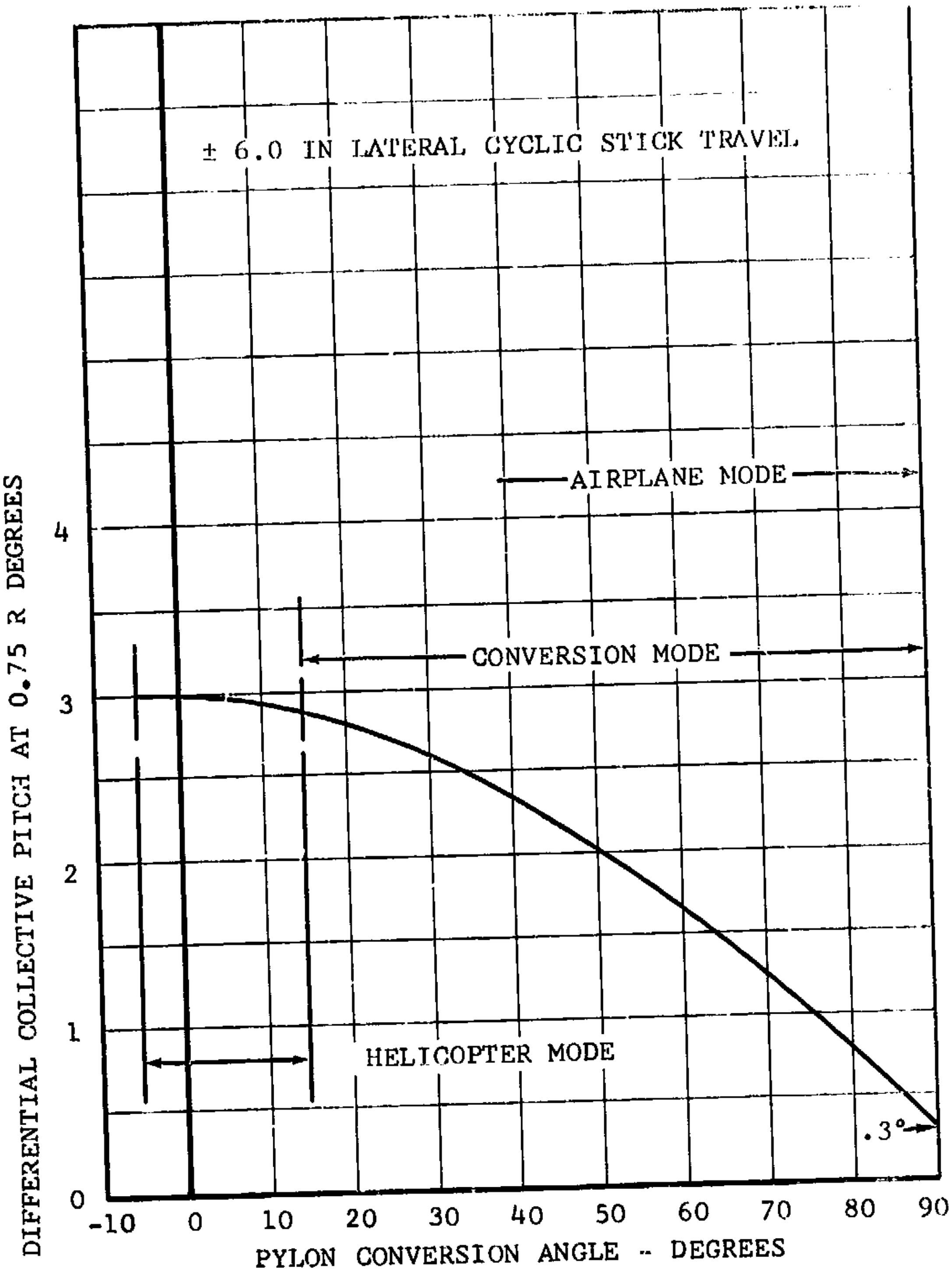


Figure III-8. Differential Collective Pitch Versus Conversion Angle.

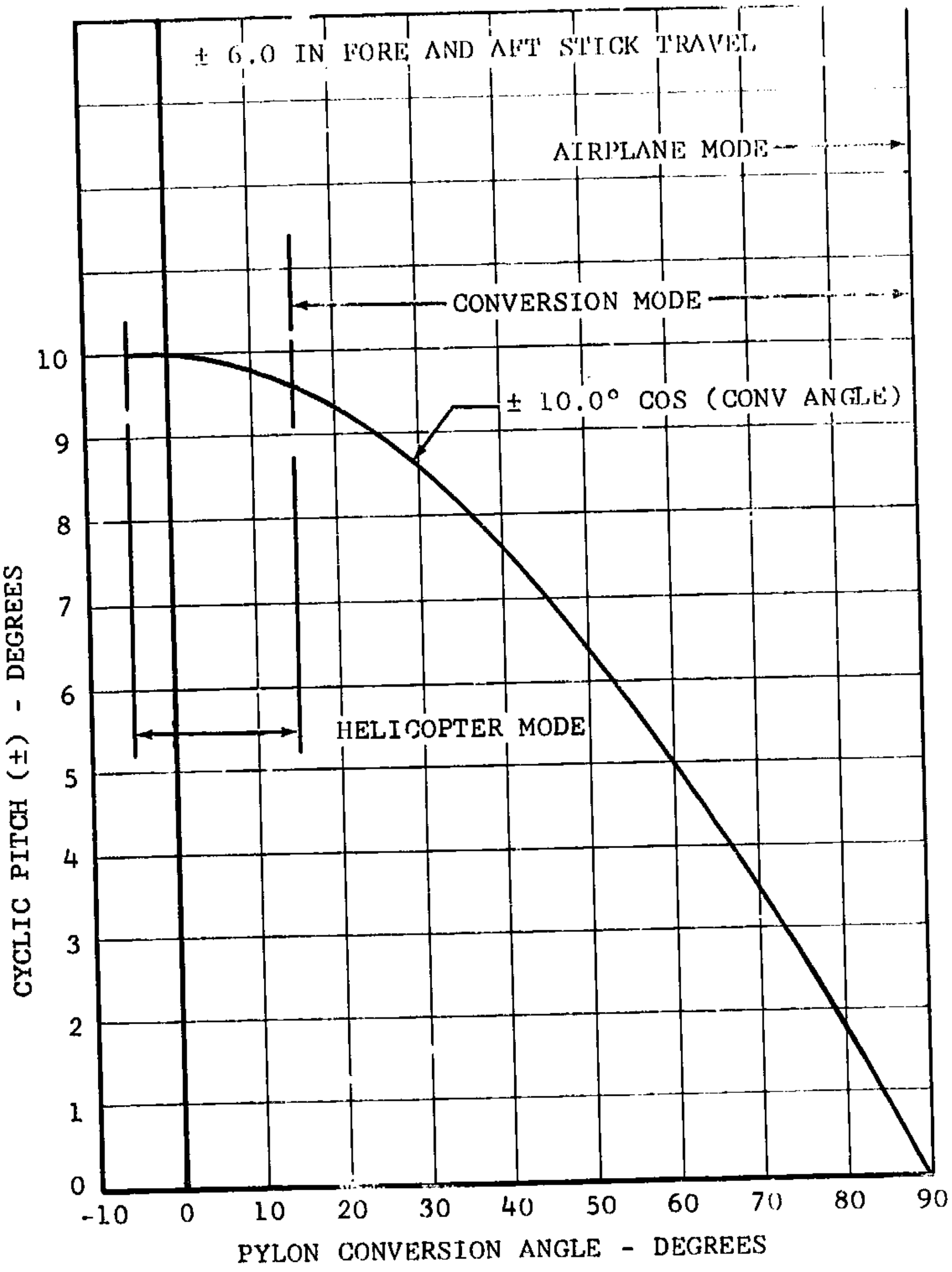


Figure III-9. Fore and Aft Cyclic Pitch Versus Conversion Angle.

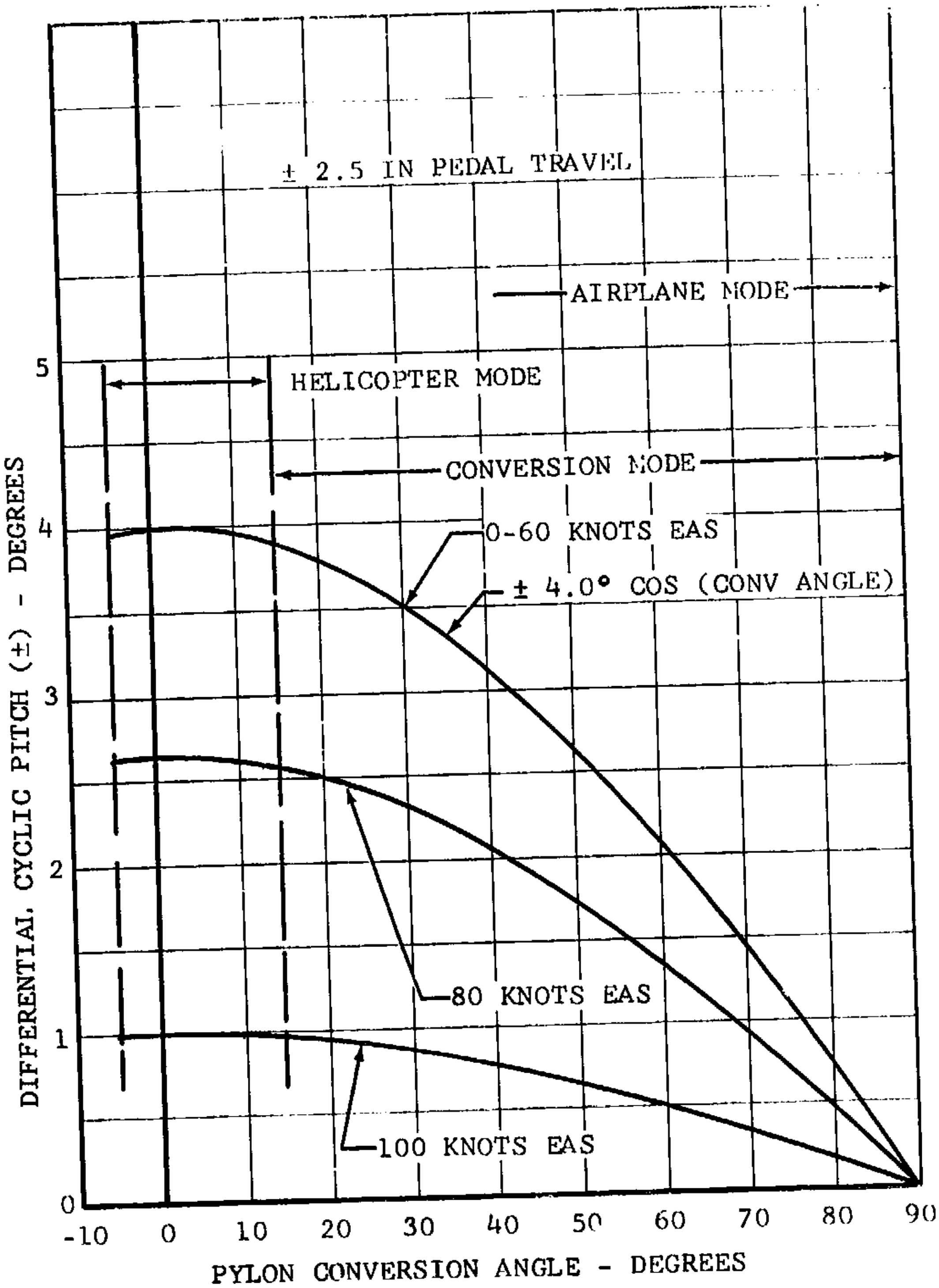


Figure III-10. Differential Cyclic Pitch Versus Conversion Angle.

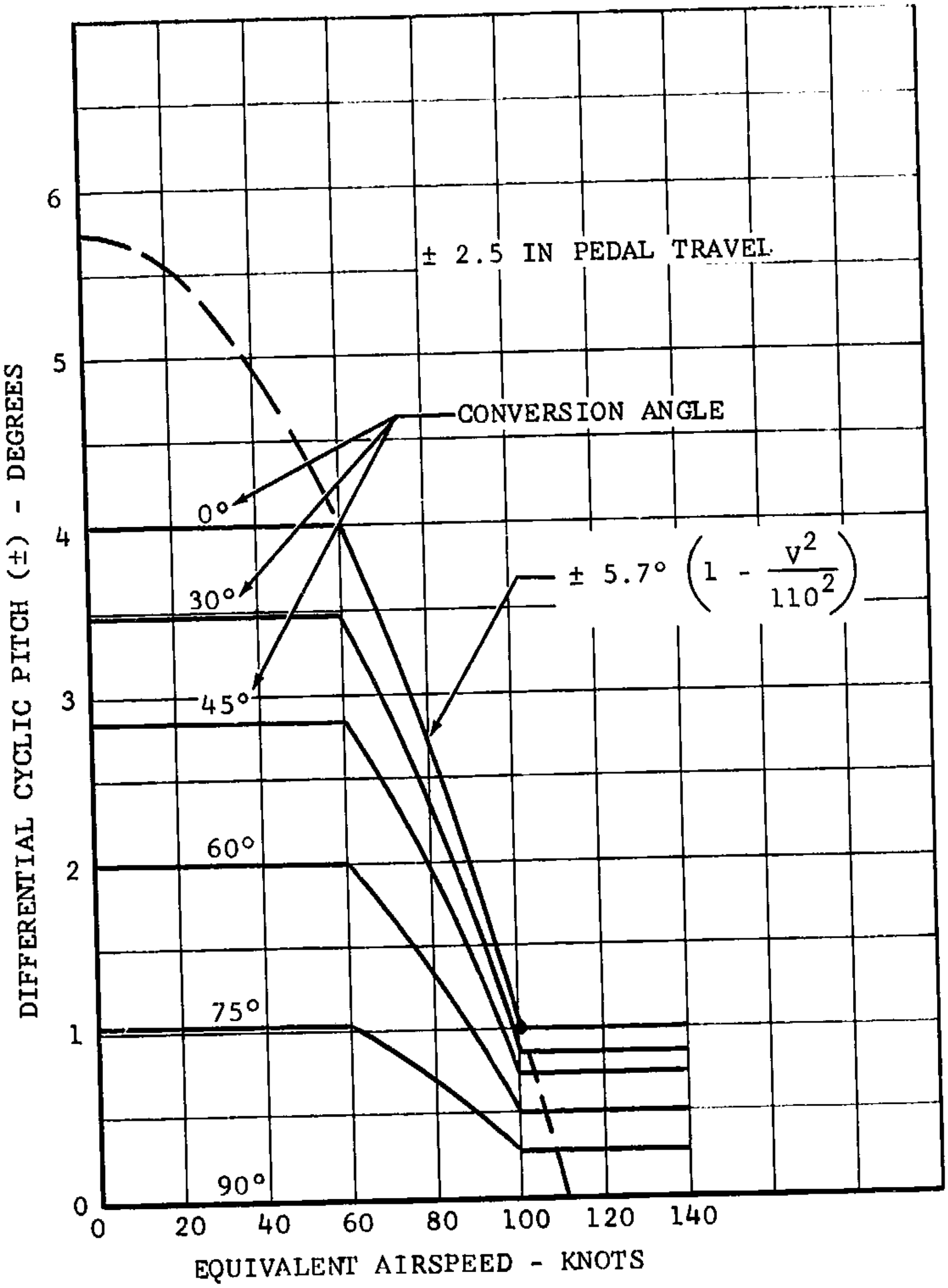


Figure III-11. Differential Cyclic Pitch Versus Airspeed.

IV. WEIGHT ANALYSIS

Weights of the Model 300 proprotor were determined from detail fabrication drawings. Weights for other major structural components such as wing, fuselage, empennage and landing gear were determined from layout drawings and statistical curves based on the weights of components of other aircraft. Transmission weights were estimated from layouts by use of empirical formula and by comparison with similar components in present use. Weights of items such as engine, instruments, and electronics, were determined from manufacturers specifications.

Layouts used for estimating and calculating weights are included in Section X. These drawings are referred to in the applicable group weight analysis discussion that follows. These discussions also detail the methods used to determine the weights.

A group weight statement, Table IV-1, shows the weight empty is 6876 pounds. Based on this weight, four mission weights have been established and are shown in Table IV-2. The research mission is shown for the normal gross weight of 9500 pounds and makes provisions for instrumentation, ejection seats, pilot and copilot, and 1600 pounds of fuel. For the civil mission, the interior is furnished to accommodate eight passengers and 240 pounds of baggage. The fuel load is adjusted to 1028 pounds to give a gross weight of 10,300 pounds, thereby allowing hover out of ground effect at 4000 feet, 95°F. The military mission gross weight is 11,200 pounds. This weight provides for a crew of three with three additional men being hoisted aboard, at the mid point of the mission, while hovering out of ground effect at 4000 feet, 95°F. The ferry mission is shown at the maximum gross weight of 12,400 pounds. A discussion of the performance for each mission is contained in Section V, Performance.

TABLE IV-1

GROUP WEIGHT STATEMENT

Rotor Group		910
Blade Assembly	580	
Hub Assembly	270	
Spinner	60	
Wing Group		700
Tail Group		245
Horizontal Tail	136	
Vertical Tail	109	
Body Group		1055
Alighting Gear		350

TABLE IV-1 Continued

Flight Controls Group		532
Cockpit Controls	53	
Rotor, Nonrotating	236	
Rotor, Rotating	171	
Fixed Wing	72	
Engine Section		167
Engine Mount	16	
Firewall	59	
Cowl	92	
Propulsion Group		2086
Engine Installation	724	
Conversion System	126	
Air Induction System	78	
Exhaust System	16	
Lubrication System	32	
Fuel System	91	
Engine Controls	59	
Starting System	59	
Rotor Governor	21	
Drive System		
Gear Boxes	761	
Transmission Drive	74	
Rotor Drive	45	
Instrument Group		110
Hydraulic and Pneumatic Group		78
Electrical Group		283
Electronics Group		84
Furnishings and Equipment		192
Personnel Accommodations	74	
Miscellaneous Equipment and Furnishings	64	
Emergency Equipment	54	
Air Conditioning Equipment		58
Undrainable Oil		12
Unusable Fuel		14
WEIGHT EMPTY, POUNDS		6876

TABLE IV-2

MODEL 300 MISSION WEIGHTS

Load Condition	Research Mission	Civil Mission	Military Mission	Ferry Mission
Crew	400	340	600	400
Passengers (8 at 170)		1360		
Fuel	1600	1028	1600	1600
Auxiliary Fuel			1059	3133
Engine Oil	35	35	35	35
Mission Equipment				
Avionics		171	361	
Armor			300	
Oxygen Installation				110
Rescue Equipment			250	
Interior				
Eight-Place Commerical		250		
Ejection Seat Increase	114			
Third Man Crew Seat			37	
Auxiliary Fuel Tank			82	246
Instrumentation	475			
Baggage		240		
Total Useful Load	2624	3424	4324	5524
Weight Empty	6876	6876	6876	6876
Gross Weight	9500	10300	11200	12400

A. Rotor Group

The Model 300 lift and thrust system consists of two three-bladed, semi-rigid proprotors, gimbal mounted with a hub spring which provides flapping restraint.

The Rotor Group weight was derived from detail computations of fabrication drawings and is 910 pounds. The proprotor assembly is shown on Bell drawing 300-010-100 in Section X.

B. Wing Group

The wing of the Model 300 is basically a two-spar, single-cell structure utilizing bonded aluminum honeycomb sandwich construction for the upper and lower panel of the main structural box as shown on Drawing 300-960-007.

Wing primary structure weight was based on panel and spar thickness requirements established by stress analysis. Weight of additional structure was estimated from layout drawings.

Figure IV-1 presents a statistical wing weight estimation curve developed from available data on utility and cargo-type aircraft. This curve includes an adjustment to compensate for the torsional and dynamic effects of supporting the pylon by a spindle assembly at the wing tips. This adjustment was based on detailed weight estimates for the Bell Model 266 tilt proprotor, the components of which were sized by a detailed stress analysis. Model 300 wing weight taken from this adjusted curve is 598 pounds. An additional 102 pounds was included because of the constant wing section with nontapered skins. The total estimated wing weight is 700 pounds.

C. Tail Group

The Model 300 tail assemblies consist of conventional stabilizer-elevator and fin-rudder configurations. Weights for the tail assemblies were estimated from layouts. The unit weights obtained were consistent with those for similar designs operating in comparable flight regimes as shown in Table IV-3.

TABLE IV-3

TAIL SURFACE UNIT WEIGHTS

Model	Horizontal Tail (lb/sq ft)	Vertical Tail (lb/sq ft)
XV-5A	1.81	2.13
XC-142	2.31	2.21
AC-1	2.23	1.86
262	2.44	2.44
266 Bell	2.68	2.28
300 Bell	2.18	1.88

D. Body Group

The fuselage of the Model 300 is a nonpressurized semi-monocoque structure shown on Drawing 300-960-008. The basic fuselage weight was estimated from Bell-developed equations with penalties added for the flooring, doors, windows, and windshields.

The windshield weight was based on 0.50-inch thick stretched acrylic, and the same material with a thickness of 0.188 inch is used for the side and top panels.

The estimating method of Reference 26 was used to verify the fuselage weight. Inasmuch as this method is based on aircraft operating in a higher speed regime than the Model 300, it is felt that the results will be conservative.

The estimating method considers the total fuselage weight to be the sum of the basic weight required to provide minimum skins, stringers or longerons, and circumferential stiffeners to resist basic flight loads plus weight panalties incurred to support concentrated loads and redistribute around cutouts and through joints. Basic weight, F_B, is defined by the expression

$$F_B = 1.123\ S + f\ (N_Z,\ Q,\ L,\ h)$$

The 1.123 constant is based on a minimum skin of 0.040-inch 7075 aluminum plus 0.038 equivalent gage to account for stiffeners (i.e., 0.078 inch x 0.10 pounds/cubic inch x 144 square inches/ square foot equal 1.123 pounds/square foot). Because the Model 300 loadings and design permit use of minimum skin thickness of 0.020 aluminum, this factor was reduced to (0.020 + 0.020) x 0.10 x 144 = 0.58. Therefore,

$F_B = 0.58\ S + f\ (N_Z,\ Q,\ L,\ h)$ (with the function "f" taken from Figure IV-2) = 306 pounds

Penalties were then determined as shown in Table IV-4.

TABLE IV-4

FUSELAGE WEIGHT PENALTIES

Nose Gear Penalty		33
Bulkhead = $0.00025\ W\ N_L$	5	
Body Cutout = 0.4 lb/in x 38 in	15	
Door = 2.0 lb/sq-ft x 4.1 sq-ft	8	
Door Mechanism	5	
Main Gear Penalty		196
Bulkhead = $0.001\ W\ N_L$	21	
Body Cutout = 0.8 lb/in x 58 x 2	93	
Door = 2.0 lb/sq-ft x 12.5 sq-ft x 2	50	
Door Mechanism	32	
Canopy and Windshield Penalty (from Figure IV-2)		166

TABLE IV-4 - Continued

Cockpit Penalty		116
Bulkhead = 2 lb/sq-ft x 25 sq-ft	50	
Body Cutout = 0.5 lb/in x 74 in	37	
Flooring = 1.0 lb/sq-ft x 29 sq-ft	29	
Production Joint Penalty = 0.025 x F_B x J		16
Tail Support Structural Penalty = 0.15 x W_T		41
Wing Attachment Structural Penalty = 0.0005 N_Z W		24
Equipment Support		
Penalty = 0.5 lb/cu-ft x 20 cu-ft	10	
Cargo Floor Penalty = 1.0 lb/sq-ft x 47 sq-ft	47	
Door Penalty		31
Body Cutout = 0.4 lb/in x 28 in	11	
Door = 2 lb/sq-ft x 10 sq-ft	20	
Miscellaneous		
Penalty = 0.1 x Total of Above Penalties		68
Total Fuselage Penalty Weight = F_P, Pounds		748
Total Fuselage Weight = F_B + F_P, Pounds		1054
Model 300 Fuselage Weight, Pounds		1055

Parameters and symbols used in the above equations are shown in Table IV-5.

TABLE IV-5

FUSELAGE PARAMETERS AND SYMBOLS

Parameter or Symbol	Description	Value for Model 300
F_B	Fuselage Basic Weight	306 lb
S	Fuselage Wetted Area	520 sq-ft
N_Z	Ultimate Flight Load Factor	5.0
Q	Weight of Fuselage and Controls	3483 lb

TABLE IV-5 - Continued

L	Fuselage Length	38.1 ft
h	Fuselage Depth	6.2 ft
W	Design Gross Weight	9500 lb
F_P	Fuselage Penalty Weight	748 lb
N_L	Ultimate Landing Load Factor	2.25
W_A	Windshield Area	56 sq-ft
J	Production Joints	2
W_T	Tail Group Weight	245 lb

E. Alighting Gear Group

The landing gear of the Model 300 is a hydraulically operated, retractable, tricycle configuration with a dual-wheel nose gear and a single-wheel main gear.

Gear structure weight was taken from a gear design layout and stress analysis. Hydraulic system and control system weights were estimated from layout drawings. Rolling gear components and their associated weights, which were taken from vendor catalog data, are shown in Table IV-6.

TABLE IV-6

ROLLING GEAR COMPONENT DATA

Item	Nose Gear			Main Gear		
	Number	Size	Weight (lb)	Number	Size	Weight (lb)
Tire and Tube	2	5.00 x 5	12	2	8.50 x 10	51
Wheel	2	5.00 x 5	7	2	8.50 x 10	26
Brake		-	-			14
Total			19			91

A comparison of the Model 300 gear weight as a percentage of design weight with similar data for current generation V/STOL aircraft is presented in Table IV-7.

TABLE IV-7

ALIGHTING GEAR GROUP WEIGHT COMPARISON

Model	Design Gross Weight (lb)	Gear Group Weight (lb)	Percent
XC142	37474	1211	3.23
X22A	14500	432	2.94
XV-5A	9200	420	4.56
XV-4A	7200	291	4.04
300	9500	350	3.68

F. Controls Group

The Model 300 flight control system consists of conventional helicopter cyclic and collective controls and airframe aileron, flap, elevator and rudder controls, along with phasing controls which provide the proper combination of the two systems during transition from helicopter to airplane mode.

All rotating control components and all other critical components were sized by a preliminary stress analysis and weights were estimated from layout drawings of these components. Weights for other control system components were based on routing lengths and existing hardware weights.

G. Engine Section and Nacelle Group

This group includes the engine mount, firewalls and wingtip-mounted pylon cowling.

One PT6C-40 engine is mounted at each wingtip in the pylon assembly shown on 300-960-003. The engine face is bolted directly to the transmission case. Weights for the engine support were estimated from layouts.

Firewall weight includes the forward, aft, upper and lower induction firewalls. Aluminum extrusions which support the firewalls, and act as supports for the cowling, are included in this weight. Weights were based on 0.020 titanium for webs and stiffeners.

The cowls consist of aluminum skins and support structure. Weights were based on gages determined by a preliminary stress analysis and were estimated from layout drawings.

II. Propulsion Group

1. General Description

Power is supplied by two Pratt & Whitney PT6C-40 engines. This power is transmitted to the proprotors by means of herringbone and planetary gear stages in the main transmission. Spiral bevel gears are utilized in the interconnect shaft system which provides power to both proprotors from one engine in the event of a single engine failure. A center gear box, with a set of bevel gears, is provided in the interconnect driveshaft system to account for the wing sweep angle.

2. Engine Installation

Dry weights for the PT6C-40 engines were taken from the manufacturer's specification. Estimated weights of residual fluids and installation hardware were added to complete the 724-pound total for two engines.

3. Conversion System

This system consists of an Acme screw actuator with a hydraulic motor at each pylon. An interconnected drive system is incorporated to provide for operation of both actuators simultaneously and to provide power to both actuators from one hydraulic motor in the event of a hydraulic system failure. A small control-phasing gearbox is provided at the center of the interconnect shaft. Weights for this system were estimated from layouts and taken from vendor catalog data.

4. Air Induction System

Engine inlet air is inducted from an opening adjacent to the spinner immediately aft of the proprotor rotation plane. Weight for this fiberglass duct structure, seals, screens, blower and the induction by-pass ejector was estimated from powerplant layouts.

5. Exhaust System

An 0.030 stainless steel exhaust pipe with an 0.030 stainless steel turning vane is clamped to each engine. The exhaust ejector baffle, which is considered part of the exhaust system, is 0.020 stainless steel. Weight of the exhaust pipes, attachment ring, turning vane and exhaust ejector baffle is 16 pounds.

6. Lubrication System Engine

This system contains an oil cooler, filter, plumbing and hardware with the oil tank as an integral part of the engine. The blower is carried in the air induction system. Weights were based on equivalent-sized components used in existing aircraft.

7. Fuel System

The Model 300 has two separate fuel systems, one for each engine. Bladder cells are located in each wing as shown on Drawing 300-960-007. The fuel system weight was estimated from a contractor-developed curve for fuel system weight as a function of fuel capacity as shown on Figure IV-3. Fuel system weight, based on 246-gallon capacity at 0.30 pound per gallon, is 74 pounds. Due to FAA requirements for twin-engine installation, an additional 17-pound penalty was added for a total estimated fuel system weight of 91 pounds.

8. Engine Controls

The engine control system primarily consists of droop compensation and power-lever controls for each engine. Weights for these were estimated using similar UH-1 component weights and allowing for the increased cockpit-to-engine routing distance.

9. Starting System

The 200-ampere starter-generator provided at each engine is powered by two 13-ampere-hour batteries. Weights were based on equivalent-sized components used in existing aircraft.

10. Rotor Pitch Governor Control

The rotor pitch governor is an electro-hydromechanical system which maintains selected rotor rpm. Electronic equipment, actuator, and associated linkage were estimated based on comparisons of similar items used in existing systems.

11. Drive System

Engine power is transmitted to each proprotor directly from the engine through the main transmission by herringbone and planetary gear stages as shown on Drawing 300-960-004. Design data for the entire drive train, including the interconnect shafting and center gearbox, is given in Table IV-8.

All main transmission cases are magnesium except for the ring gear (nose) case which is cast aluminum. A large pylon case, to which the firewalls and cowls are mounted, houses the bevel drives for the interconnect shaft system; the steel spindle attaches to this case.

TABLE IV-8

DRIVE SYSTEM DATA

Component	Speed (rpm)	Design (hp)	Max Cont Torque (in-lb)
Engine Output	30000	1150	2412
Main Transmission Output	458	946	130127
Interconnect Drive Shaft	6485	633	6144

Gear Ratios

Engine Output	Basic
Interconnect Driveshaft	0.2162
Rotor Shaft	0.0188
Hydraulic Pump Drive	0.2010
Engine N_{II} Governor	0.1400
Center Gearbox	0.2162

Gear Stage Data

Stage	Speed (rpm)	Max Cont Torque (in-lb)
*Engine Output	30000	2412
*Main Transmission Input	8487	8532
First Planetary Output	1776	33557
Second Planetary Output	458	130127
*Main Transmission Spur Output	8487	4704
*Main Transmission Bevel Output	6485	6144

*These components designed by single engine operation power.

Weights for the Model 300 drive system were estimated from preliminary layouts by use of empirical formulas and, where possible, by comparison with similar components in present use.

The main transmission weights were verified by the estimating method contained in Reference 27. Gear stage weights were read from Figure IV-4, which was taken from Reference 27. All weights were read at the upper end of the weight range and then reduced by ten percent to account for improvements in materials and methodology occurring since the report was written. The total of these reduced weights was multiplied by the factor for magnesium cases, given in the report, to derive a basic housing weight. Weights were then added for special mounting and shape provisions and accessory drives.

These weights, taken from design estimates, are shown in Table IV-9.

TABLE IV-9

TRANSMISSION SPECIAL PROVISIONS

Case extension to support engine firewalls and cowl and attach conversion spindle	56
Freewheeling unit	7
Lube system (with oil)	36
Accessory drives	10
Torquemeter	17
Support installation (spindle and bearings)	39
TOTAL, Pounds	165

Figure IV-5 shows the main transmission gear stage schematic and gear stage weights derived from Figure IV-4, along with a tabulation of the data used and the summation of total estimated weight by the verification method as compared to Model 300 weight.

I. Instrument Group

The instrument group consists of engine, flight and navigation instruments, transmitters, and installations as shown in Table IV-10. Weights for the instruments and transmitters were based on those currently in use on present day helicopters. Installation weights are assumed to be the same except for wiring, which has been increased to compensate for greater distance between the cockpit and propulsion group.

TABLE IV-10

INSTRUMENT GROUP WEIGHTS

Instrument	No.	Indicators (lb)	Transmitters (lb)	Installation (lb)	Total (lb)
Altimeter	1	1.5			1.5
Airspeed	2	1.7	0.9	0.9	3.5
Clock	2	1.0			1.0
Standby Compass	1	0.8			0.8
Angle of Attack	2	2.5			2.5
Vertical Speed	2	2.4			2.4
Turn and Slip	2	3.8			3.8
Attitude	1	2.8			2.8
Vertical Gyro	1	6.0	9.0	4.2	19.2
Gyro Compass	1	6.5	7.3	4.8	18.6
Outside Air Temperature	1	0.2			0.2
Fuel Flow	2	1.8			1.8
Transmission Oil Pressure	2	1.8	2.0		3.8
Engine Oil Temperature	2	1.8	0.4		2.2
Engine Oil Pressure	2	1.8	2.0		3.8
Fuel Pressure	1	0.6	1.2		1.8
Transmission Oil Temperature	2	1.8	0.4		2.2
Gas Producer Tachometer	2	1.8	1.6		3.4
Fuel Quantity	1	0.5	3.0		3.5
Dual Torquemeter	2	1.8	2.0	1.2	5.0
Triple Tachometer	1	4.4	2.4	1.2	8.0
Hydraulic Pressure	2	1.8	2.0	2.0	5.8
Turbine Inlet Temperature	2	1.8			1.8
RPM Warning	1	0.3	2.2		2.5
Position, Flap	2	2.2	0.1	0.5	2.8
Position, Main Gear	1	0.3	0.9	2.0	3.2
Conversion	2	1.1	0.1	0.5	1.7
TOTAL INSTRUMENT GROUP WEIGHT					109.6

J. Hydraulics Group

Hydraulic power is utilized to extend and retract the landing gear, to power the aileron pylon conversion actuators, and to provide boost capability in the proprotor cyclic and collective control systems. The Model 300 has a completely dual 1500 psi hydraulic system. Only the weights of the pumps, reservoirs, accumulators, valves, and interconnecting plumbing are included in the main system weight. Weights of components and plumbing providing power to a specific system are carried in the weight of that system.

K. Electrical Group

The ac-dc electrical system on the Model 300 is powered by starter-generators attached to the engines. Two 13-ampere-hour batteries are provided to furnish power for the starter-generators. Weights of these and other major components are based on vendor data. Wiring and hardware weights were estimated from wiring diagrams and routing layouts. The weights of the components are listed in Table IV-11.

TABLE IV-11

ELECTRICAL GROUP WEIGHTS

	Weight (lb)
DC System	
Batteries	48
Battery Installation	2
Transformer	4
Voltage Regulator	6
Switches, Rheostats and Panels	4
Relays	19
Wiring and Miscellaneous	87
Equipment Supports	16
AC System	
Inverter	26
Ammeters and Voltmeters	2
Switches, Rheostats, and Panels	26
Circuit Breakers and Fuses	10
Junction and Distribution Boxes	3
Relays	1
Wiring and Miscellaneous	14
Lights	15
TOTAL ELECTRICAL GROUP WEIGHT	283

L. Electronics Group

Electronic equipment consists of AN/ARC-114 VHF-FM and AN/ARC-115 VHF radios, Collins 613L-2 transponder system and a four-station

ICS, C-6533. Weights for these systems were taken from existing installations used in current aircraft.

M. Furnishings and Equipment Group

The furnishing and equipment group includes crew accommodations, furnishings, miscellaneous equipment, and emergency equipment. The weights shown in Table IV-12 were based on similar equipment presently in use on Bell helicopters.

TABLE IV-12

FURNISHINGS AND EQUIPMENT GROUP WEIGHTS

	Weight (lb)
Accommodations	
Crew Seats	62
Crew Safety Belts	6
Crew Shoulder Harness and Inertia Reels	6
Miscellaneous Equipment	
Windshield Wiper	14
Instrument Panel	15
Consoles	20
Furnishings	
Soundproofing (cockpit)	15
Emergency Equipment	
Fire Detection System	5
Portable Fire Extinguisher	7
Engine Fire Extinguisher	42
TOTAL FURNISHINGS AND EQUIPMENT GROUP WEIGHT	192

N. Air-Conditioning Equipment Group

The air-conditioning system of the Model 300 is used for forward window defogging and heating and cooling of the cockpit compartment. The environment control unit used is the same as that now installed on the Bell Model AH-1G.

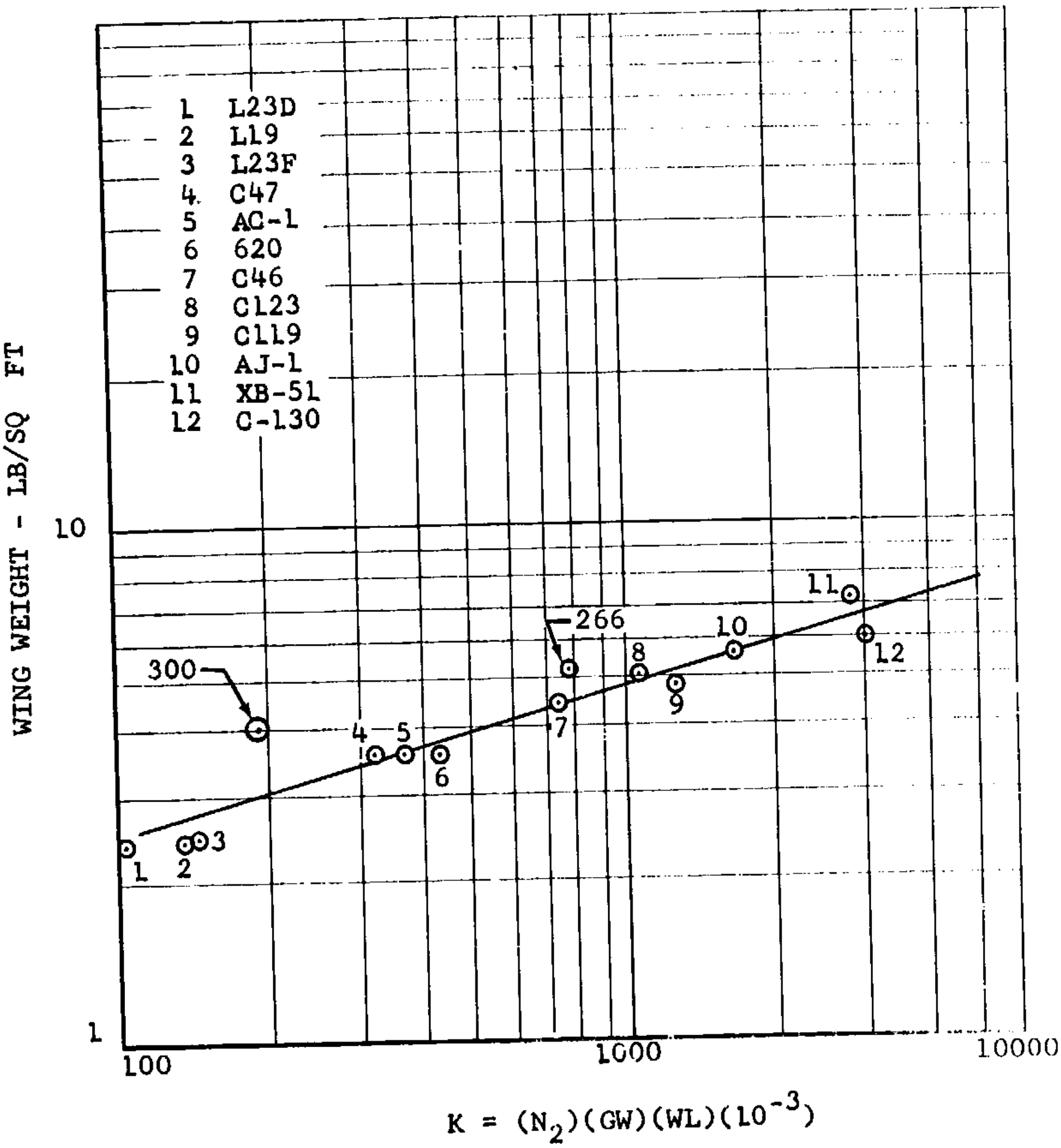
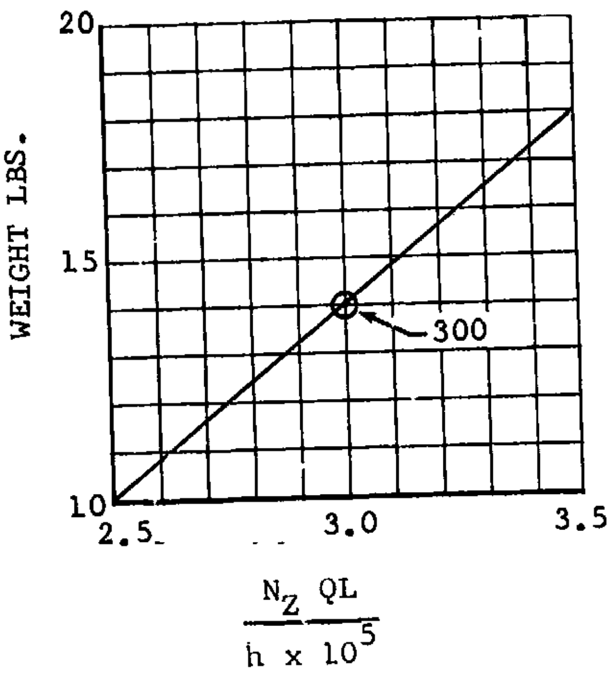


Figure IV-1. Wing Weight.

FUSELAGE BASIC WEIGHT PENALTY



WINDSHIELD OR CANOPY WEIGHT

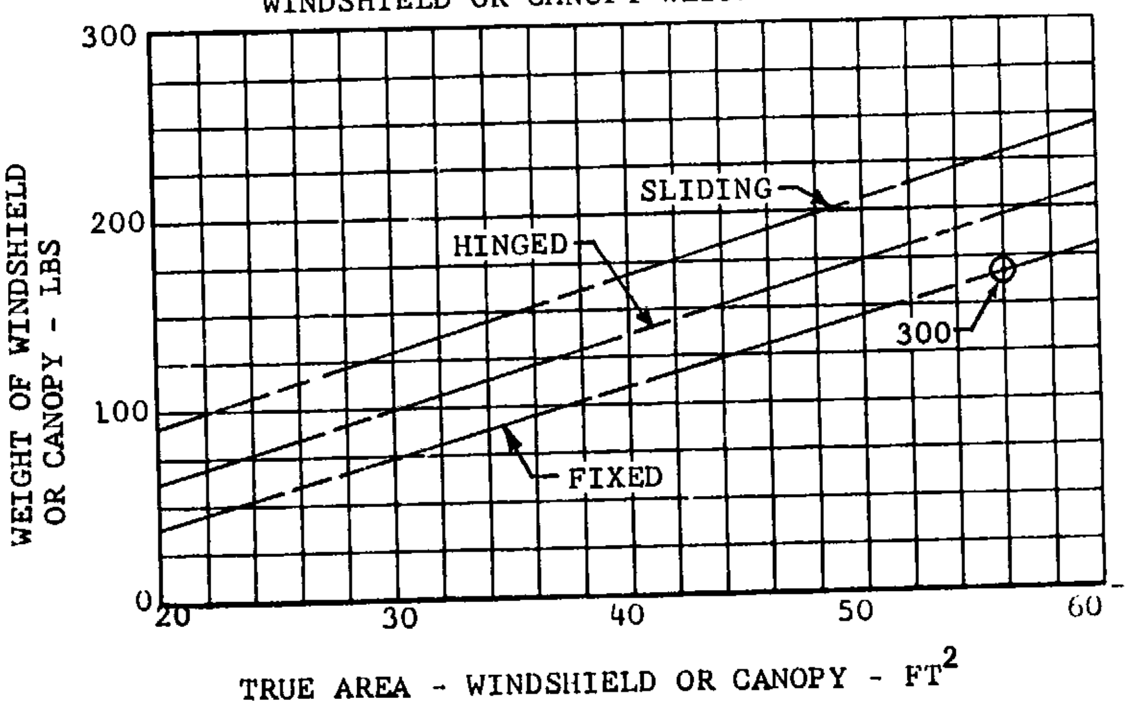


Figure IV-2. Fuselage Weight Estimation Parameters.

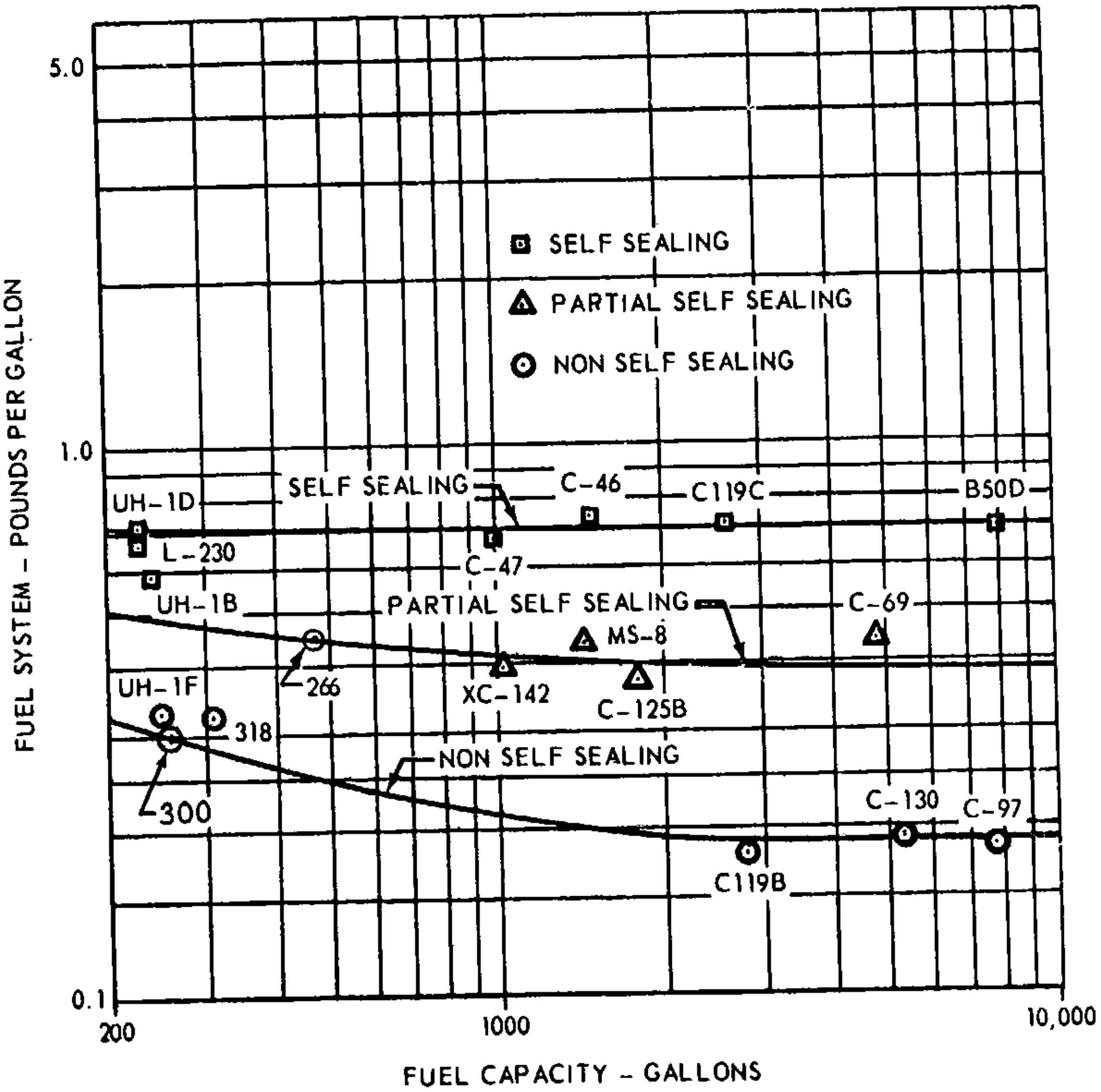


Figure IV-3. Fuel System Weight.

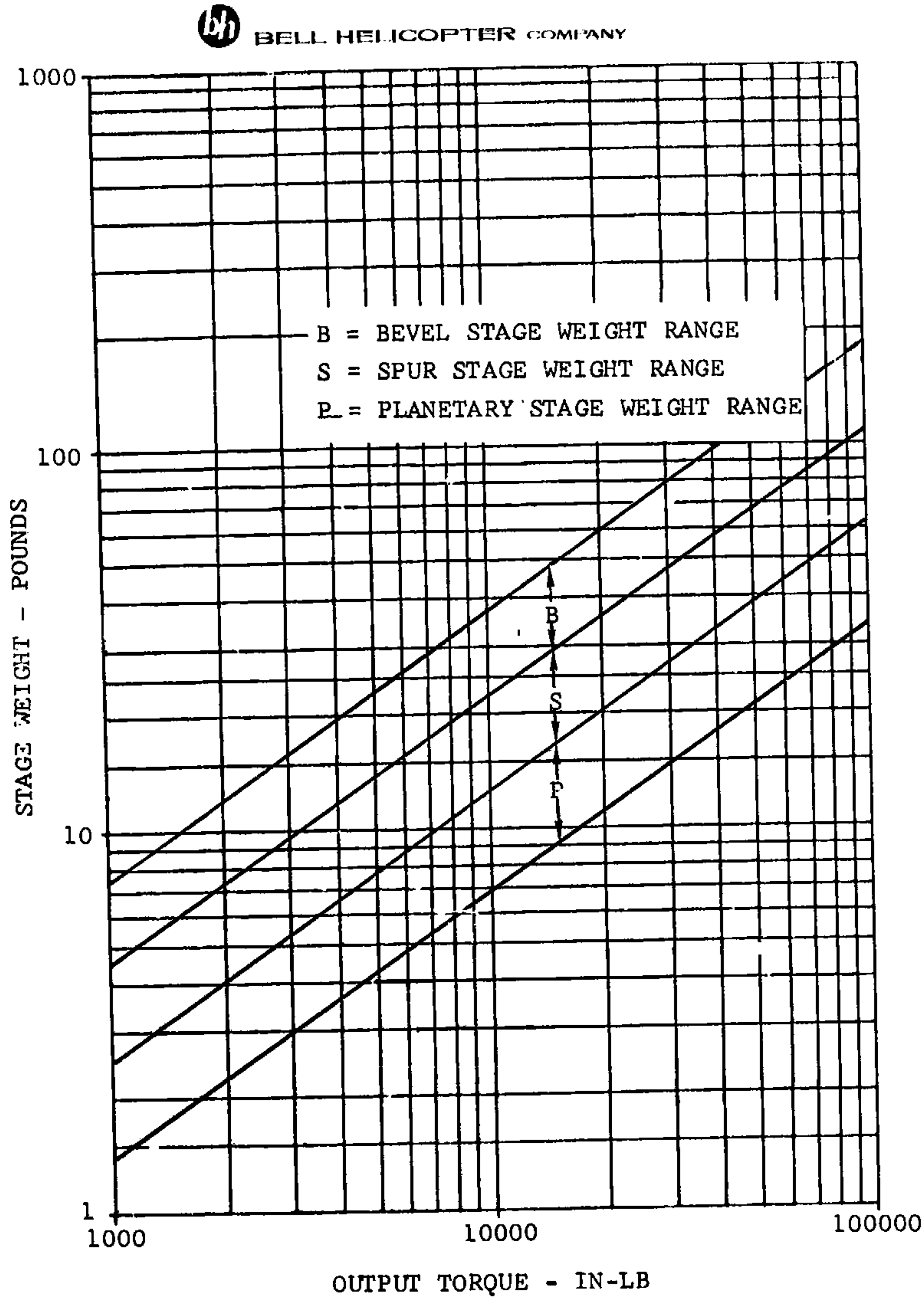
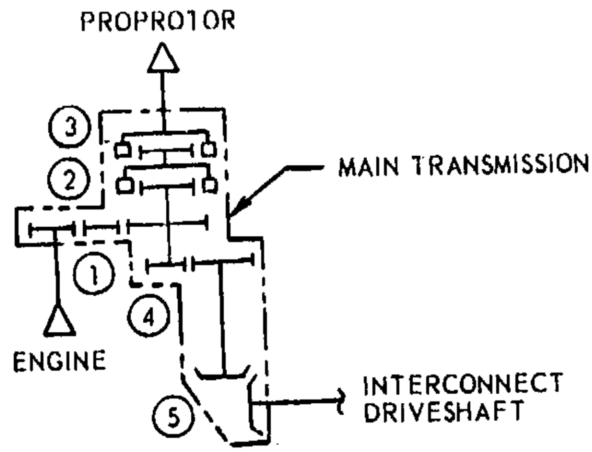


Figure IV-4. Gear Stage Weight.



Gear Stage		Design Output/Side			W_G/Side
Number	Type	(hp)	(rpm)	Torque (in-lb)	(lb)
Engine		1150	30000	2412	
1	Spur & Idler	1150	8487	8532	20 *6 26
2	Planet	946	1776	33557	29
3	Planet	946	458	130127	74
4	Spur	633	8487	4704	13
5	Bevel	633	6485	6144	26

Weight Data/Side

Item	Main Transmission (lb)
Gears, bearings, shafts, etc. = W_G (derived from Figure IV-4 with a 10% reduction)	151
Housing = 0.372 W_G	53
Special provisions	165
Total weight per side	369
Model 300 weight per side (estimated)	366

*Left transmission has two idlers; right transmission, one. Weight shown is average for both sides.

Figure IV-5. Main Transmission Schematic and Weight Data.

V. PERFORMANCE

A. Summary

The performance characteristics of the Model 300 are summarized in Table V-1 for four mission weights. These missions are: flight research, simulated civil mission, simulated military mission and ferry mission.

The aircraft has three-bladed 25-foot-diameter proprotors with 14-inch chord blades. Disc loading is 9.7 pounds per square foot at the normal gross weight of 9500 pounds. Power is supplied by two Pratt and Whitney PT6C-40 direct-drive free-turbine engines with takeoff and 30-minute ratings of 1150 horsepower and a normal power rating of 995 horsepower. The broad range of rpm for efficient operation of the free turbine permits full power to be extracted at maximum rpm during takeoff in helicopter mode and near optimum specific fuel consumption to be obtained in airplane cruise at reduced rpm. Proprotor tip speed is 740 feet per second in helicopter mode and 600 feet per second in airplane mode.

At 9500 pounds gross weight the aircraft hovers out of ground effect at 11,600 feet on a standard day and 6400 feet on a 95°F day. At this weight the service ceiling is 26,000 feet. Maximum speed is 312 knots at 15,000 feet. Hover ceiling is 2600 feet and service ceiling is 20,000 feet at the maximum gross weight of 12,400 pounds. A STOL takeoff can be made in 900 feet at 4000 feet on a 95°F day at this weight. Maximum single engine speed is 215 knots at 10,000 feet for this weight.

Maximum range for the research mission increases from 397 nautical miles at sea level to 630 nautical miles at 20,000 feet. Ferry range is 1870 nautical miles at 20,000 feet.

The performance of the Model 300 is sufficient to demonstrate civil and military VTOL applications A typical civil mission could be demonstrated carrying the equivalent weight of eight passengers and 240 pounds of baggage at a cruise speed of 260 knots (300 miles per hour) for a range of 306 nautical miles (350 statute miles). An aircrew recovery mission is selected to demonstrate military application. With a crew of three and 550 pounds of armor and rescue equipment, the aircraft can dash at 300 knots at 10,000 feet, pick up three downed airmen at a distance of 375 nautical miles, make a hovering out-of-ground effect takeoff at 4000 feet 95°F and return.

B. Airframe Aerodynamics

A one-fifth scale model of the Model 300 airframe was tested in March 1969 in the Ling-Temco-Vought 7-by-10-foot low-speed wind tunnel (Figure V-1). This test will be reported in Reference 28. The model was tested with a smooth finish and also with artificial roughness. The roughness was introduced to insure that

TABLE V-1

PERFORMANCE SUMMARY

Mission	Units	Research	Civil	Military	Ferry
Takeoff Gross Weight	lb	9500	10300	11200	12400
Crew	lb	400	340	600	400
Fuel Weight	lb	1600	1028	2659	4733
Payload	lb	475*	1600	550**	-
Helicopter Mode					
Hovering Ceiling, OGE					
Standard Day	ft	11600	9000	6200	2600
95°F Day	ft	6400	4000	1600	-
Maximum Speed, NRP					
Twin Engine, Sea Level	kt	134	133	132	131
Maximum Rate of Climb at Sea Level					
Twin Engine, NRP	ft/min	3480	3100	2740	2300
Single Engine, 30 min power	ft/min	1200	1020	820	550
Airplane Mode					
Maximum Speed, 30 min power	kt	312	312	312	311
Altitude for Maximum Speed	ft	15800	15400	14800	14200
Single Engine Maximum Speed at 10,000 ft	kt	224	221	218	215
Maximum Rate of Climb					
Twin Engine, Sea Level	ft/min	3750	3200	2800	2300
Single Engine, Sea Level	ft/min	1400	1100	820	500
Service Ceiling					
Twin Engine	ft	26200	24400	21600	20000
Single Engine	ft	13000	10800	8200	4500
Range					
Sea Level	nm	397	250	675	1220
10,000 Feet	nm	522	320	870	1580
20,000 Feet	nm	630	375	1030	1870
Average Cruise Speed					
Sea Level	kt	228	230	230	230
10,000 Feet	kt	239	243	242	245
20,000 Feet	kt	256	266	267	270

*Payload is 475 pounds of test instrumentation.

**Armor and rescue equipment are the payload for this mission.

boundary layer transition was consistent with that anticipated on the full-scale vehicle.

All airframe components, with the exception of the extended landing gear were tested in the wind tunnel. Full-scale lift and drag characteristics were determined by application of Reynolds number corrections to wind-tunnel data. Since the fixed elevator settings in the wind-tunnel test did not represent the full-scale Model 300 trim setting, values for trim, lift and drag were obtained by computation.

1. Lift Analysis

Both model and full-scale lift curves are given in Figure V-2 for flap settings of 0 and plus 30 degrees. Comparison of the model and full-scale curves shows that the major effects of the Reynolds number on the lift data were a negative shift in angle of zero lift and an increase in the lift curve slope. The minus 1.4 degree shift in the angle of zero lift was due to the Reynolds number effect on the 23-percent thick wing section. This unusual behavior was determined by extrapolating the data from Reference 29.

The 8.5-percent increase in the overall lift curve slope shown in Figure V-2 was due to an increase in the wing lift curve slope after correction for Reynolds number effects. Not apparent in Figure V-2 is an increase in wing maximum lift coefficient. A 0.15 increase over wind-tunnel data was estimated when Reynolds number corrections were applied, but this effect was offset by the elevator trim force requirements. No Reynolds number adjustments for lift were made to the fuselage, pod, and empennage lift curves.

2. Drag Analysis

Full-scale airframe drag was computed from wind-tunnel data by correcting the value for each individual component for Reynolds number differences between the model and the full-scale ship. The resulting total airframe drag coefficient is shown in Figures V-3 and V-4 and compared with the measured model test data. The method used to compute the full-scale drag is outlined below.

a. Lift and drag values for each component were obtained by comparing different test "runs" with and without the component being evaluated.

b. The profile, or parasite drag coefficients, were obtained by subtracting the induced drag from the measured total for each component. The induced drag coefficient was calculated from

$$C_{D_i} = C_L^{\,2}/\pi e AR_w$$

$$e = 0.90$$

c. Reynolds number corrections were applied to the model profile drag coefficients to obtain the full-scale coefficients. Correction factors are shown in Table V-2.

d. The corrected profile drag was then added back to the induced drag at a given lift coefficient to obtain the final component value.

e. The profile drag coefficient for the elevator was obtained from Reference 30, using interpolation for the desired airfoil section. In order to account for roughness, the drag values from Reference 30 were increased first by "fairing out" the drag "buckets" and then by increasing the resultant minimum C_{D_O} by 25 percent. The induced drag for the elevator was obtained from calculations based on the required lift and for flight conditions.

f. The total full-scale aircraft drag values were then found by summing the component values plus a five percent increase in total profile drag to account for leakage, protuberances, irregular surfaces, etc. A full-scale drag breakdown in terms of equivalent flat-plate area is shown in Table V-3 for a lift coefficient of 0.38.

In Figure V-4 it is seen that the minimum drag for the full-scale flaps up condition does not occur at zero lift but at $C_L = 0.20$. This comes about because of the fuselage and pod effects on total lift. The wing alone minumum drag occurs at $C_L = 0.07$, but with the addition of the fuselage and pod, the minimum shifts over to $C_L = 0.20$ because of the negative lift carried on these components for angles of attack up to 4 degrees.

Figure V-5 compares the aircraft drag as determined from the model test data with the drag predicted by conventional drag estimating methods. Predicted parasite drag area is 6.4 square feet with a resulting $C_{D_O} = 0.0364$.

For the helicopter mode of operation an additional flat-plate drag area of 4 square feet and 32.9 square feet were added to account for the extended landing gear and vertical pods, respectively.

C. Proprotor Performance

1. Description

Each of the 25-foot diameter proprotors has three blades. The chord, thickness, twist and amount of camber change as function of the radius. These distributions are shown in Figure III-1 and are simulated in the performance analysis by dividing the blade into four sections. These divisions are made at the nondimensional blade Stations of 0.075, 0.45, 0.70, 0.90, and 1.00. The blade is not represented between Stations 0.0 and

TABLE V-2

PARASITE DRAG CORRECTION PARAMETERS

Component	Model Reynolds Number*	Full Scale Reynolds Number**	Profile Drag Correction (%)	Reference	Remarks
Wing	1.23×10^6	9.97×10^6	-30	TN 1945	NACA 64A223 Mod section with roughness at 0.05 C
Fuselage	8.9×10^6	73.4×10^6	-26	TN 614	Slope of C_D vs Reynolds No. curve independent of shape
Pod (horizontal)	3.08×10^6	25.2×10^6	-23	TN 614	Slope of C_D vs Reynolds No. curve independent of shape
Pod (vertical)	1.04×10^6	8.4×10^6	0		No correction applied due to shape

*Reynolds No. based on maximum dimension

**For 250 knot speed, 10,000 feet, standard day

0.075 as this segment is contained within the spinner. For calculation purposes, each of these sections is further subdivided such that the entire blade is represented by 22 blade "strips".

Airfoil section characteristics were estimated for each of the four blade sections based on the Bell Helicopter airfoil section tests of Reference 31. FiguresV-6 through V-9 show samples of the airfoil section data for Mach numbers of 0.3, 0.5, 0.6 and 0.7.

2. Method and Correlation

All proprotor performance is obtained with the aid of the Bell Helicopter computer program F35(J). This program is primarily a helicopter performance computation program and is described in detail in Reference 32. The validity of this program has been proven in the past by the excellent correlation with flight-test data for both of the flight regimes that a normal helicopter rotor experiences: hover and helicopter forward flight.

TABLE V-3

DRAG BREAKDOWN - FULL SCALE AIRCRAFT

Component	Flat Plate Drag Area[1]		
Fuselage	1.60		
Wing	1.57		
Horizontal Tail	0.59		
Vertical Tail	0.55		
Pods - Horizontal	1.37		
Miscellaneous (5% Parasite)	0.28		
Airplane Parasite		5.96	
Airplane Induced (C_{D_i})		1.62[2]	
Total Airplane			7.58
Vertical Pod Increment	32.90		
Flaps Down Increment	8.63		
Undercarriage[3]	4.00		
Helicopter Parasite		51.49	
Helicopter Induced		2.82[2]	
Total Helicopter			54.31

1. These values reflect flight conditions that exist for a gross weight of 10,300 pounds. and 10,000 feet altitude, 250 knots in the airplane mode, or sea level, 50 knots in the helicopter mode (airframe C_L = 0.38).
2. Induced drag calculated on individual components based on the relation $C_L^2/\pi eAR$. Where e = 0.9.
3. Estimated undercarriage not tested.

Correlation of F35(J) with proprotor performance in the airplane mode is shown in Figure V-10. The test results shown are for a 13-foot diameter Boeing-Vertol proprotor model with "E" blades. This test was conducted in the NASA-Ames 40-by-80-foot wind tunnel. Blade parameters for the proprotor are shown in Figure V-11. The calculated values used for correlation were obtained by using both a limited amount of Boeing-Vertol section data and a set of Bell Helicopter drooped airfoil section data. The variation in the two sets of calculated results show the importance of using the proper section data. It is concluded that the computer program F35(J) can accurately predict proprotor performance for any flight regime when the proper airfoil section data are used.

3. Isolated Proprotor Performance

Figures V-12 through V-15 show proprotor performance in hovering and airplane flight. The hovering performance is shown in Figure V-12 in the form of power required versus thrust. Performance in airplane flight is shown in Figures V-13 through V-15, in the form of propulsive efficiency as a function of proprotor shaft horsepower, for sea level, 10,000 feet, and 20,000 feet standard day operation. A dashed line is shown on these figures for the steady level flight condition at a gross weight of 10,300 pounds.

D. Powerplant Performance

Figures V-16 and V-17 show the power available and fuel flow, respectively, for the hovering and helicopter forward flight performance. Figures V-18 and V-19 are carpet plots for the airplane forward flight power available as a function of airspeed and altitude on a standard day for takeoff and normal rated power. Also shown on those figures is the 1720 horsepower design torque limit. A typical fuel flow carpet plot is presented in Figure V-20 for standard day operation at 10,000 feet. The power available numbers reflect all engine and transmission losses.

Table V-4 shows the losses for each engine used in estimating the installed engine performance. The power available and fuel flow were obtained by inputting these losses into the computer program supplied by the engine manufacturer.

The ram efficiency factor (η_{RAM}) was determined from Section IX, Figure 18 of Reference 33 and is expressed by the relation

$$\eta_{RAM} = 1.0 - 14.4 \left(\frac{W_a}{\sigma' V}\right)^2$$

where V is the velocity in knots, W_a is the weight flow of air in pounds per second, and σ' is the air density ratio.

TABLE V-4

ENGINE LOSSES

Inlet Pressure Loss	5.8 in H_2O SLS
Exhaust Pressure Loss	3.0 in H_2O SLS
Inlet Temperature Rise	2.7°F
Ram Recovery Loss	$(1 - \eta_{RAM})$ q in H_2O SLS
Extracted Horsepower Loss	11.5 hp
Twin Engine Transmission Efficiency	0.98
Single Engine Transmission Efficiency	0.97

E. Helicopter Performance

All helicopter performance is calculated for 30,000 engine rpm and out-of-ground-effect conditions.

1. Hovering Performance

Hover ceilings are shown in Figure V-21 for standard and 95°F day conditions. Included in these data is a seven-percent download that is experienced by the vehicle in the hovering mode. The effect of this download is made evident in Figure V-22 which shows both the isolated proprotor and the overall Figure of Merit as a function of the gross weight. Also included in the overall Figure of Merit is a 0.98 transmission efficiency.

All data shown in Figures V-21 and V-22 were derived from the isolated proprotor data shown in Figure V-12 and the power available data shown in Figure V-16.

2. Helicopter Level Flight Power Required

Figure V-23 shows the helicopter level flight power required for sea-level standard-day operations and for a gross weight range of 8500 to 13,500 pounds. Included in these data is the sharing of the total vehicle lift between the airframe and proprotors as a function of airspeed as shown in Figure V-24. A 15-percent decrease in rotor induced power was also used to account for the side-by-side effect of the proprotors. All data were determined for a 15-degree mast angle and a 30-degree flaps down condition.

3. Rate of Climb

Standard day maximum rate of climb at normal rated power as a function of altitude is shown in Figure V-25 for a range of weights. These values were determined by using the relation:

$$R/C \text{ (helicopter mode)} = \frac{\text{excess power} \times 33000 \times 0.85}{\text{gross weight}} \text{ ft/min}$$

where excess power is the difference between normal rated power available and minimum power required. The 0.85 factor is a climb efficiency number.

F. Airplane Performance

1. Thrust Horsepower Required

Data for thrust power required versus airspeed for sea level, 10,000 feet and 20,000 feet on standard day conditions are shown in Figures V-26 through V-28 for several gross weights. These data were determined using the airframe lift and drag coefficients shown in Figures V-2 through V-4 and the expression:

$$\text{Thrust horsepower required} = \frac{DV}{550}$$

where

$$D = C_D \, qS .$$

Figures V-29 through V-31 contain the above mentioned thrust horsepower required data expression as proprotor shaft horsepower. The relation used for this conversion was:

$$\text{Proprotor shaft horsepower required} = \frac{\text{thrust horsepower required}}{\eta_{prop}}$$

where η_{prop} is the proprotor propulsive efficiency, obtained from Figures V-13 through V-15.

2. Flight Envelope and Maximum Speed

Figure V-32 contains the standard day airplane flight envelope for normal power rating and the maximum speed for takeoff power rating, both for a gross weight of 10,300 pounds. The low-speed end of the flight envelope is governed by wing stall and the high-altitude end by the absolute ceiling. The maximum speed was determined by the intersection of the power available and power required curves for the given gross weight of 10,300 pounds.

3. Maximum Rate of Climb

Rate of climb in the airplane mode is calculated using the relation:

$$\text{R/C (airplane mode)} = \frac{\text{excess thrust horsepower x 33000}}{\text{gross weight}}$$

where excess thrust horsepower is the difference between the thrust horsepower available and required. The maximum rate of climb for normal rated power is plotted versus altitude for various gross weights and is shown in Figure V-33 for standard day operations.

4. Specific Range

Figures V-34 through V-36 contain nautical miles per pound of fuel versus true airspeed for various gross weights at sea level, 10,000 feet and 20,000 feet on a standard day. These curves were prepared from the fuel flow and power required data.

5. Single Engine Performance

Figure V-37 shows the single engine performance for both the helicopter and airplane modes superimposed on a twin engine hovering ceiling. For this data a 150 and 200 foot-per minute rate of climb was maintained for the helicopter and airplane modes respectively.

6. Mission Profiles

Mission capability of the aircraft is illustrated in Figures V-38 through V-41. Mission profiles are shown for the following:

- Research mission
- Civil mission
- Military mission
- Ferry mission

The civil and military missions are shown to illustrate the capability of the Model 300 to demonstrate economic feasibility on simulated missions. The civil mission is an intercity commuter flight of 8 passengers and 240 pounds of baggage. Range is 306 nautical miles (350 statute miles) at a cruise speed of 260 knots (300 mph). The military mission is an aircrew recovery. A dash speed of 300 knots is used to traverse 375 nautical miles, pick up three survivors, and return at long-range cruise speed. Takeoff weight includes a crew of three and 550 pounds of armor and rescue equipment. At the mid-point pickup the aircraft can hover 4000 feet on a 95°F day or 9000 feet on a standard day.

The research and ferry missions are performed at long-range cruise speed; whereas, the civil and military missions are flown at higher speeds more appropriate for the particular mission. Payload range curves are shown in Figures V-38 and V-41 for the takeoff weights of the research and ferry missions.

7. STOL Performance

STOL takeoff distance to clear a 50-foot obstacle is shown in Figure V-42 for a STOL takeoff at 4000 feet on a 95°F day. The following takeoff technique is assumed: pylons are rotated to a 20 degree conversion angle, maximum power and full forward cyclic are applied as the brakes are released, the aircraft accelerates to 70 knots, aft cyclic is applied to rotate the aircraft and lift off, climb out is made at 70 knots. The pylon tilt and forward cyclic tilt the tip path plane 30 degrees. The 70-knot lift-off speed was selected to provide 1.25g maneuver capability at the maximum gross weight. A lift off at lower speed would decrease the takeoff distance. Takeoff distance is 900 feet at 12,400 pounds and 700 feet at the 10,300 pound weight where the aircraft can hover out of ground effect.

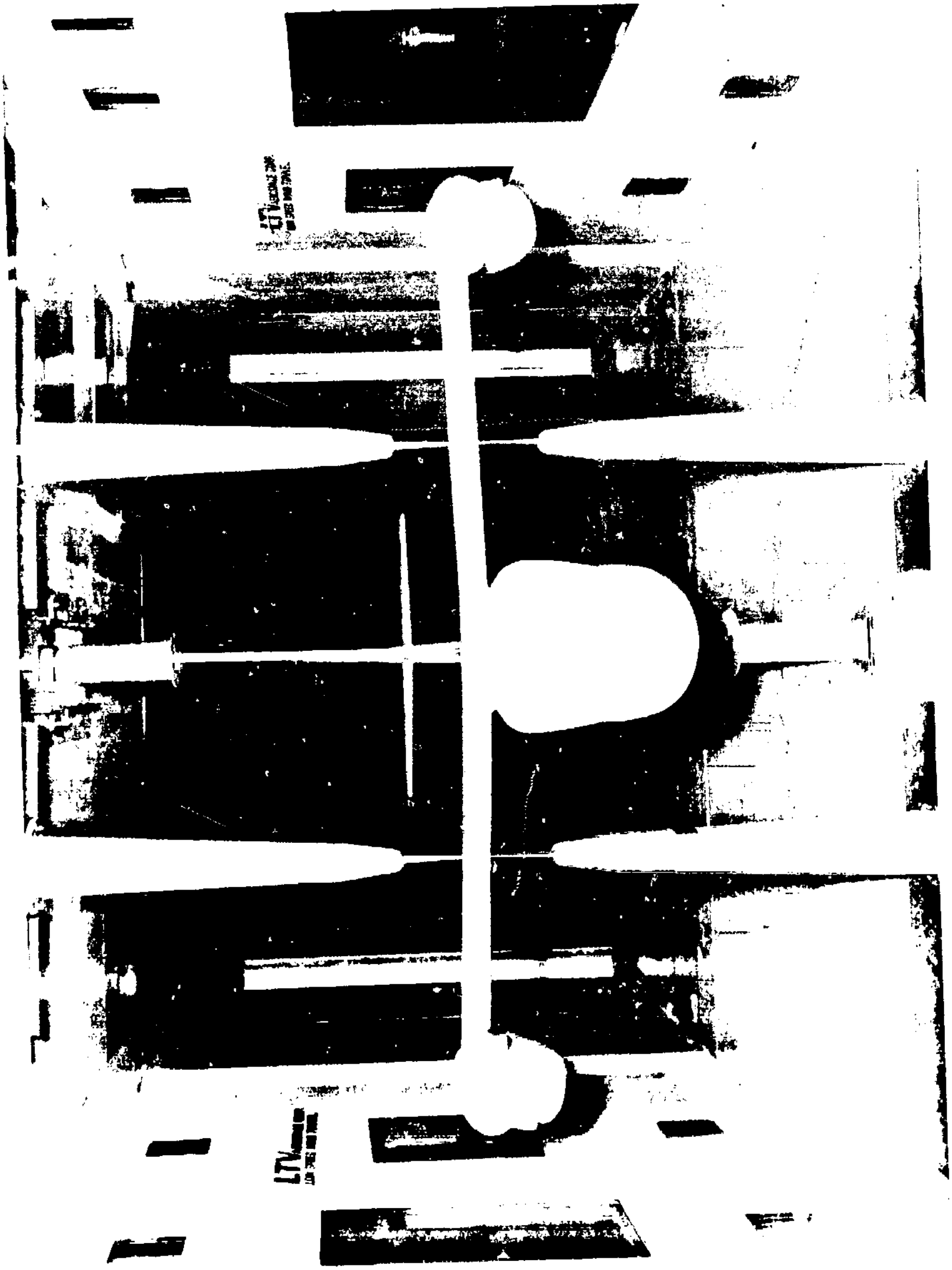


Figure V-1. Model 300 Aerodynamic Model in 7-by-10-Foot Tunnel.

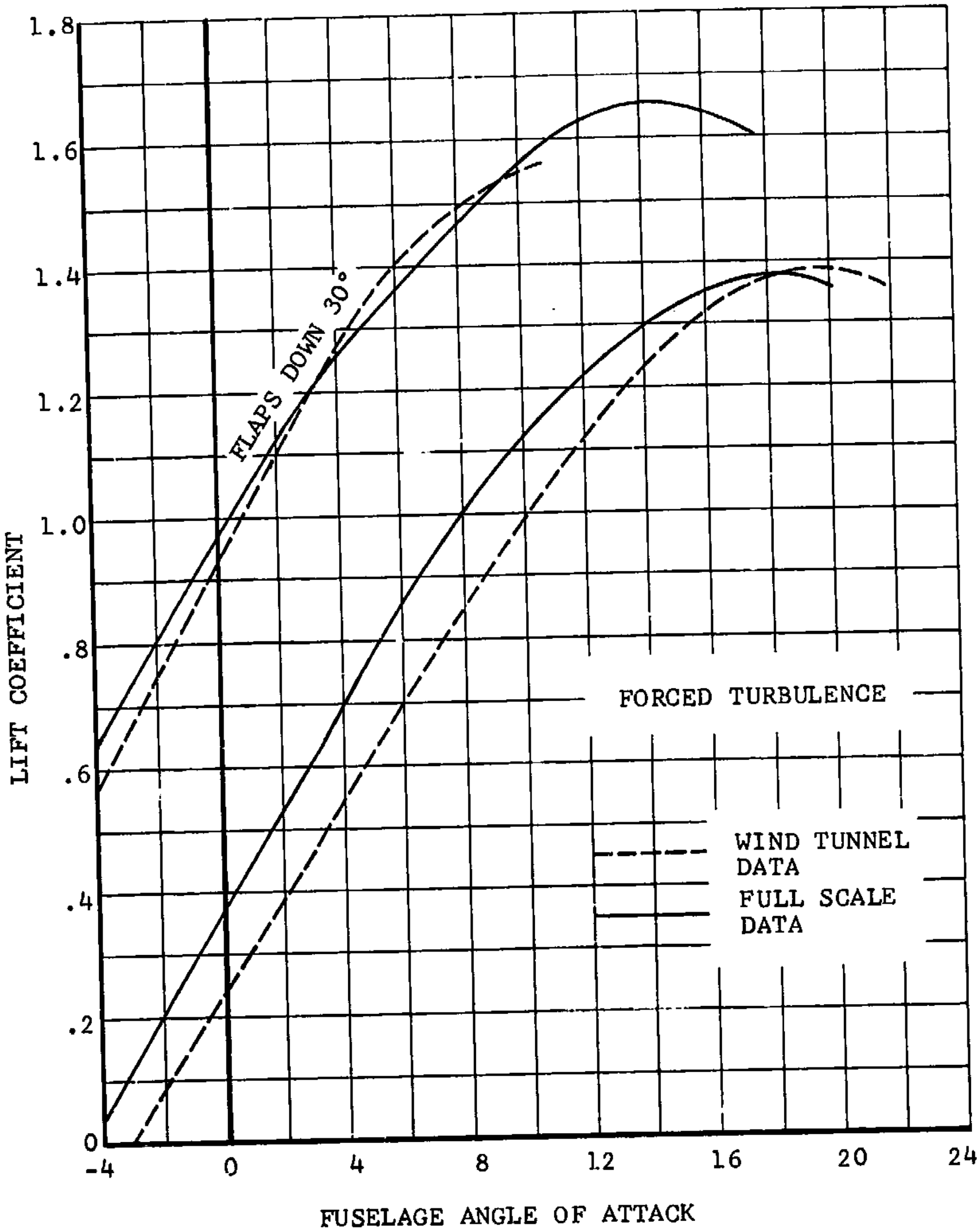


Figure V-2. Airframe Lift Coefficient Versus Fuselage Angle of Attack.

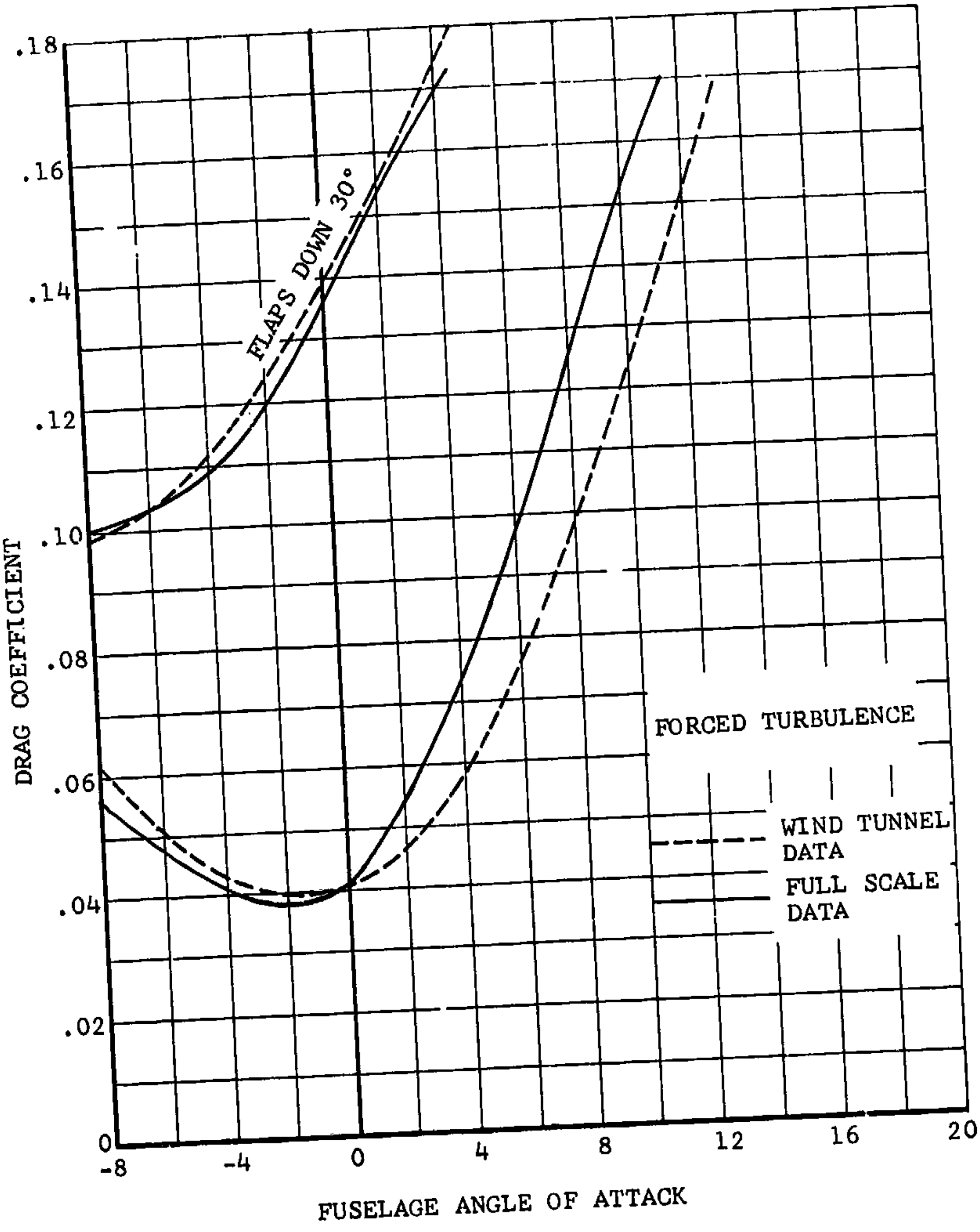


Figure V-3. Airframe Drag Coefficient Versus Fuselage Angle of Attack.

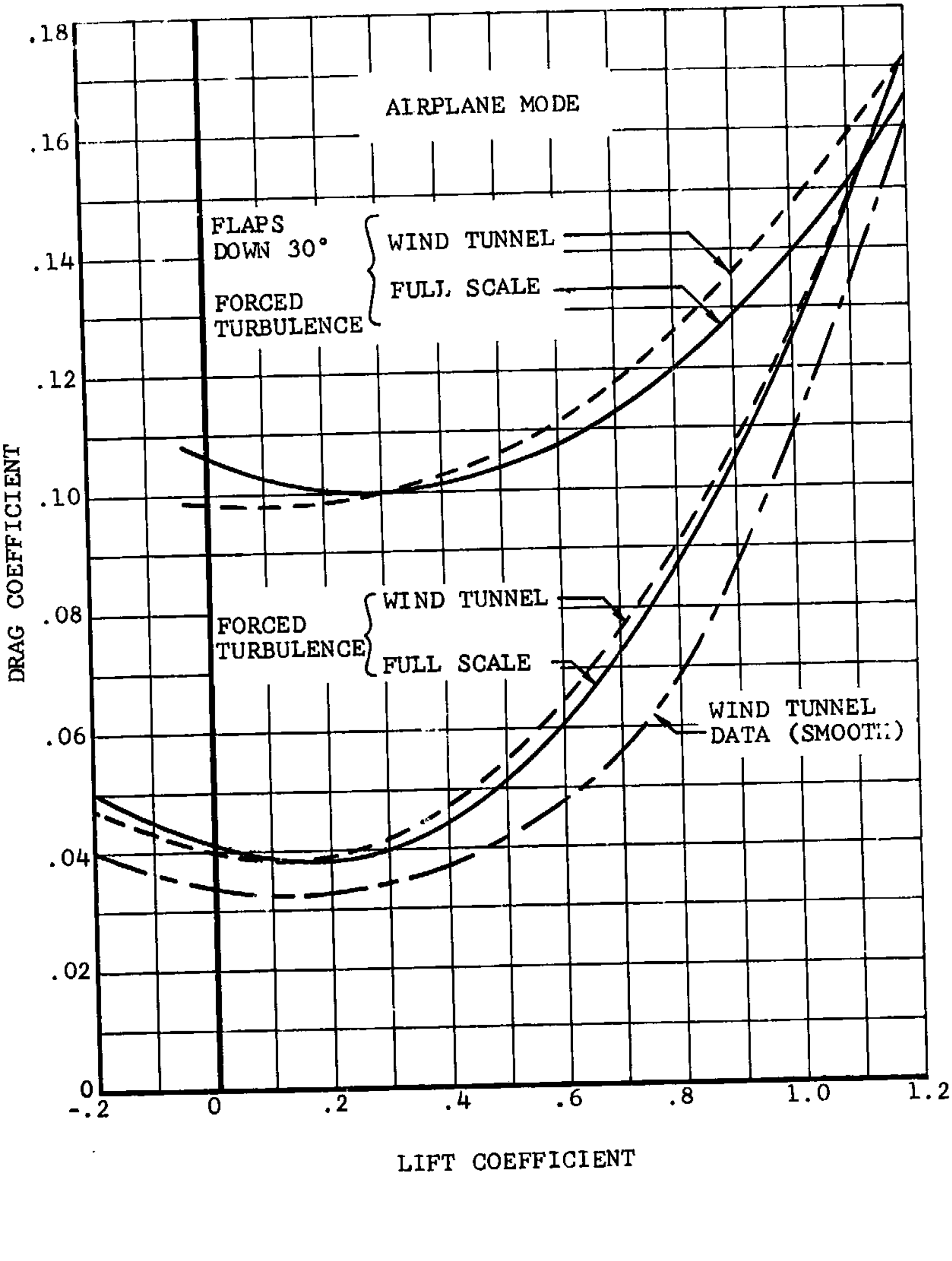


Figure V-4. Airframe Drag Coefficient Versus Lift Coefficient.

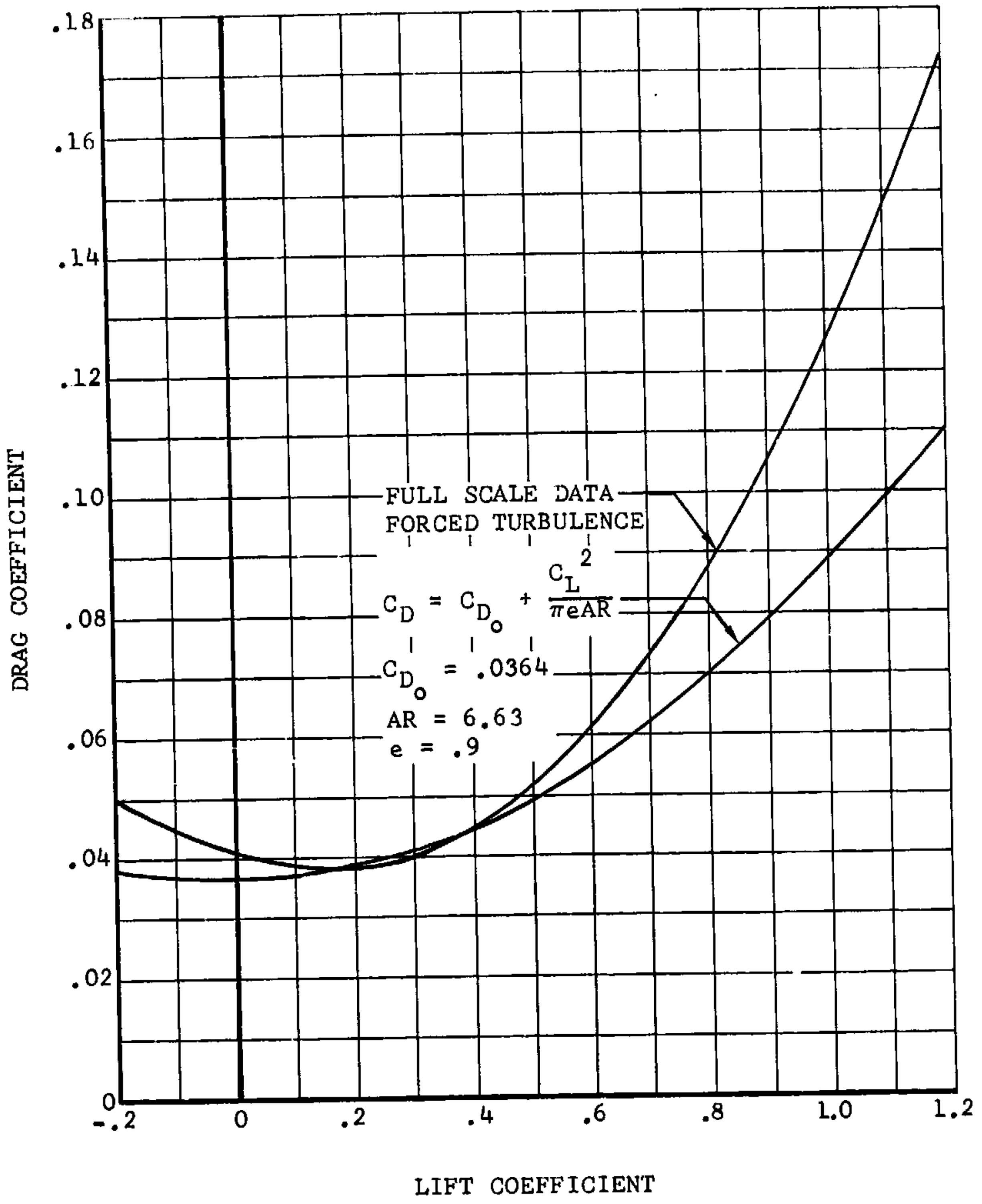


Figure V-5. Comparison of Full-Scale Aircraft Drag, Based on Tunnel Results and Calculated Data.

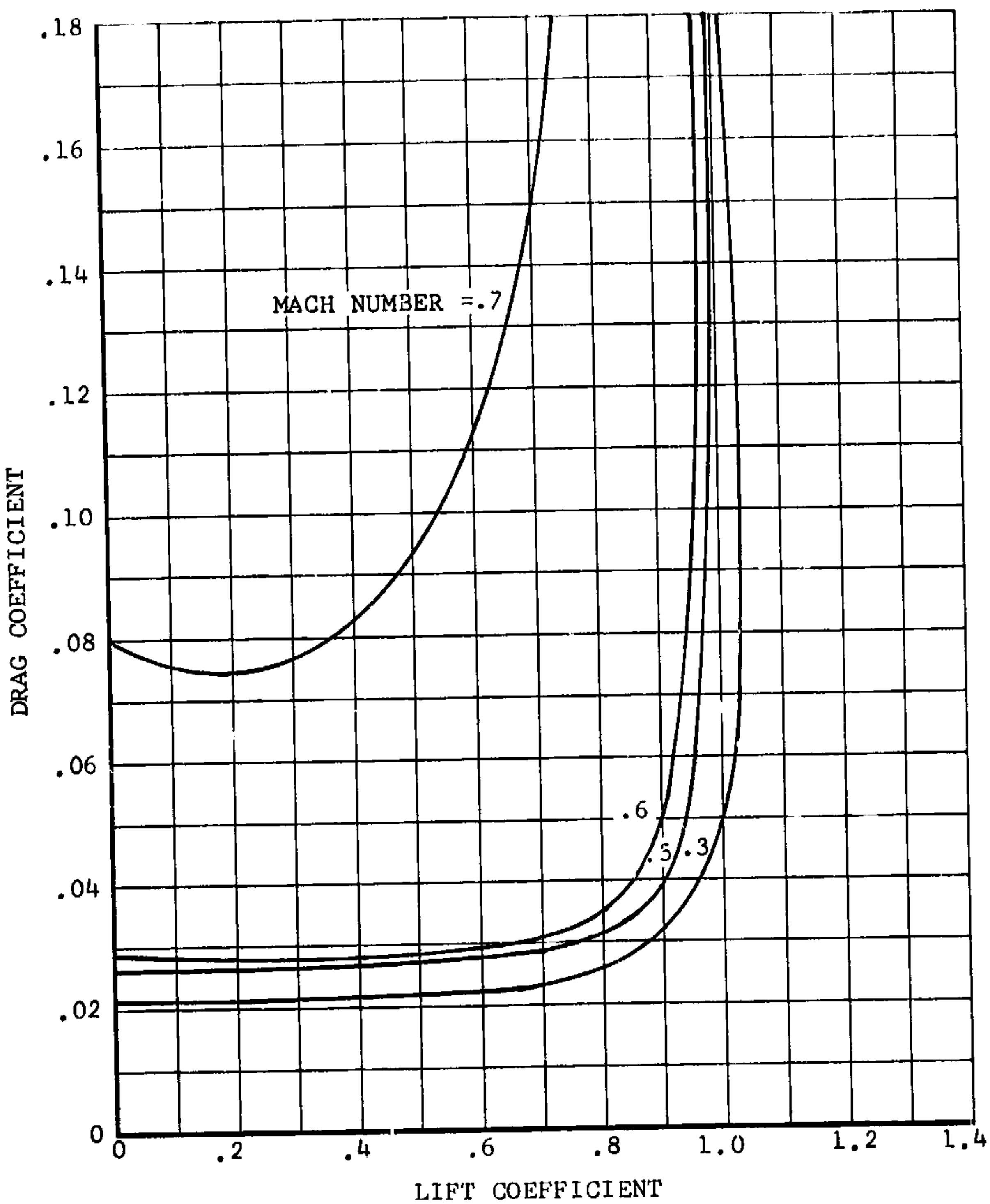


Figure V-6. Proprotor Blade Station Data, Radial Station 0.075 to 0.45.

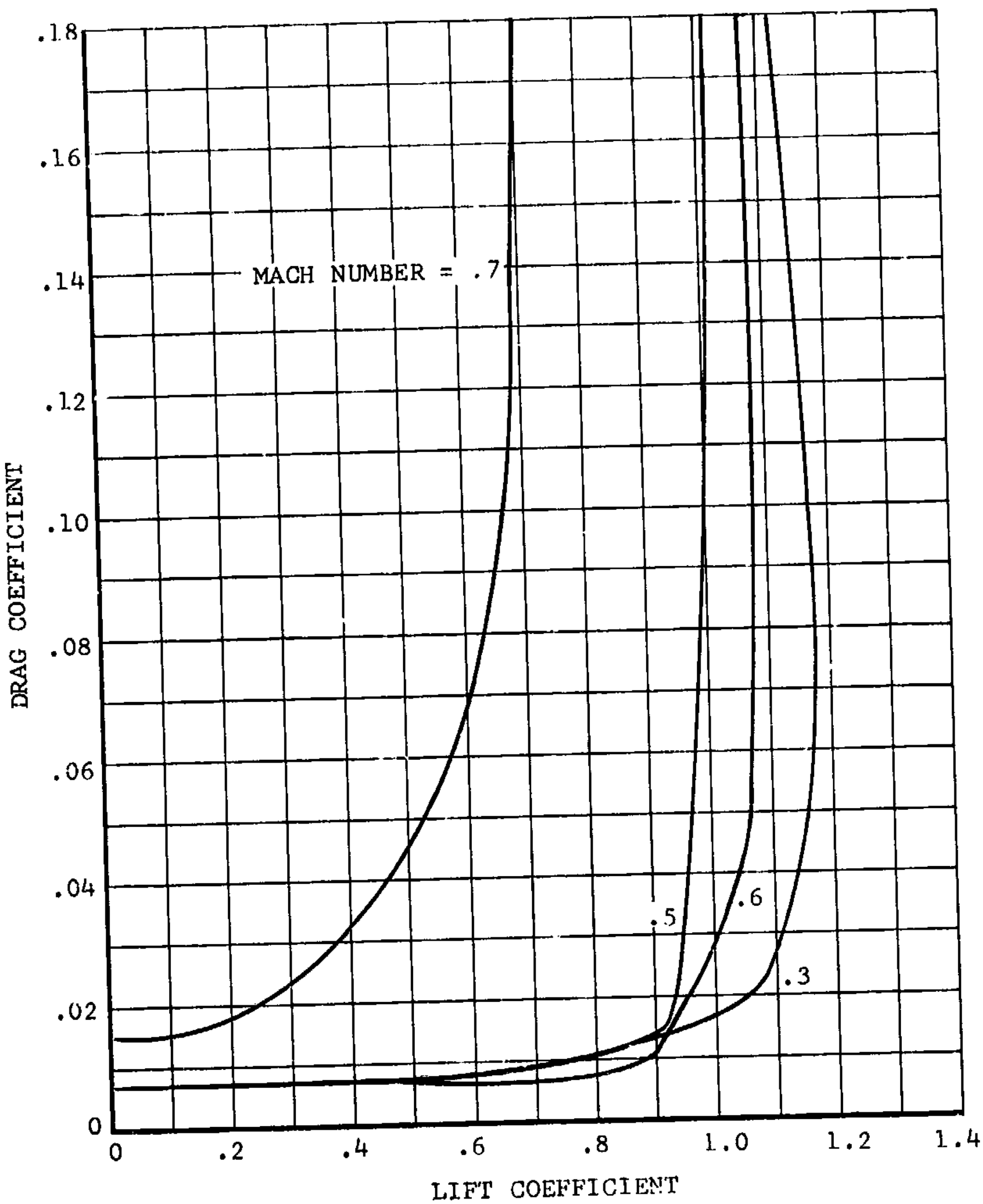


Figure V-7. Proprotor Blade Section Data Blade Station 0.45 to 0.70.

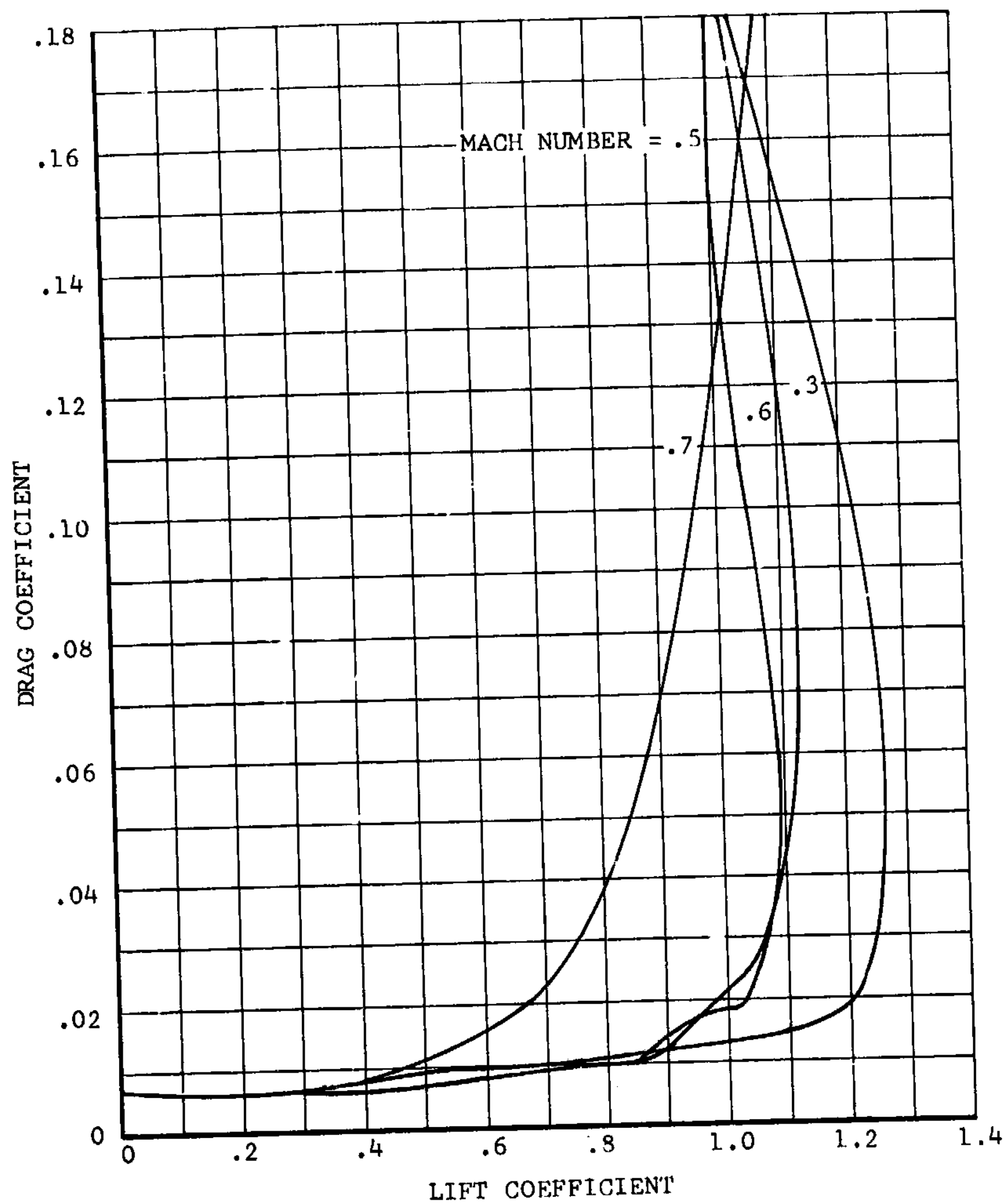


Figure V-8. Proprotor Blade Station Data, Radial Station 0.70 to 0.90.

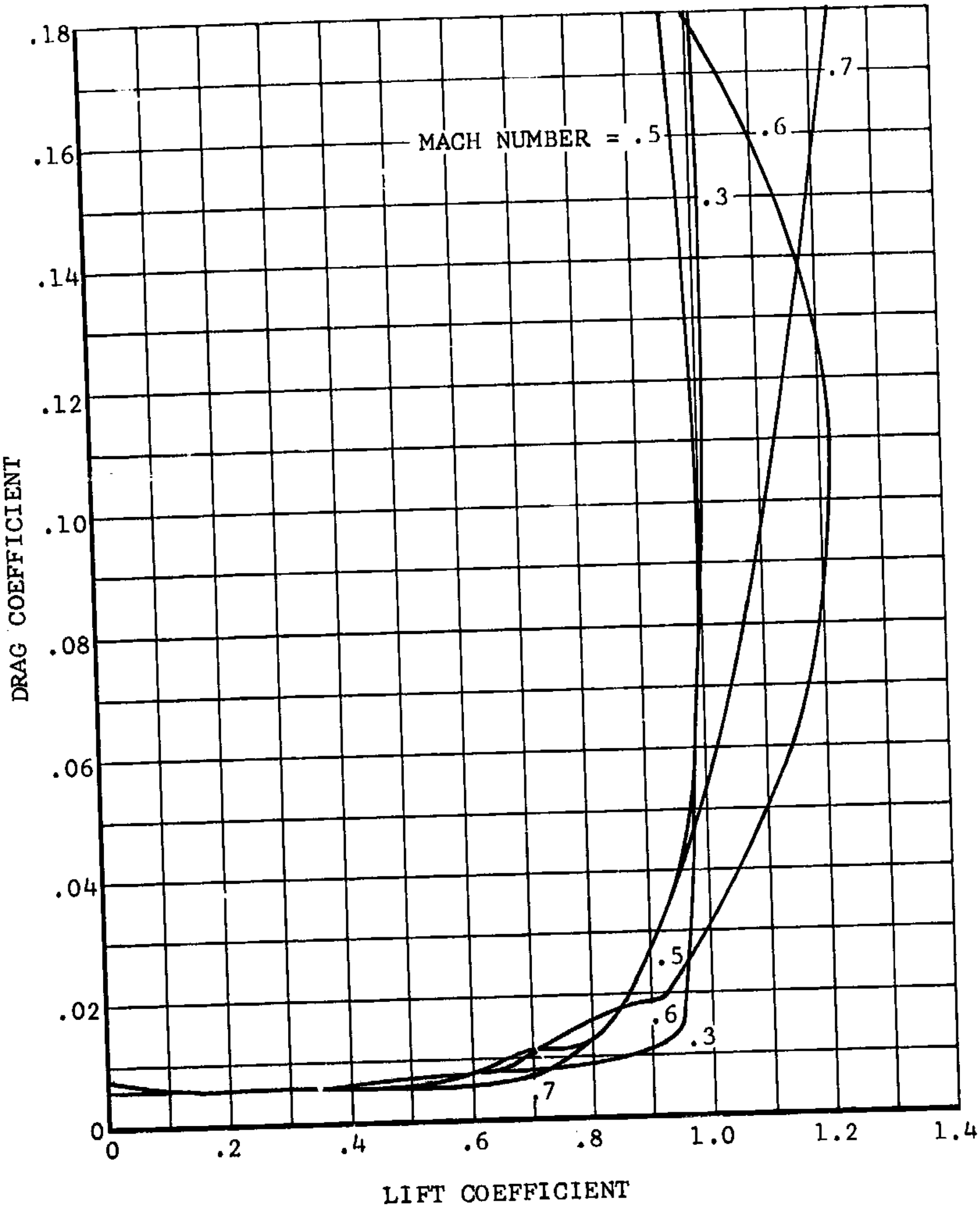


Figure V-9. Proprotor Blade Section Data, Radial Station 0.90 to 1.00.

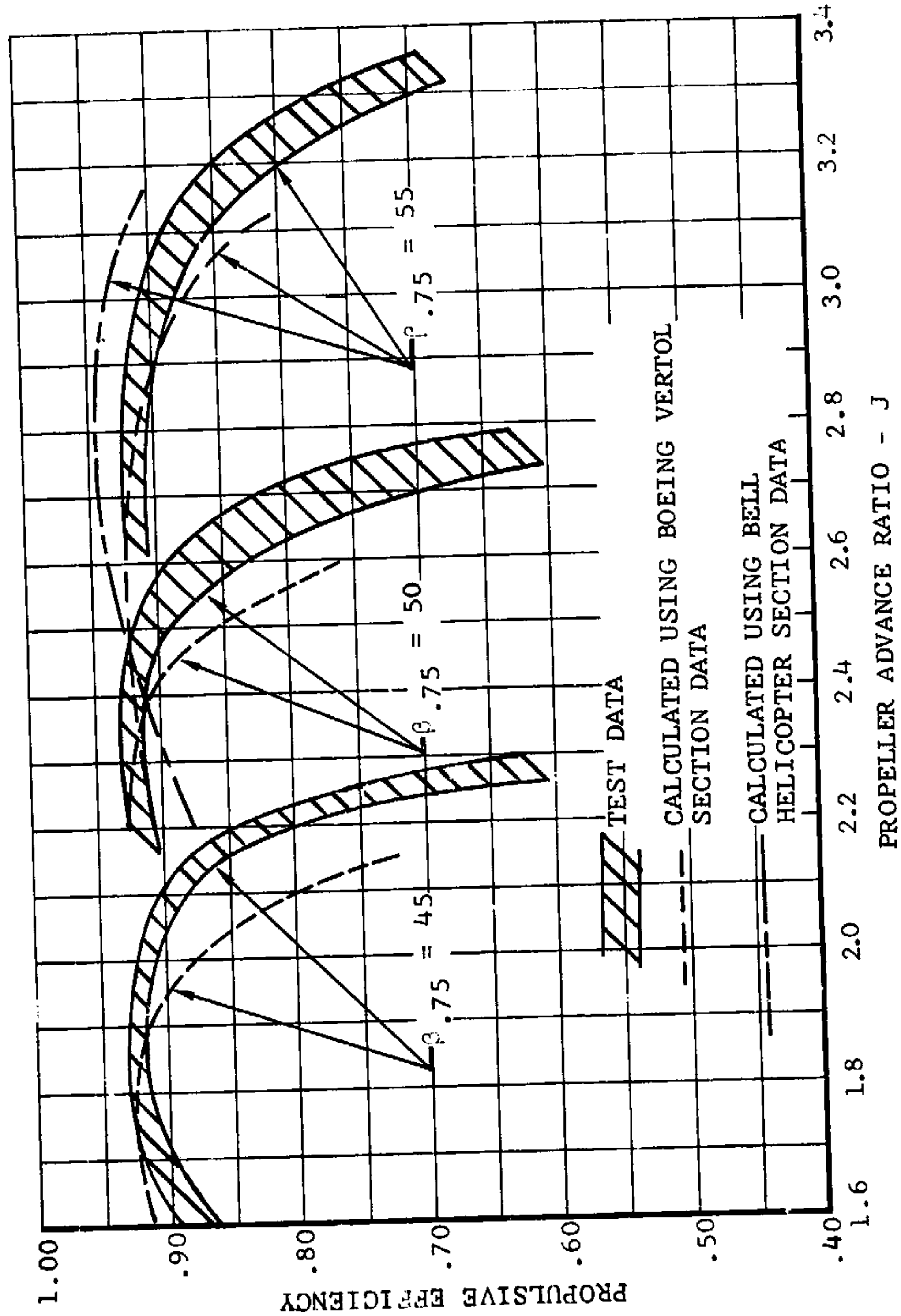


Figure V-10. Proprotor Performance Correlation.

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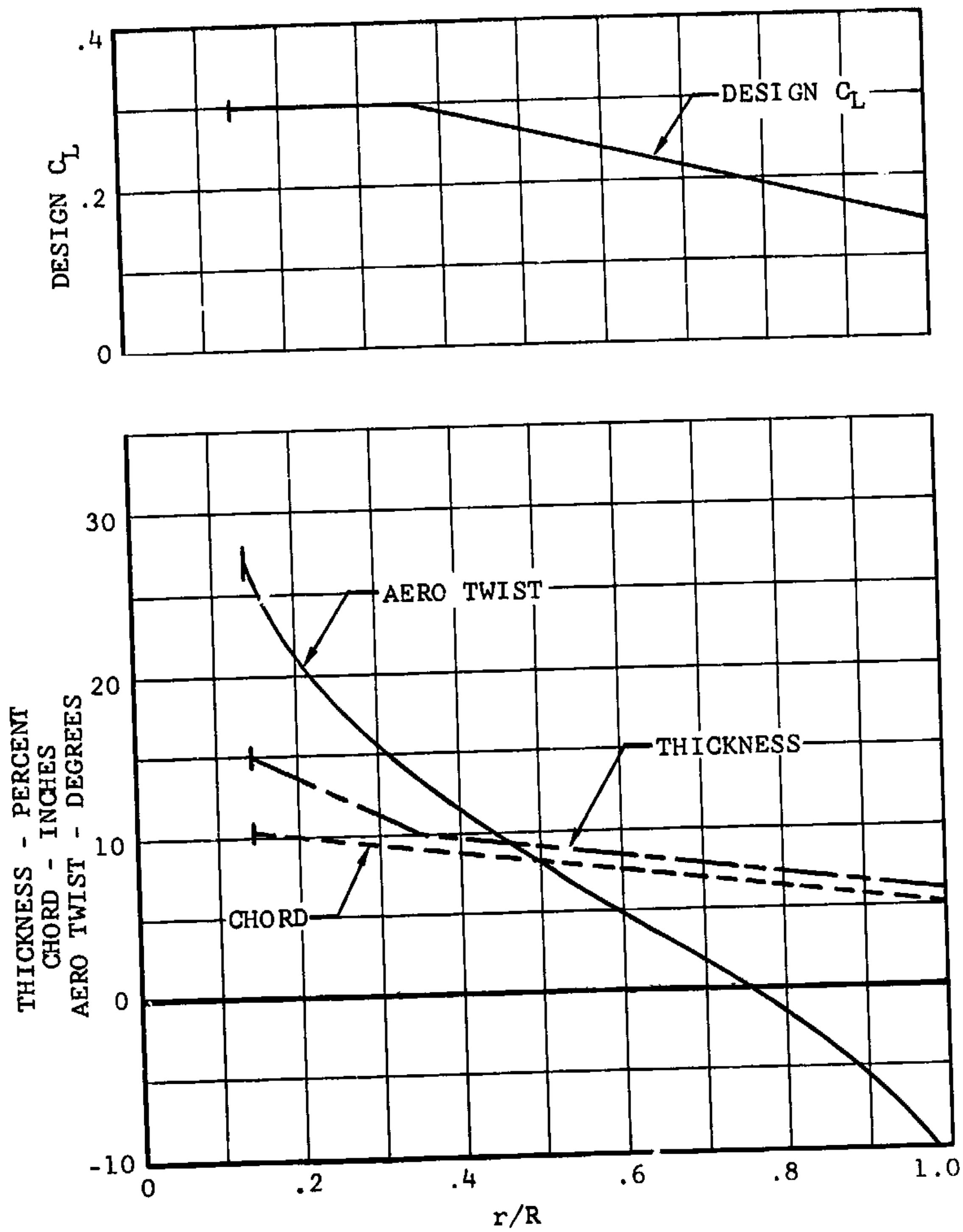

Figure V-11. Boeing-Vertol 13-Foot Diameter Proprotor Blade Parameters (Blade "E").

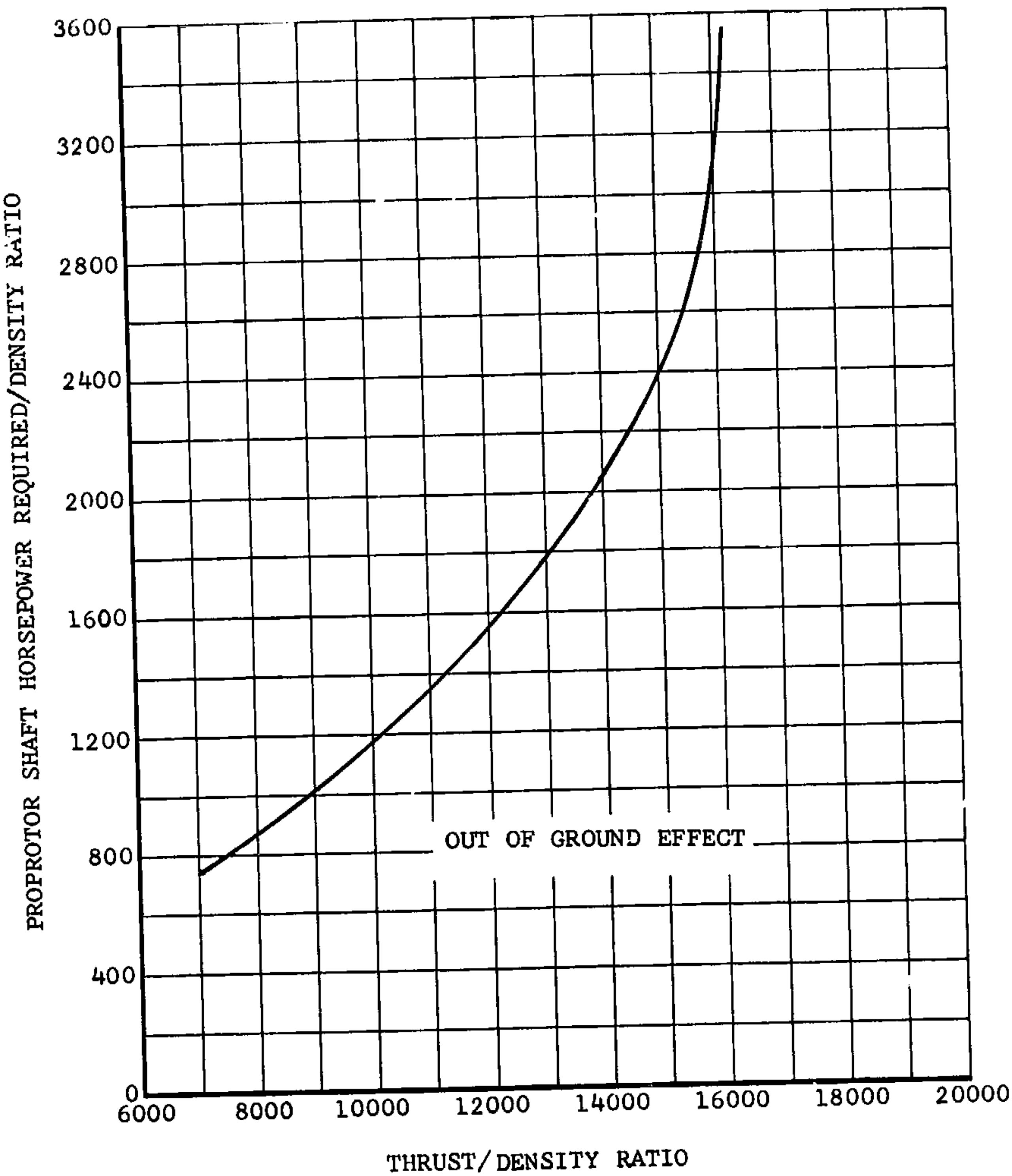

Figure V-12. Proprotor Hovering Power Required Versus Thrust.

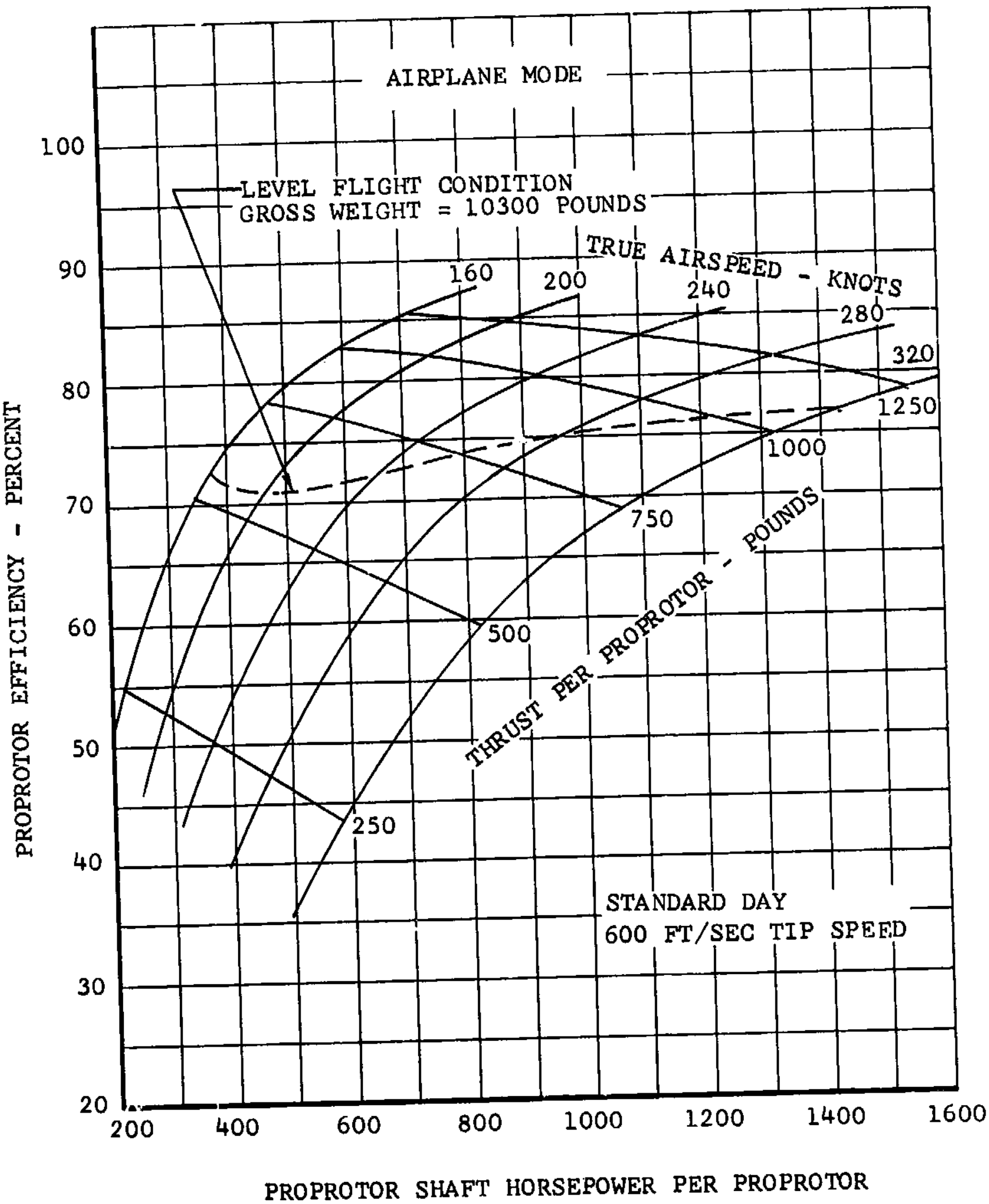

Figure V-13. Proprotor Efficiency Versus Shaft Horsepower Sea Level.

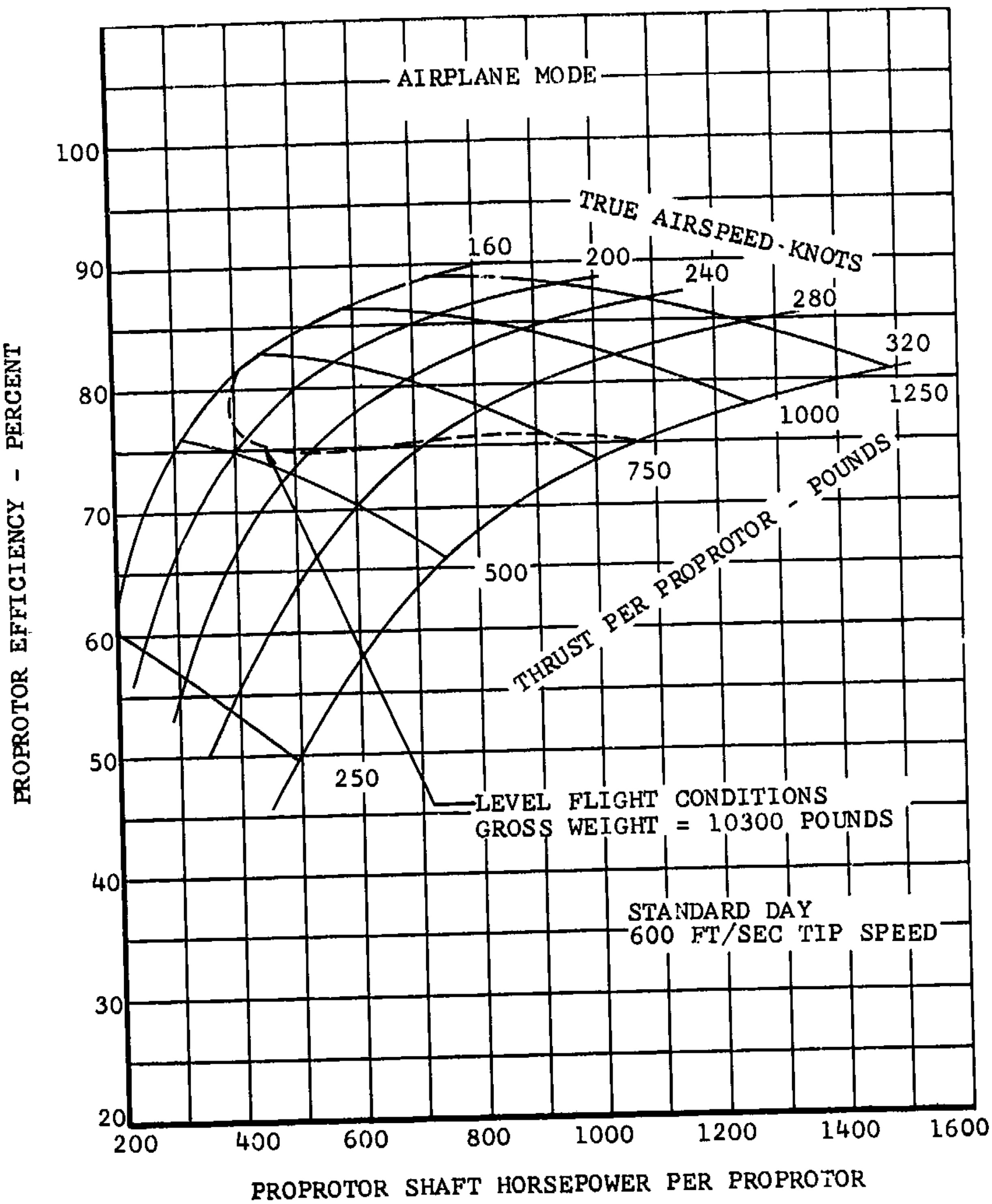

Figure V-14. Proprotor Efficiency Versus Shaft Horsepower 10,000 Feet.

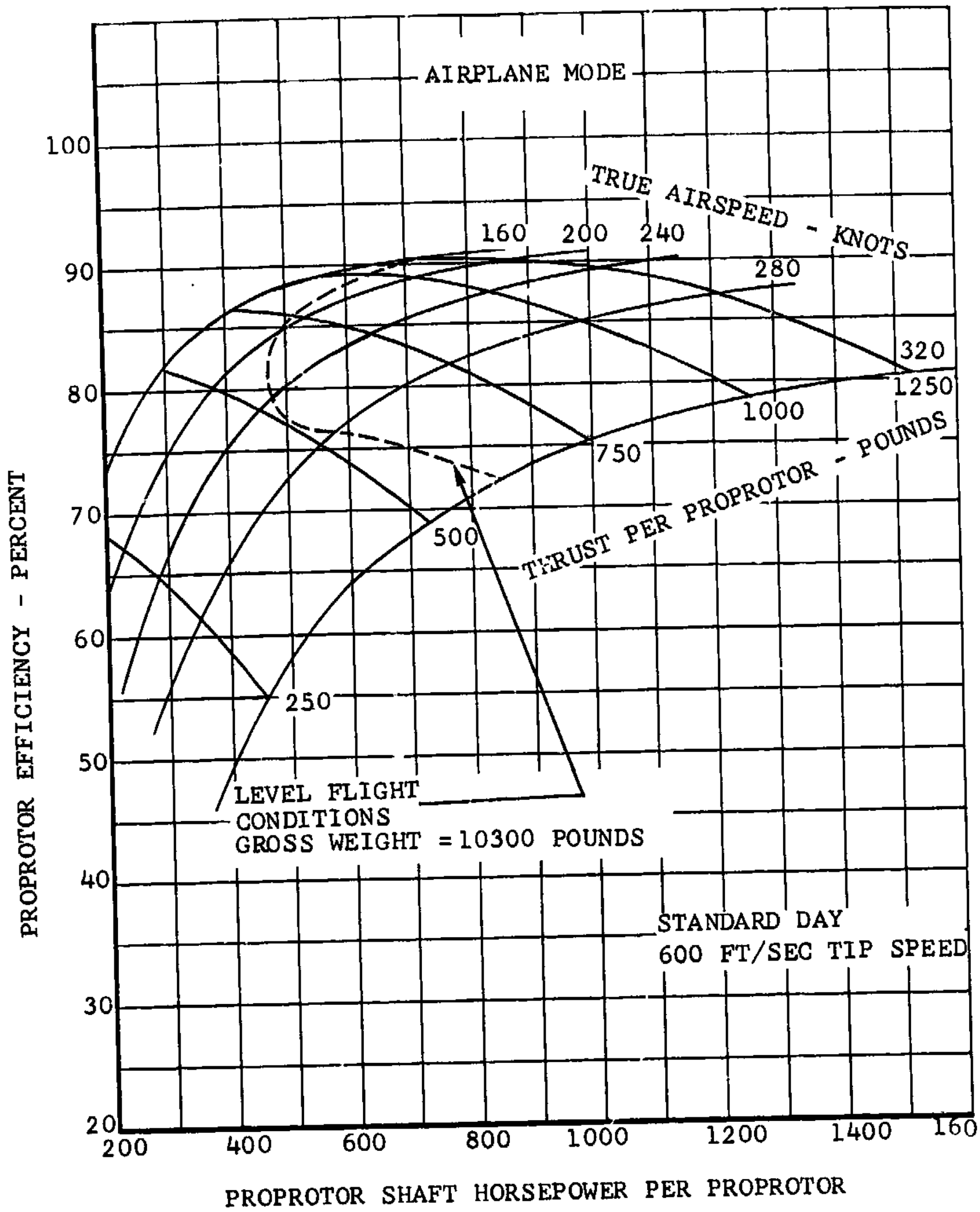

Figure V-15. Proprotor Efficiency Versus Shaft Horsepower 20,000 Feet.

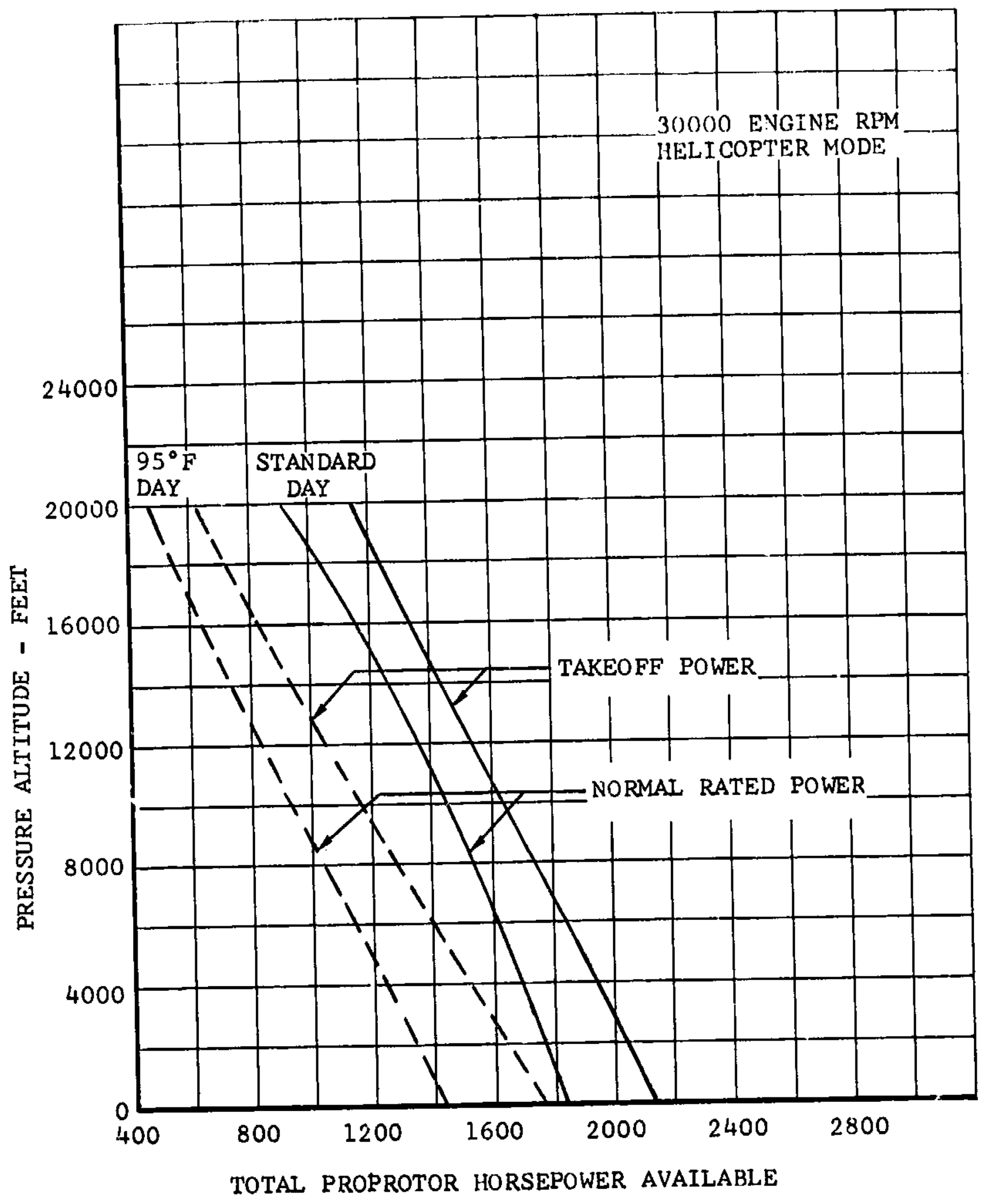

Figure V-16. Twin Engine Proprotor Shaft Horsepower Available, Helicopter Mode.

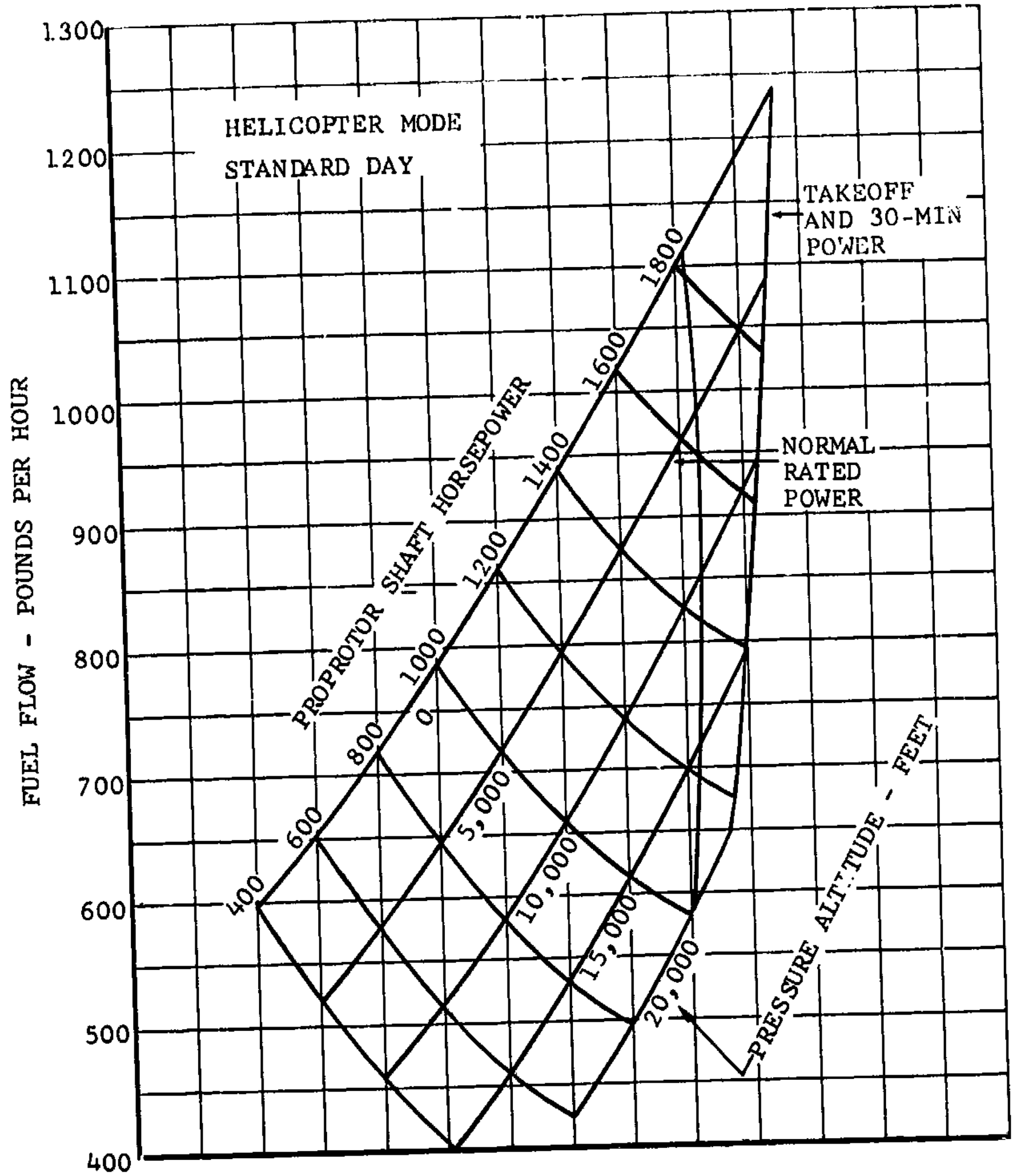

Figure V-17. Fuel Flow, Helicopter Mode.

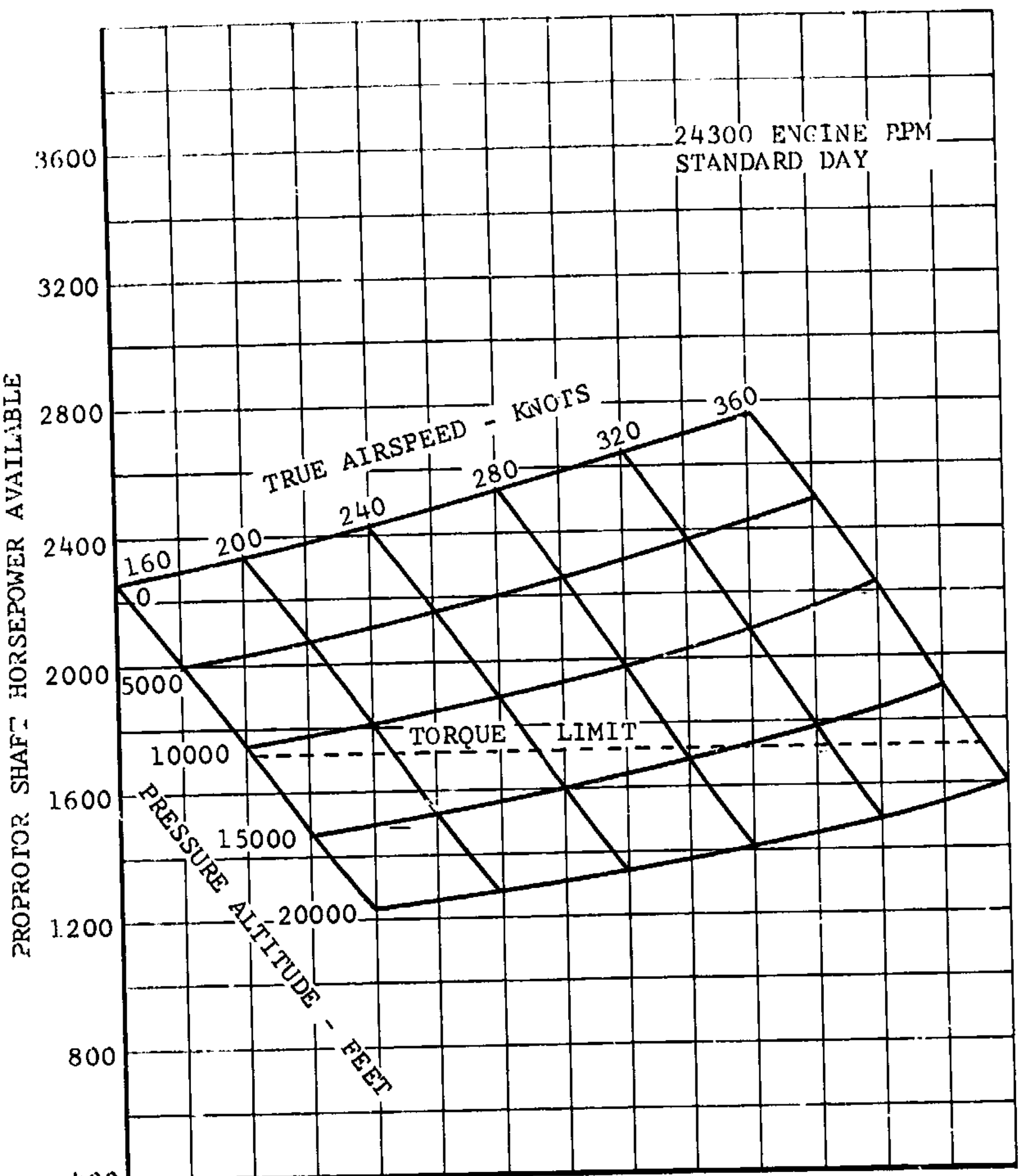

Figure V-18. Proprotor Shaft Horsepower Available, Takeoff and 30-Minute Power.

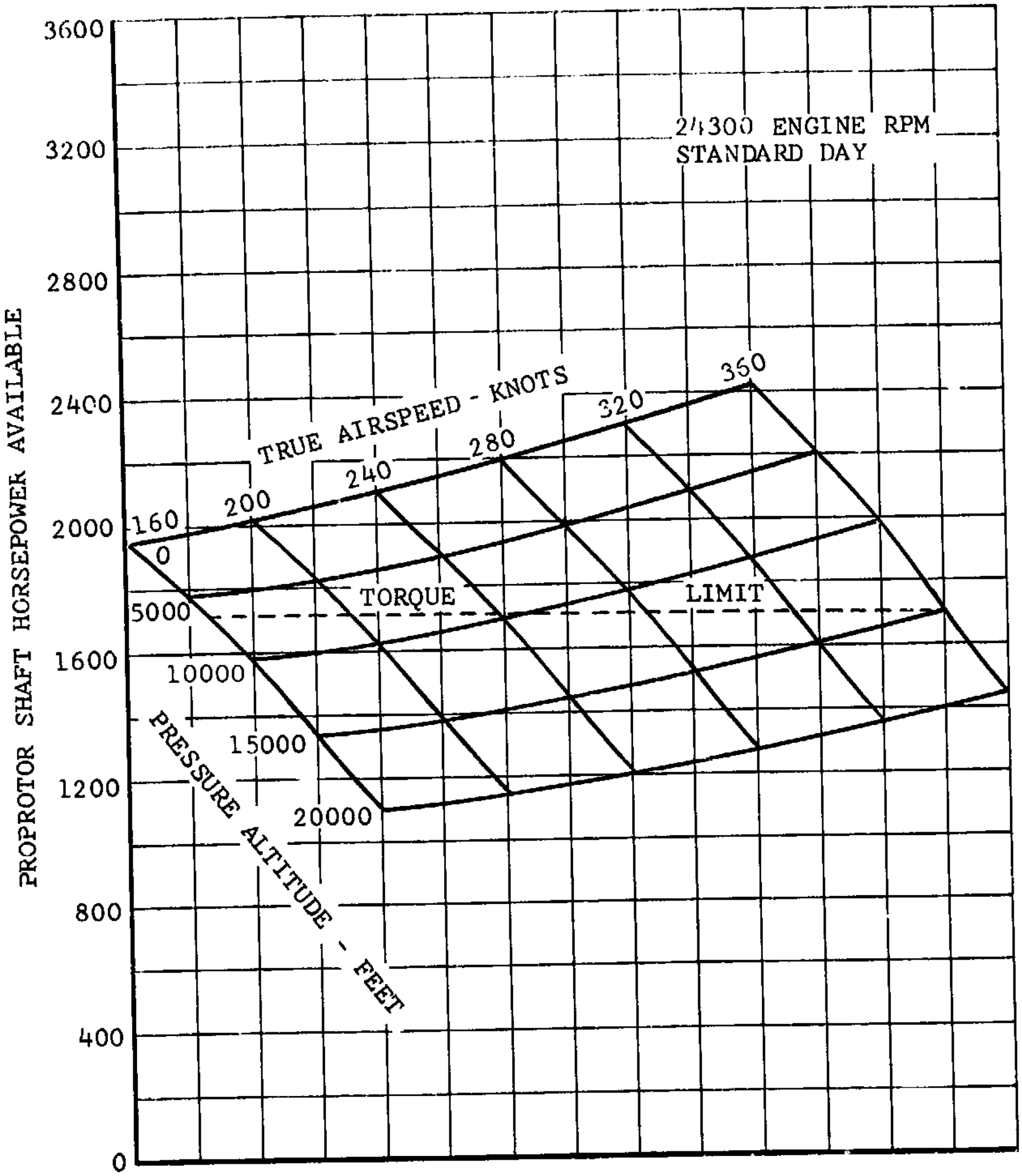

Figure V-19. Proprotor Shaft Horsepower Available, Normal Rated Power

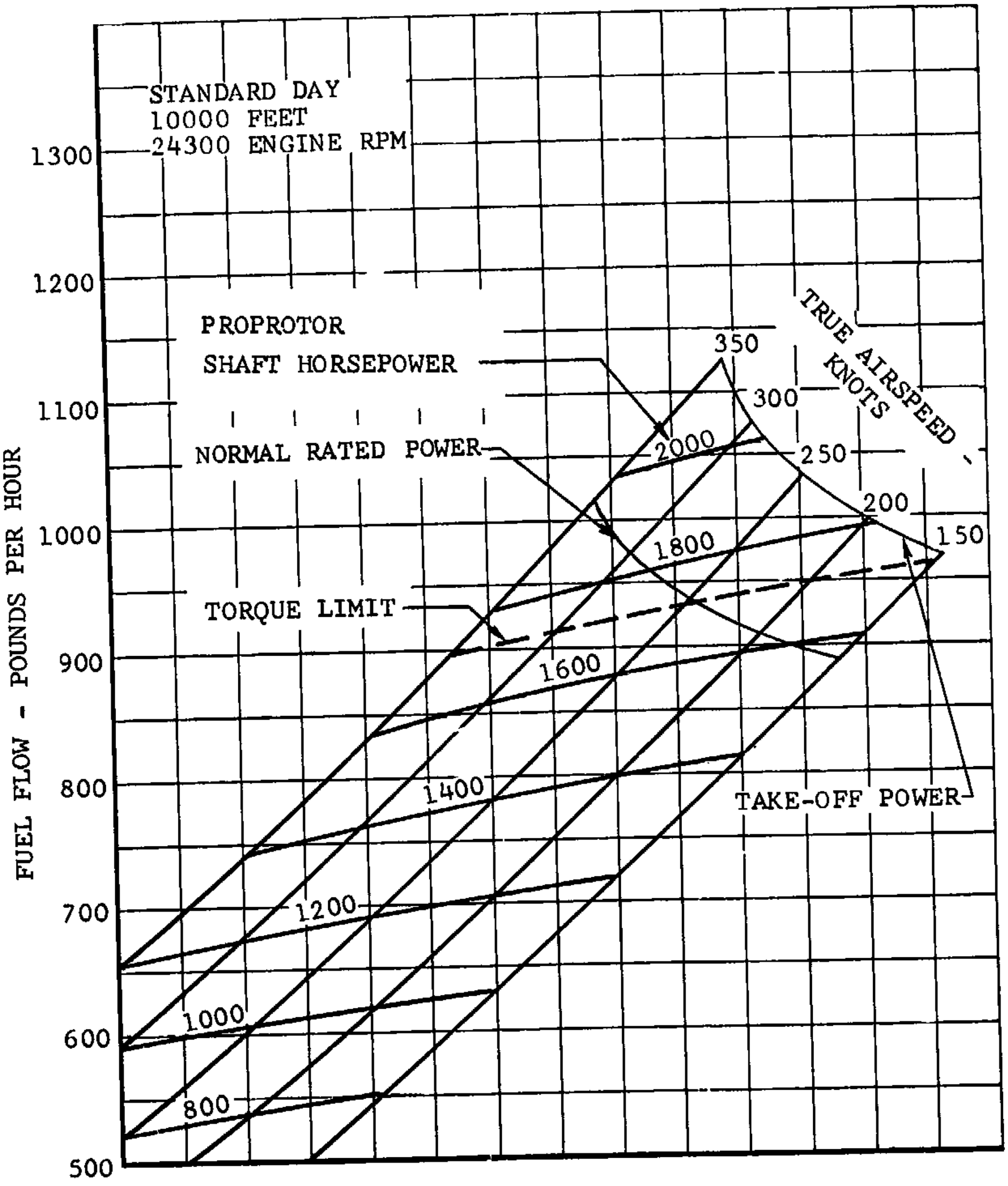

Figure V-20. Fuel Flow, Airplane Mode.

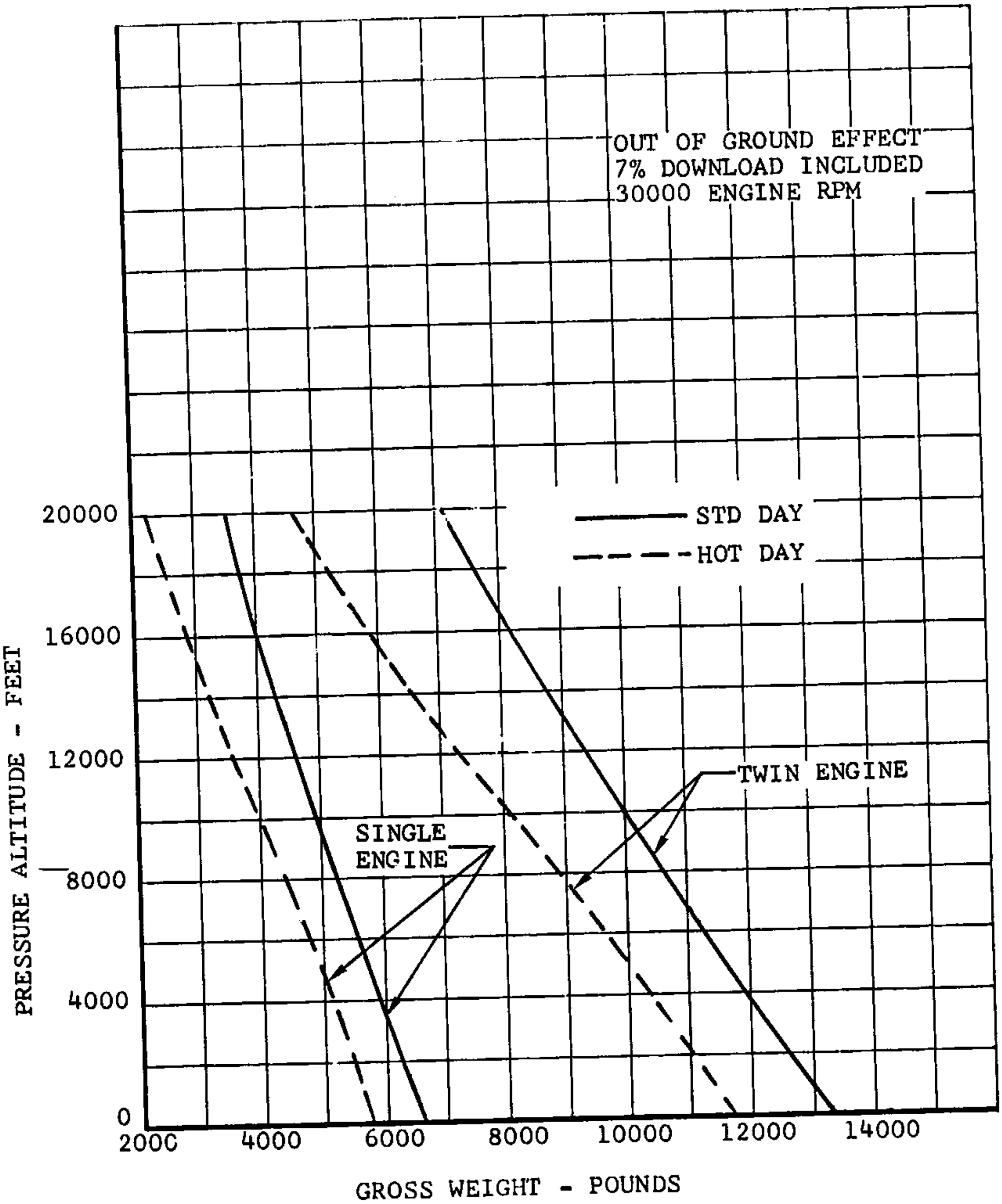

Figure V-21. Hover Ceilings.

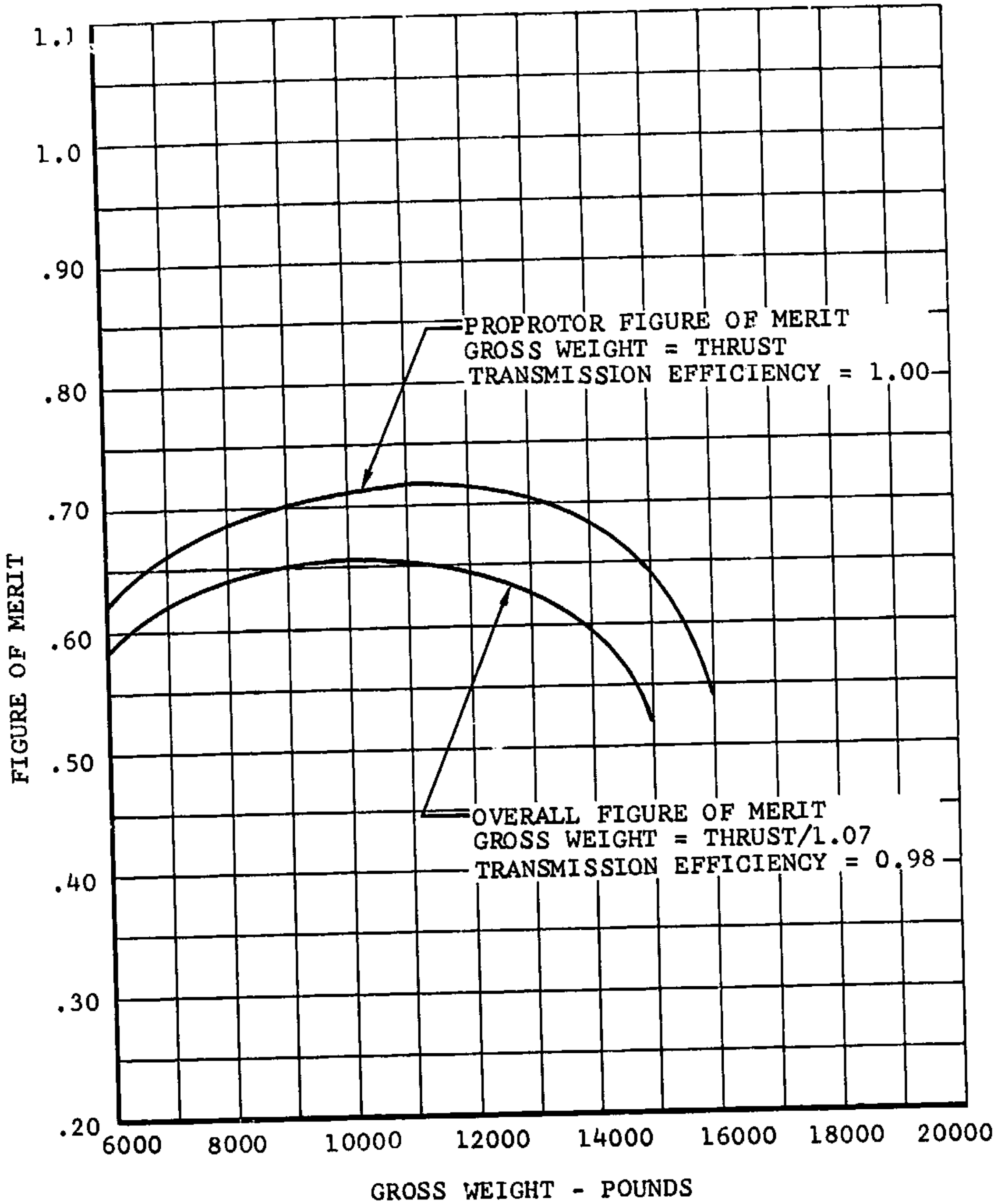

Figure V-22. Hovering Figure of Merit.

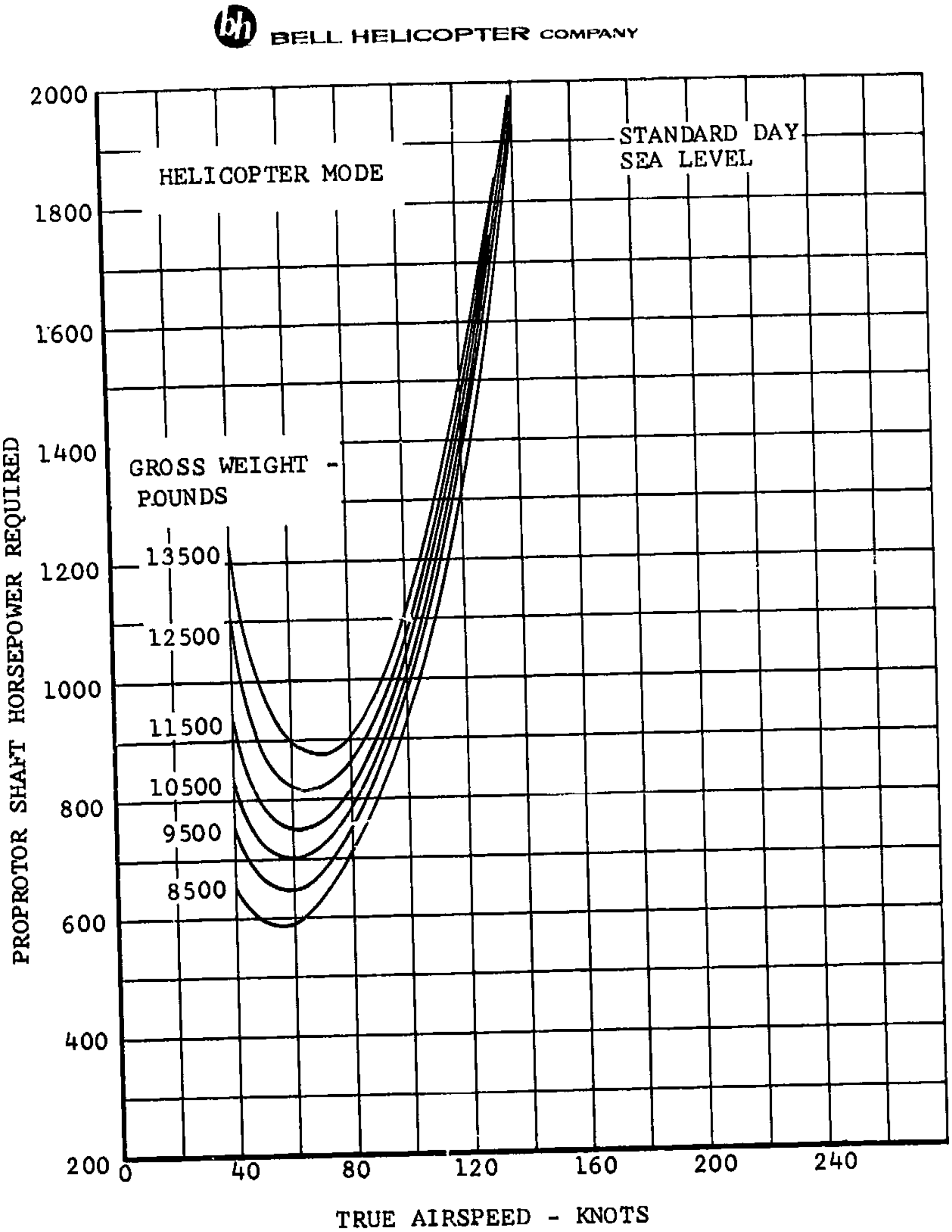

Figure V-23. Proprotor Shaft Horsepower Required Versus True Airspeed, Helicopter Mode.

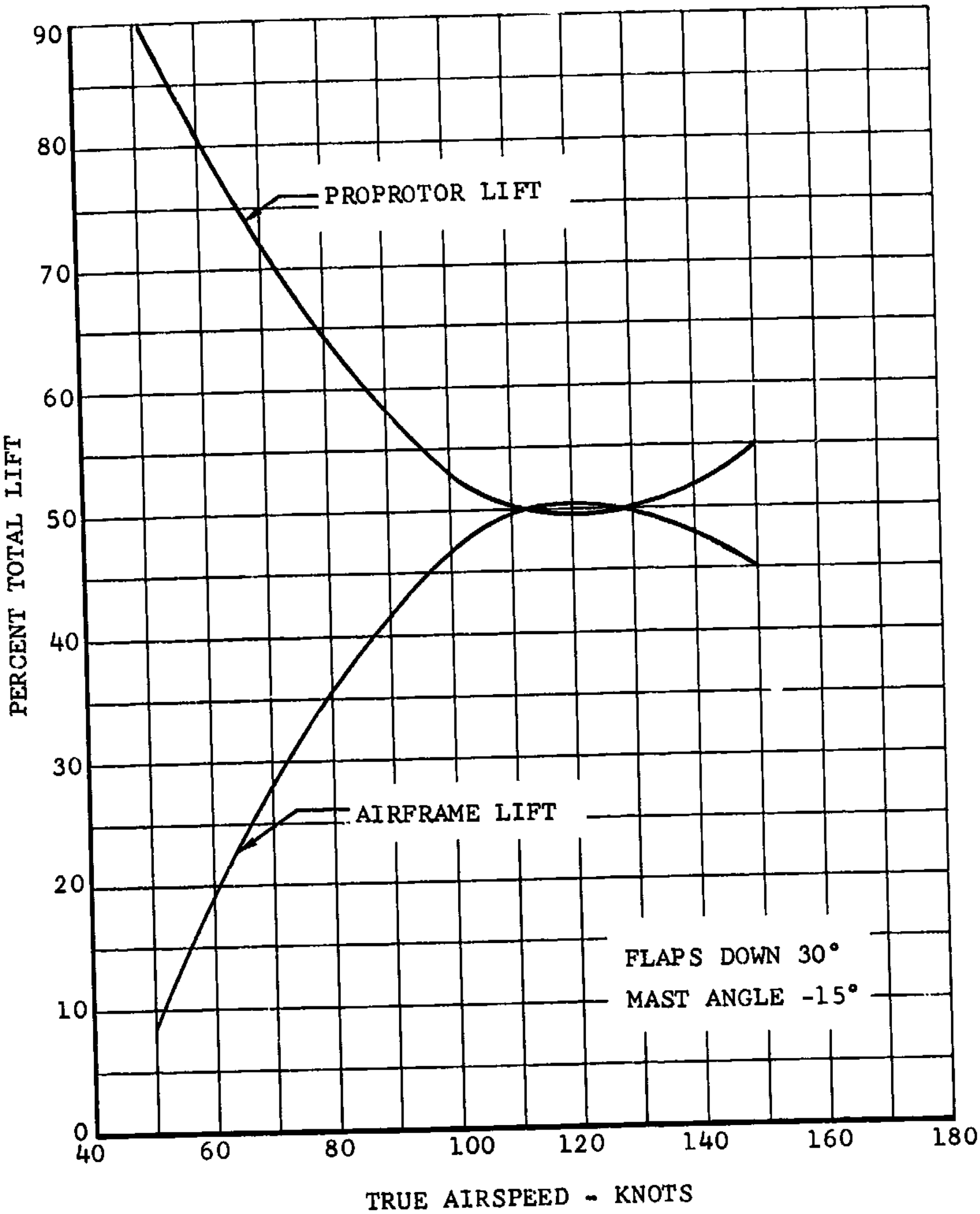

Figure V-24. Lift Distribution Between Proprotor and Airframe in Helicopter Level Flight.

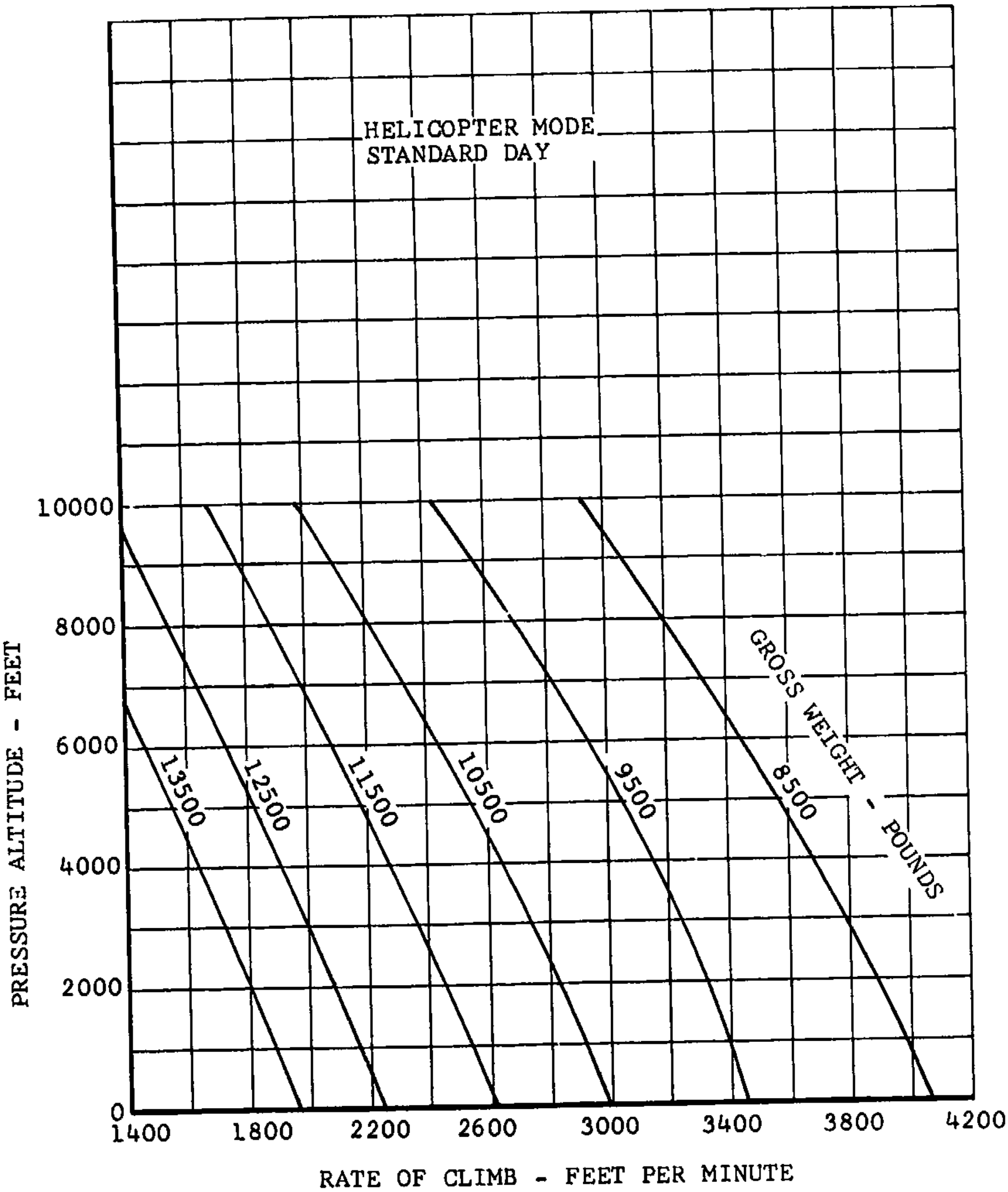

Figure V-25. Rate of Climb, Helicopter Mode.

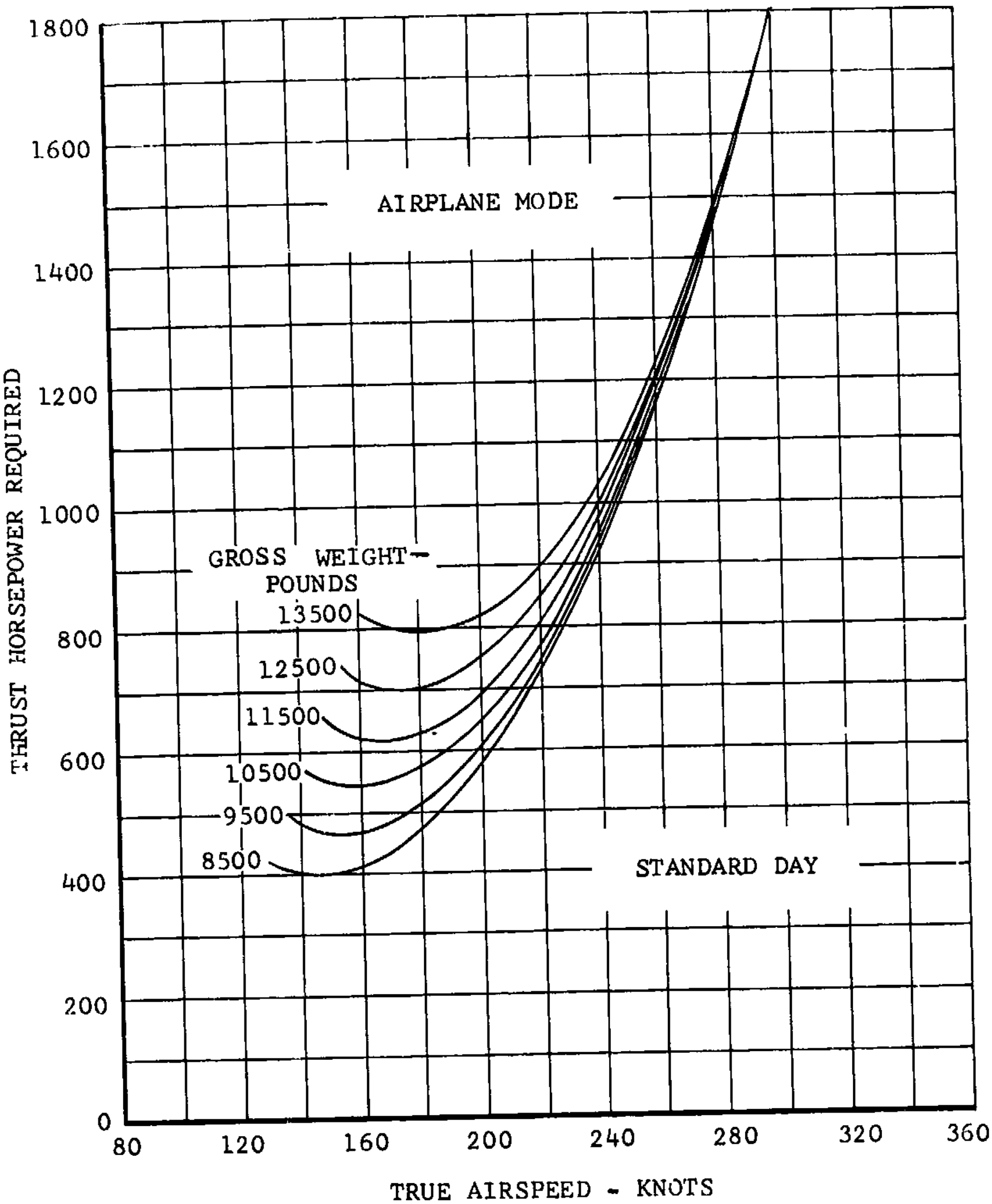

Figure V-26. Thrust Horsepower Required Versus True Airspeed, Sea Level.

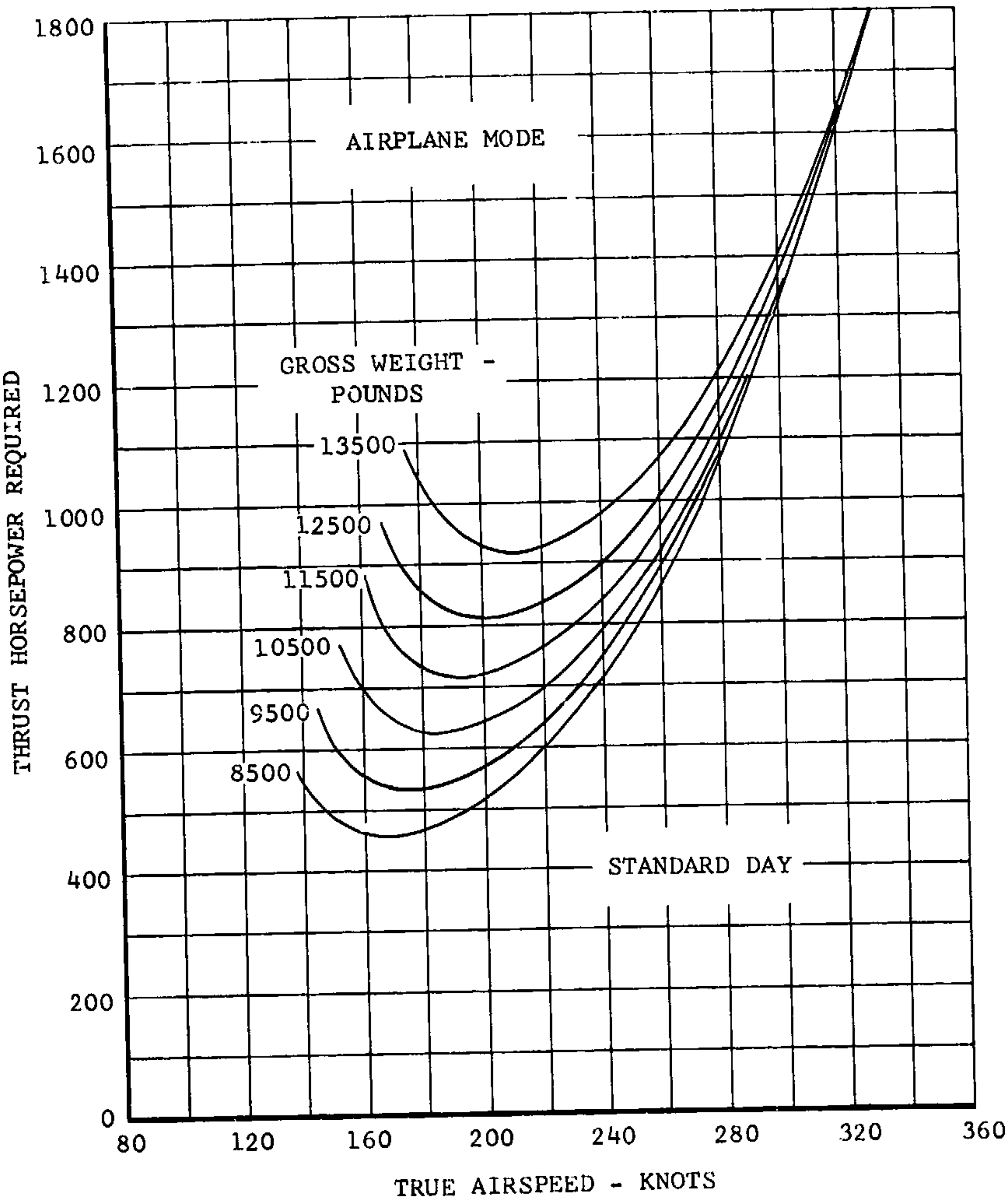

Figure V-27. Thrust Horsepower Required Versus True Airspeed, 10,000 Feet.

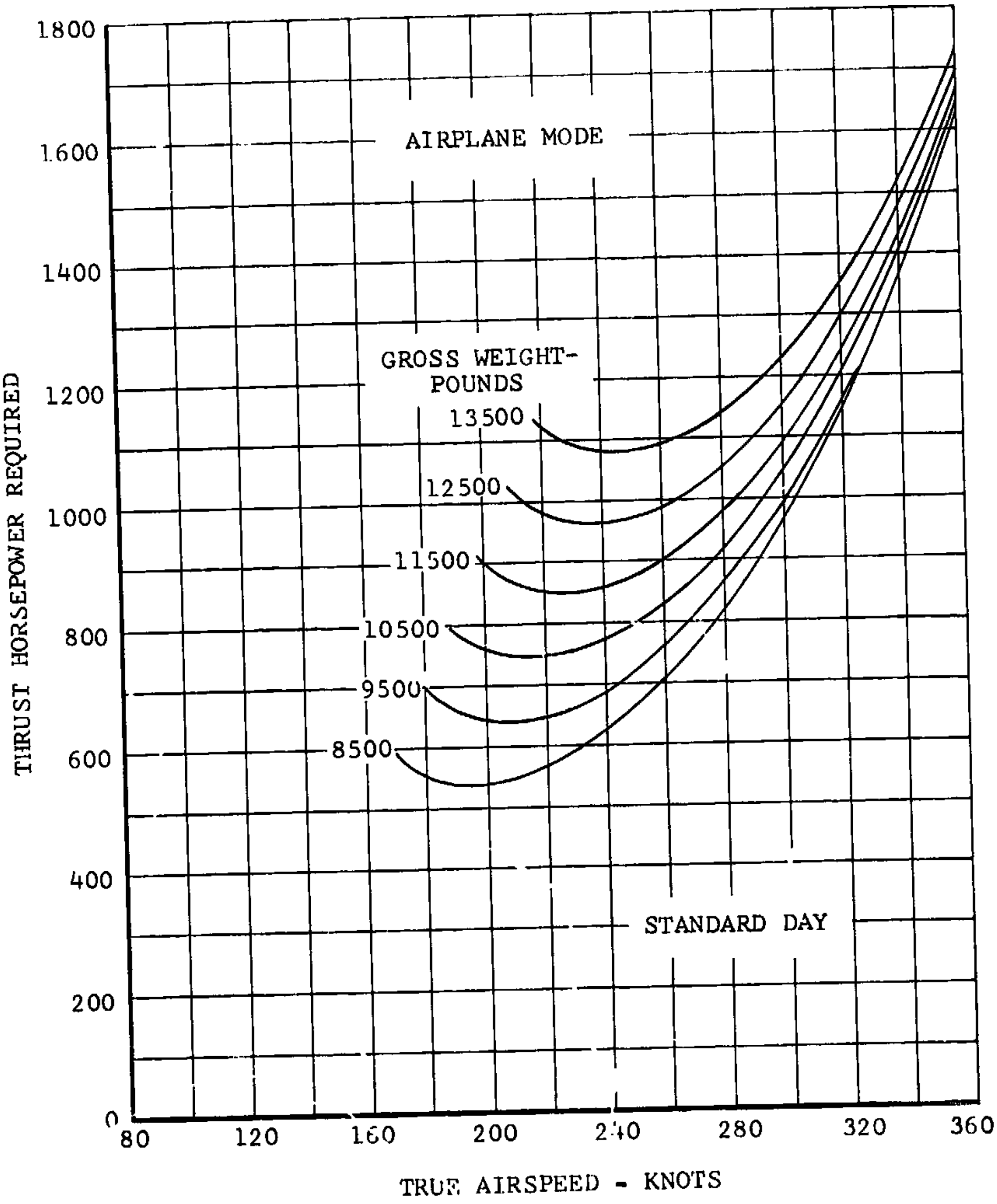

Figure V-28. Thrust Horsepower Required Versus True Airspeed, 20,000 Feet.

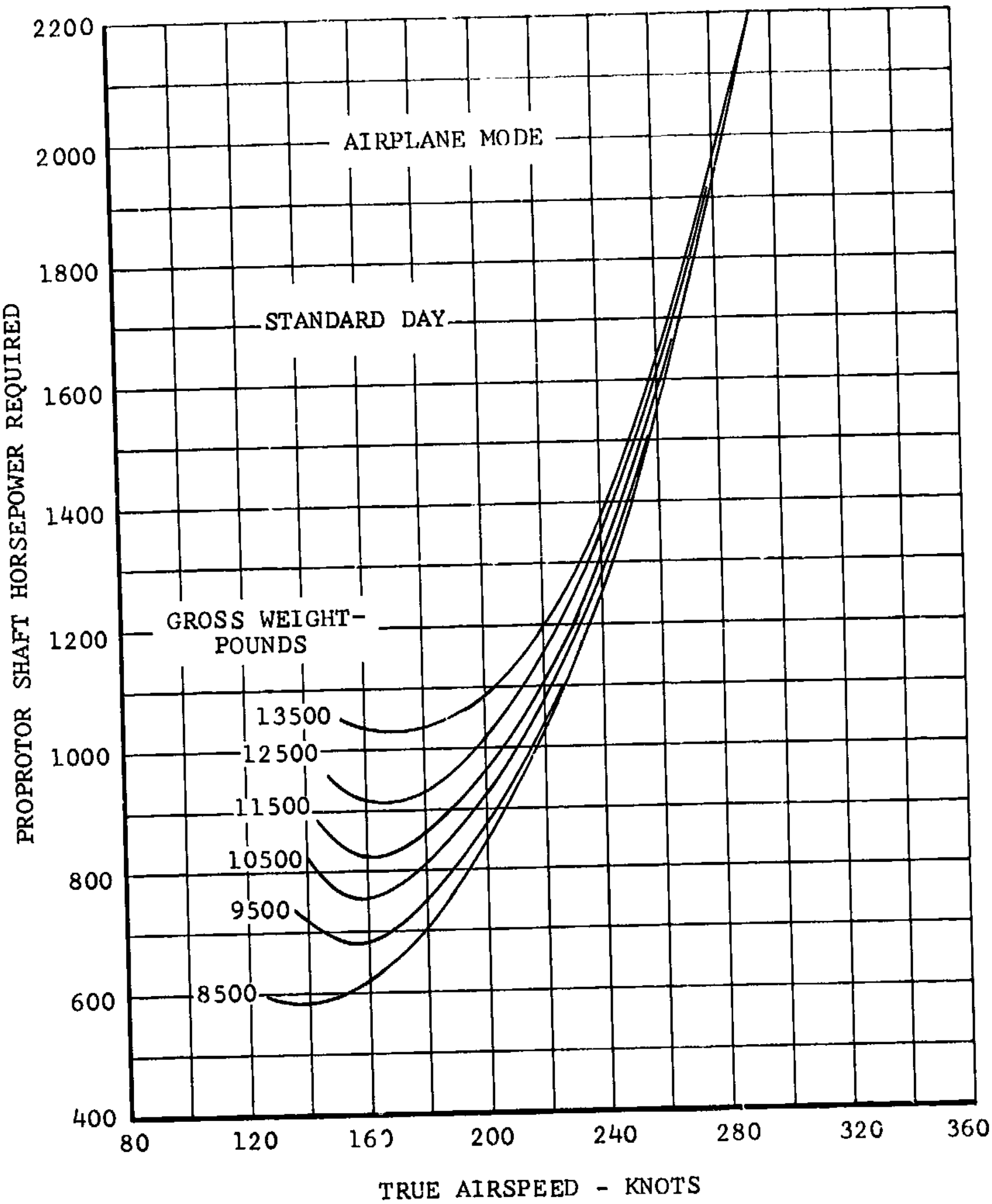

Figure V-29. Proprotor Shaft Horsepower Required Versus True Airspeed, Airplane Mode, Sea Level.

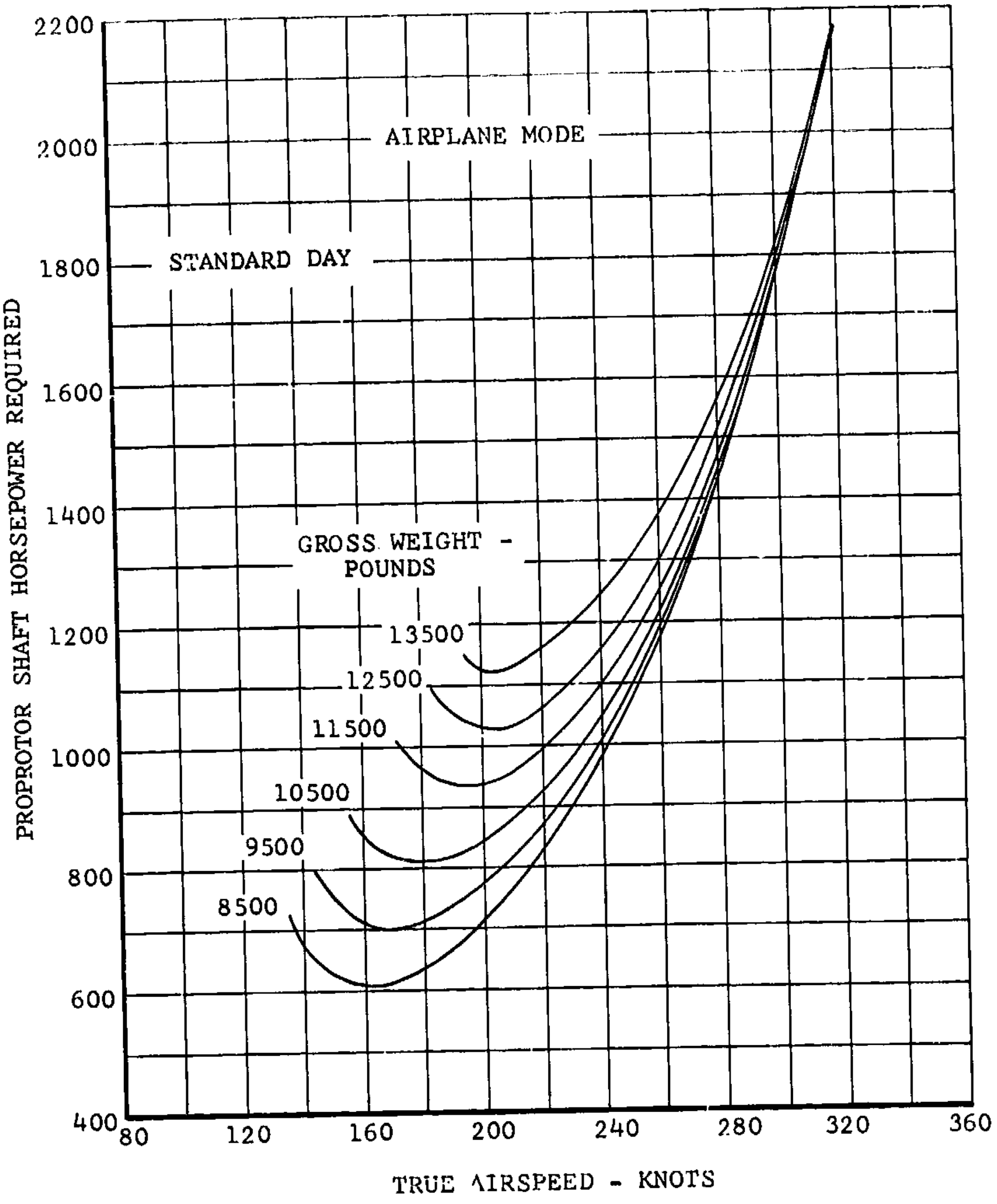

Figure V-30. Proprotor Shaft Horsepower Required Versus True Airspeed, Airplane Mode, 10,000 Feet.

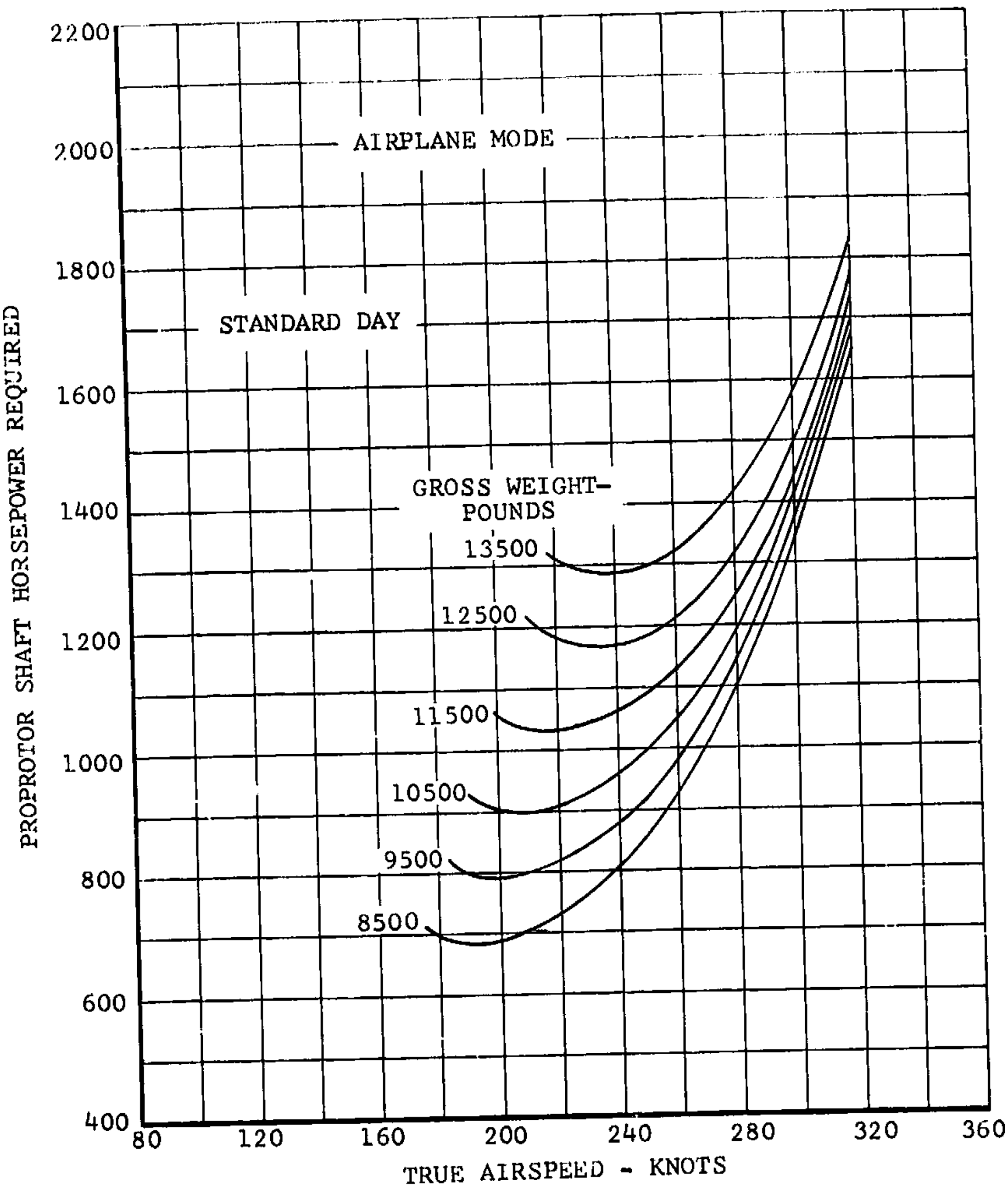

Figure V-31. Proprotor Shaft Horsepower Required Versus True Airspeed, Airplane Mode, 20,000 Feet.

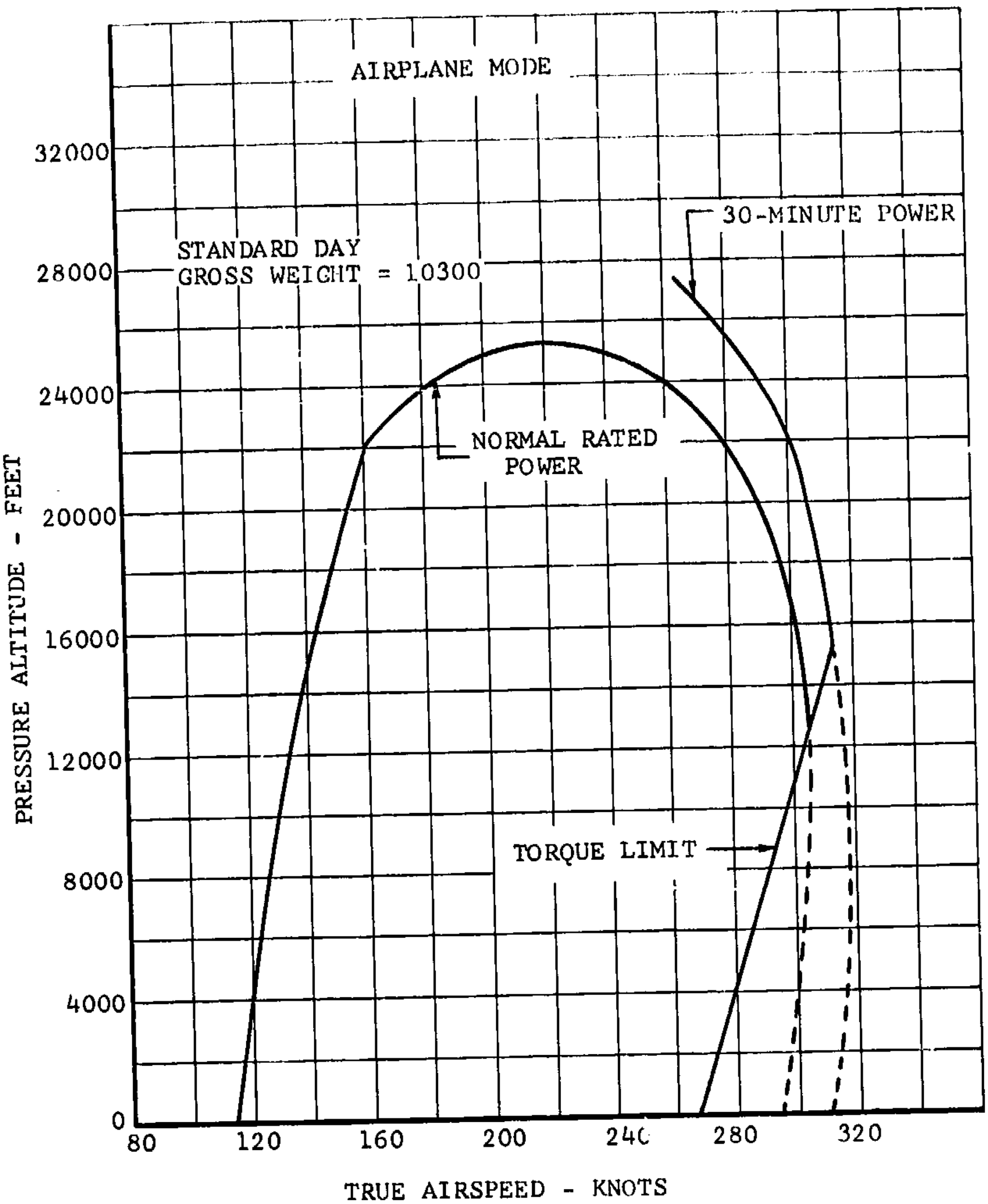

Figure V-32. Flight Envelope and Maximum Speed, Airplane Mode.

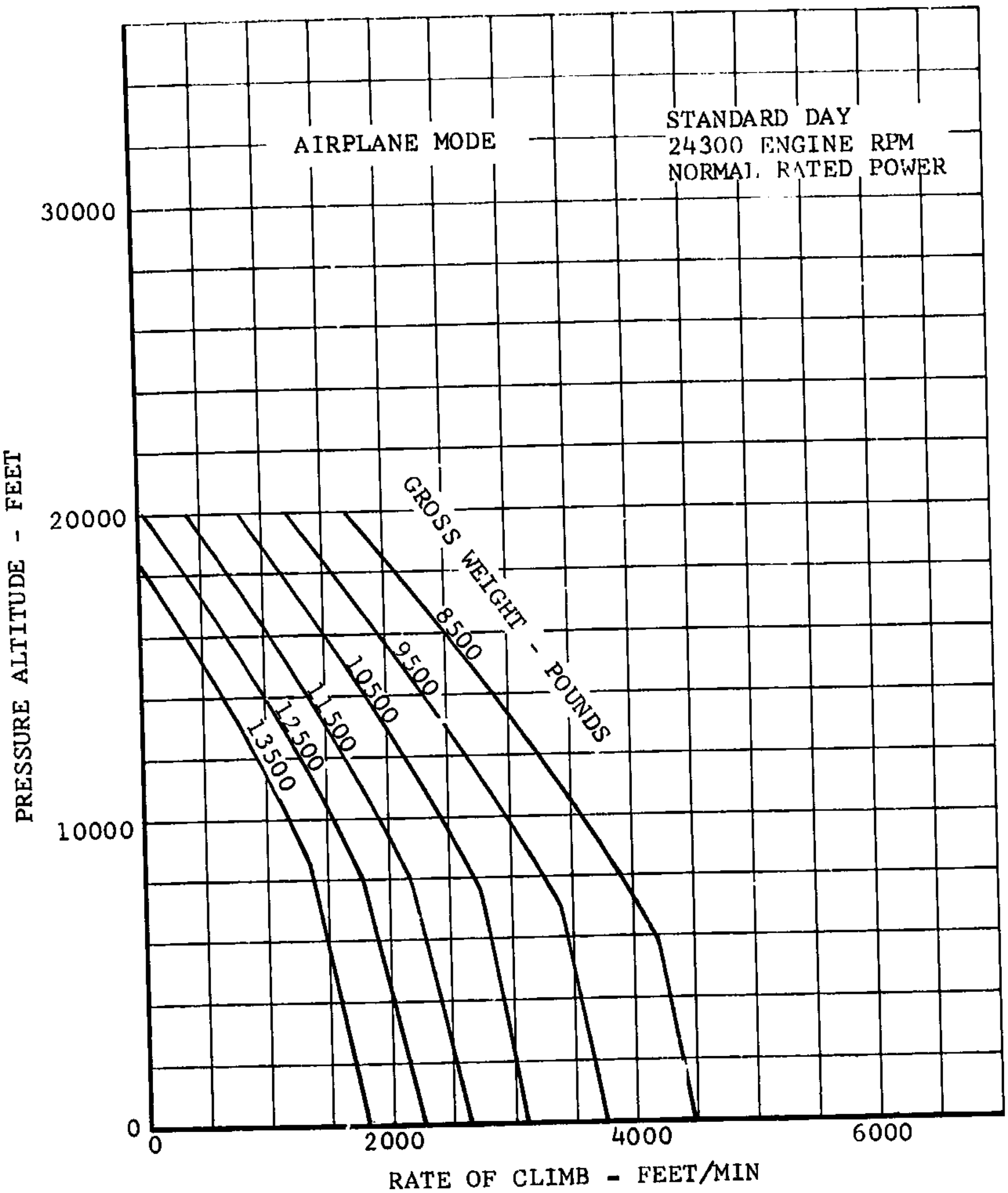

Figure V-33. Maximum Rate of Climb, Airplane Mode.

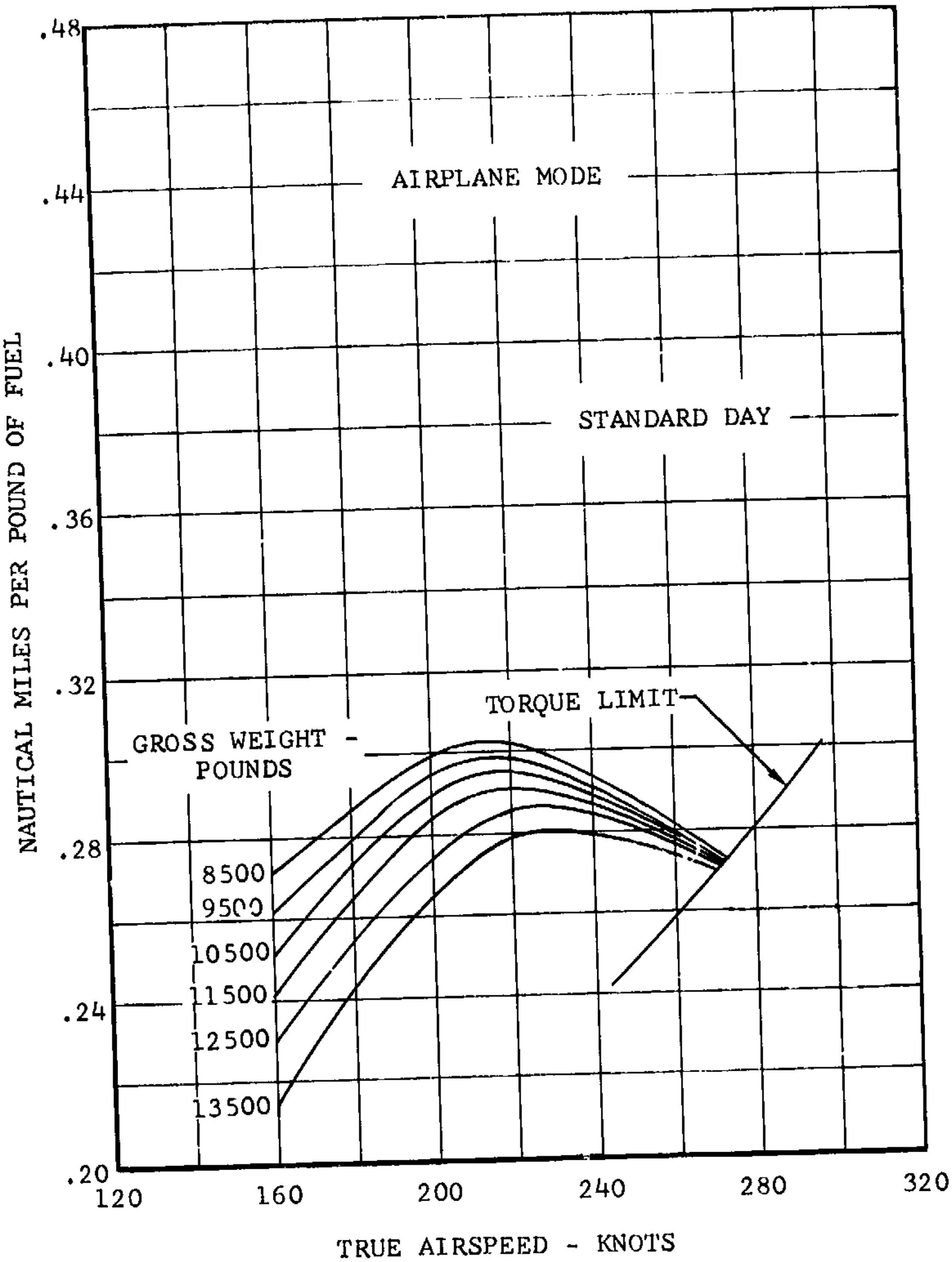

Figure V-34. Nautical Miles Per Pound of Fuel Versus True Airspeed, Airplane Mode, Sea Level.

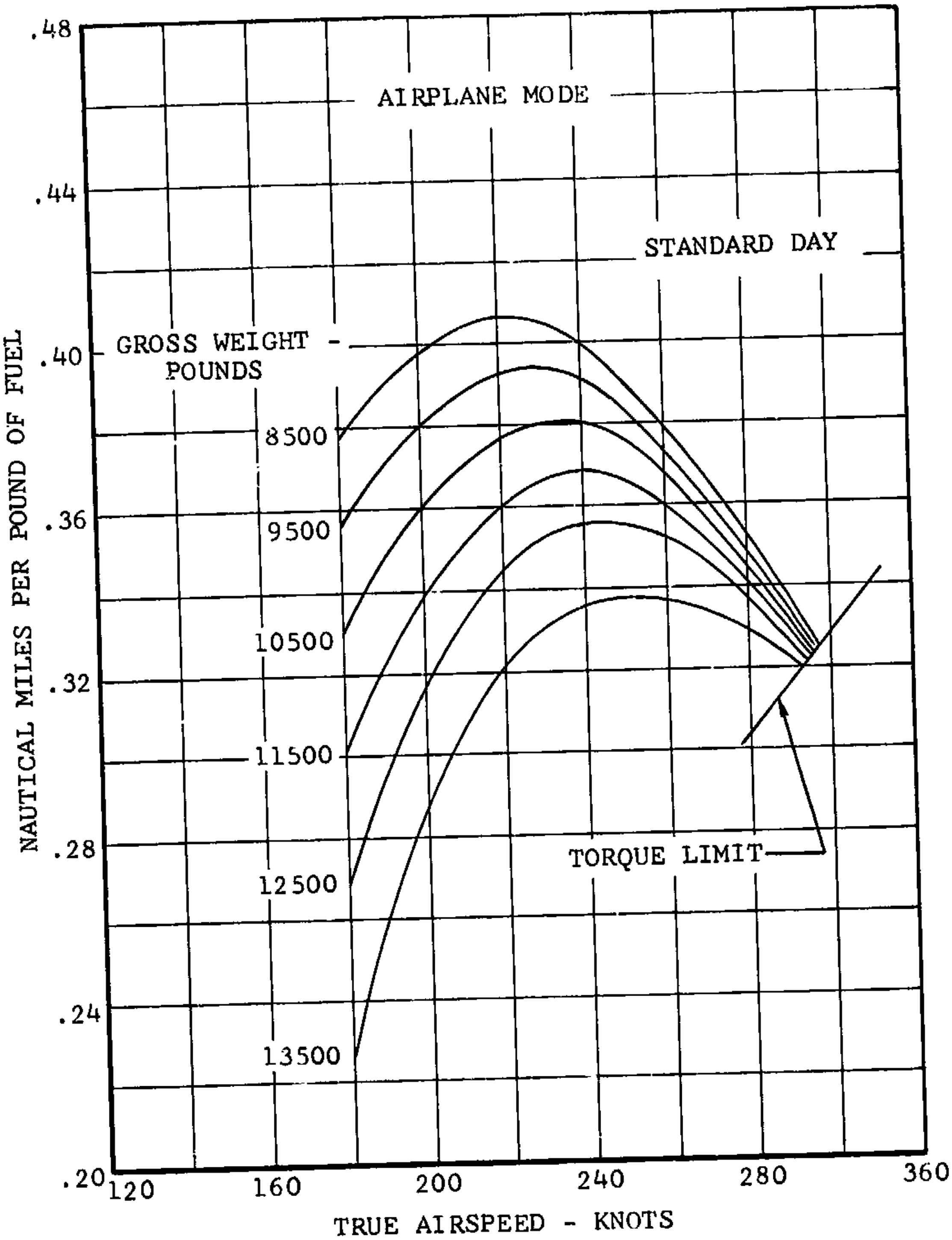

Figure V-35. Nautical Miles Per Pound of Fuel Versus True Airspeed, Airplane Mode, 10,000 Feet.

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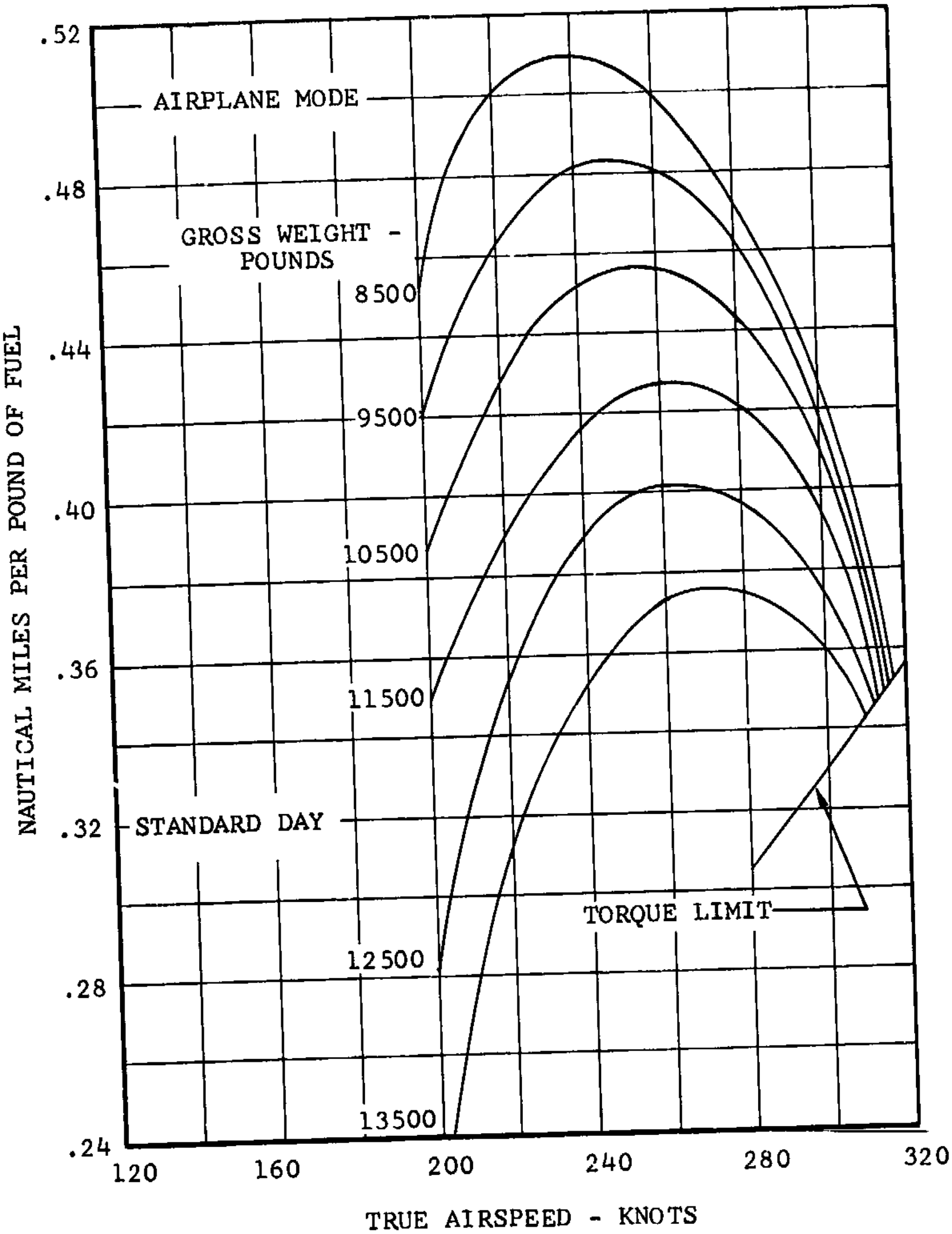


Figure V-36. Nautical Miles Per Pound of Fuel Versus True Airspeed, Airplane Mode, 20,000 Feet.

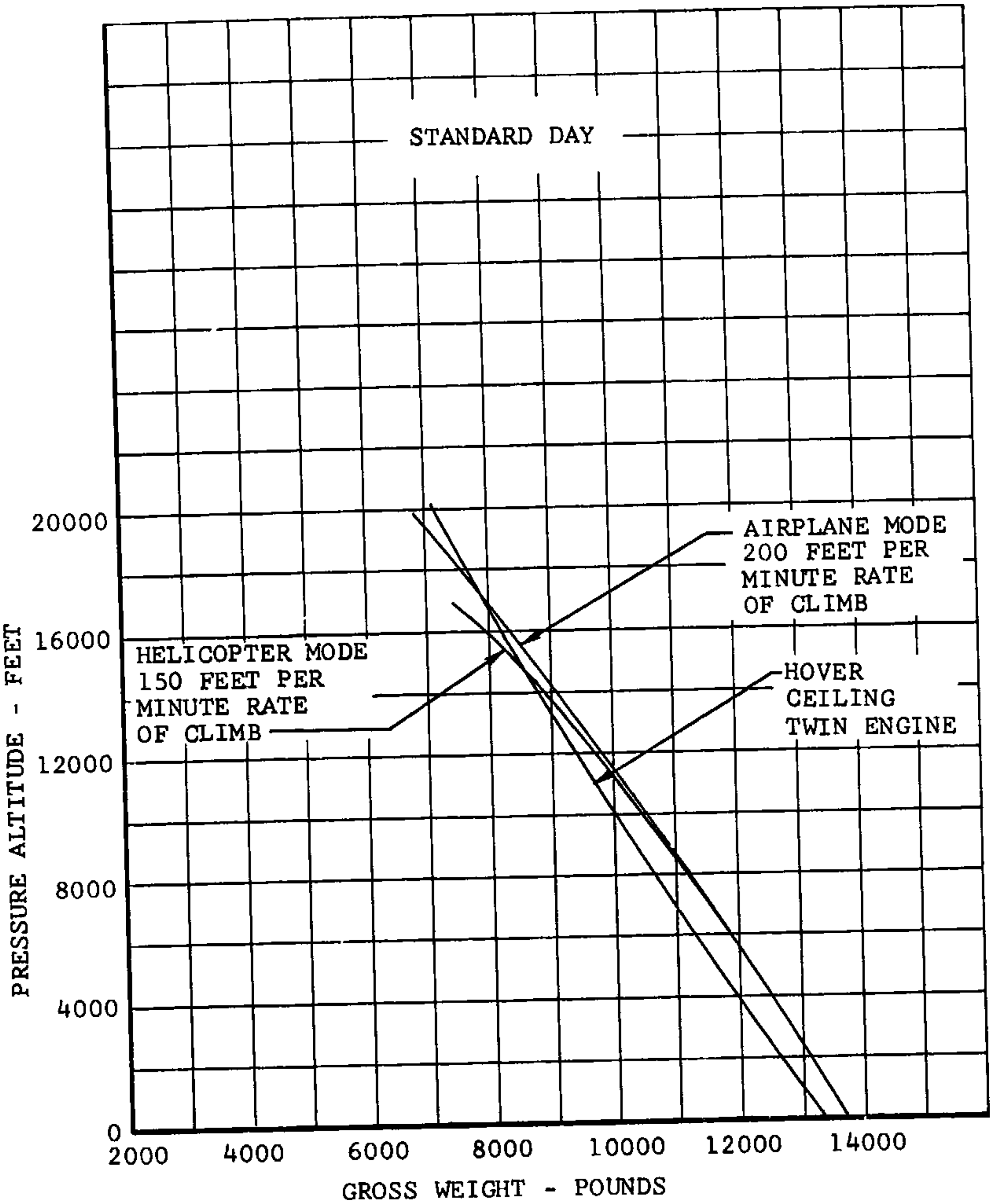
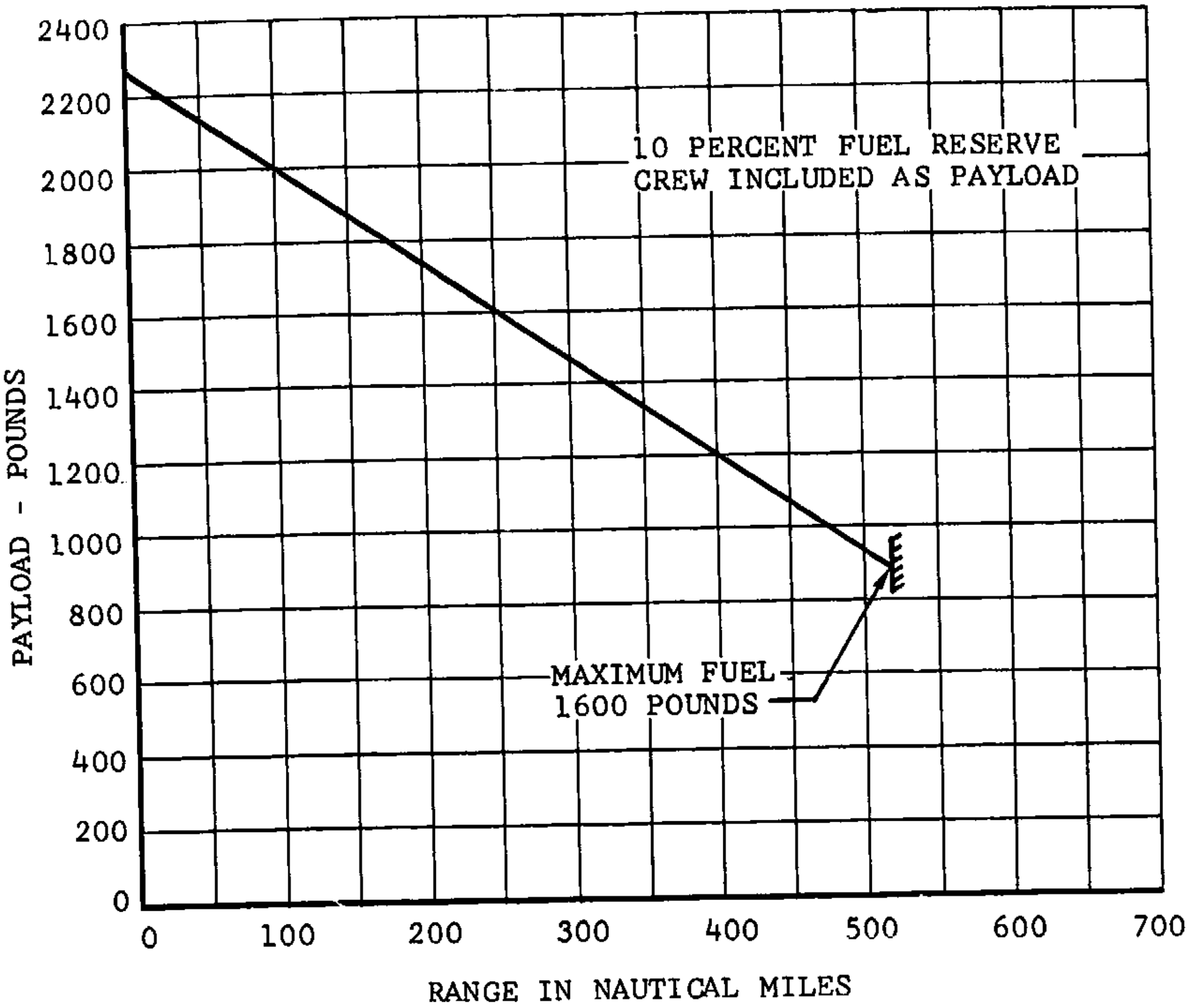


Figure V-37. Single Engine Performance.



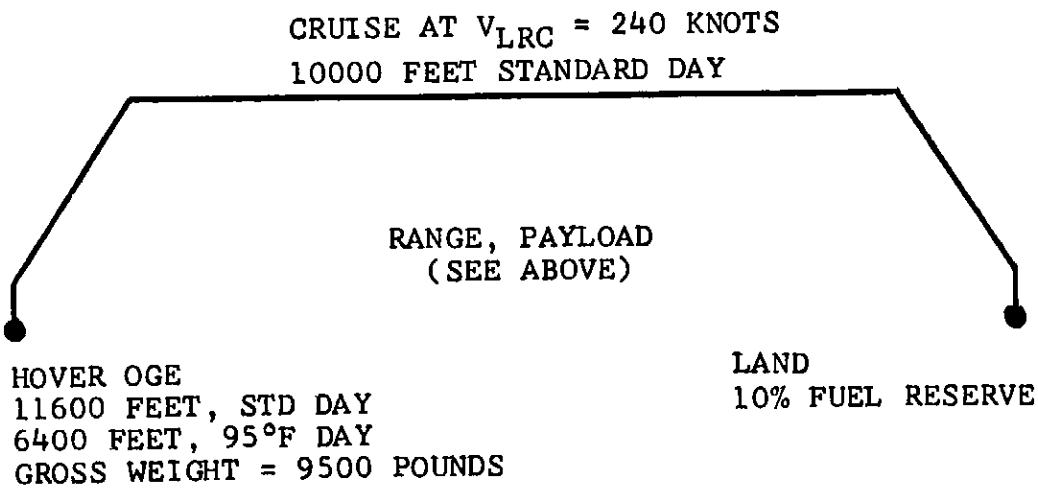


Figure V-38. Research Mission.

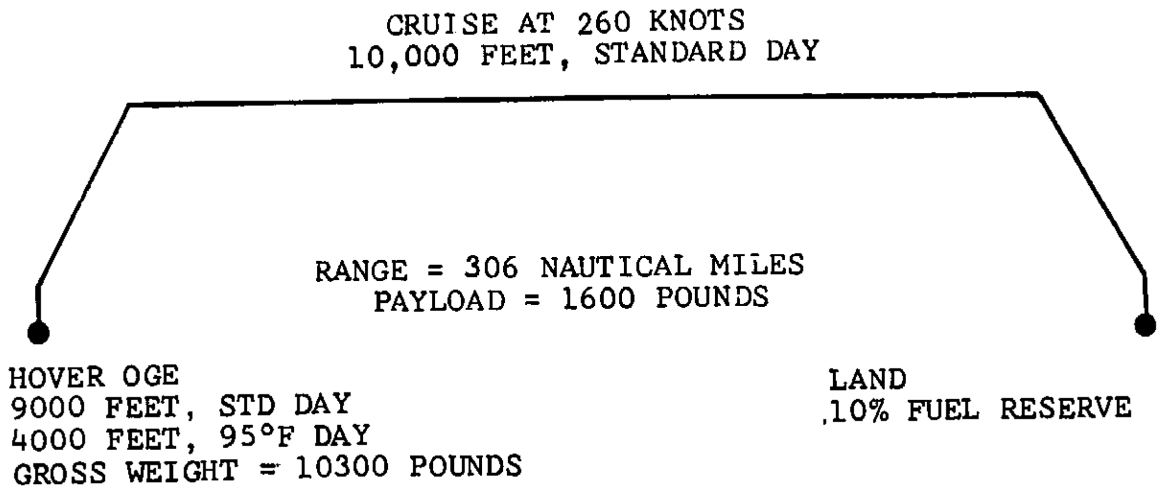


Figure V-39. Civil Mission.

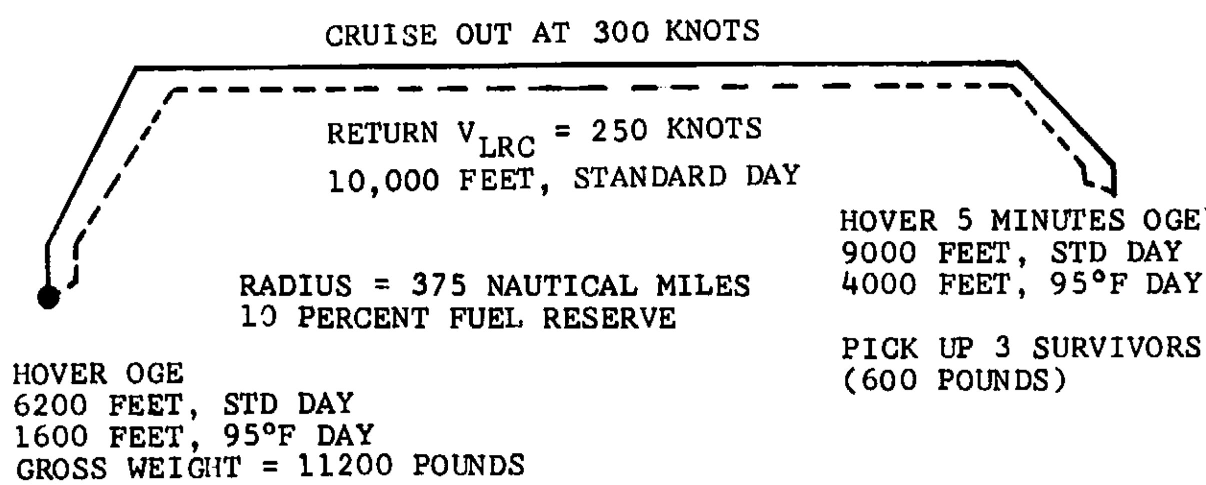


Figure V-40. Military Mission.

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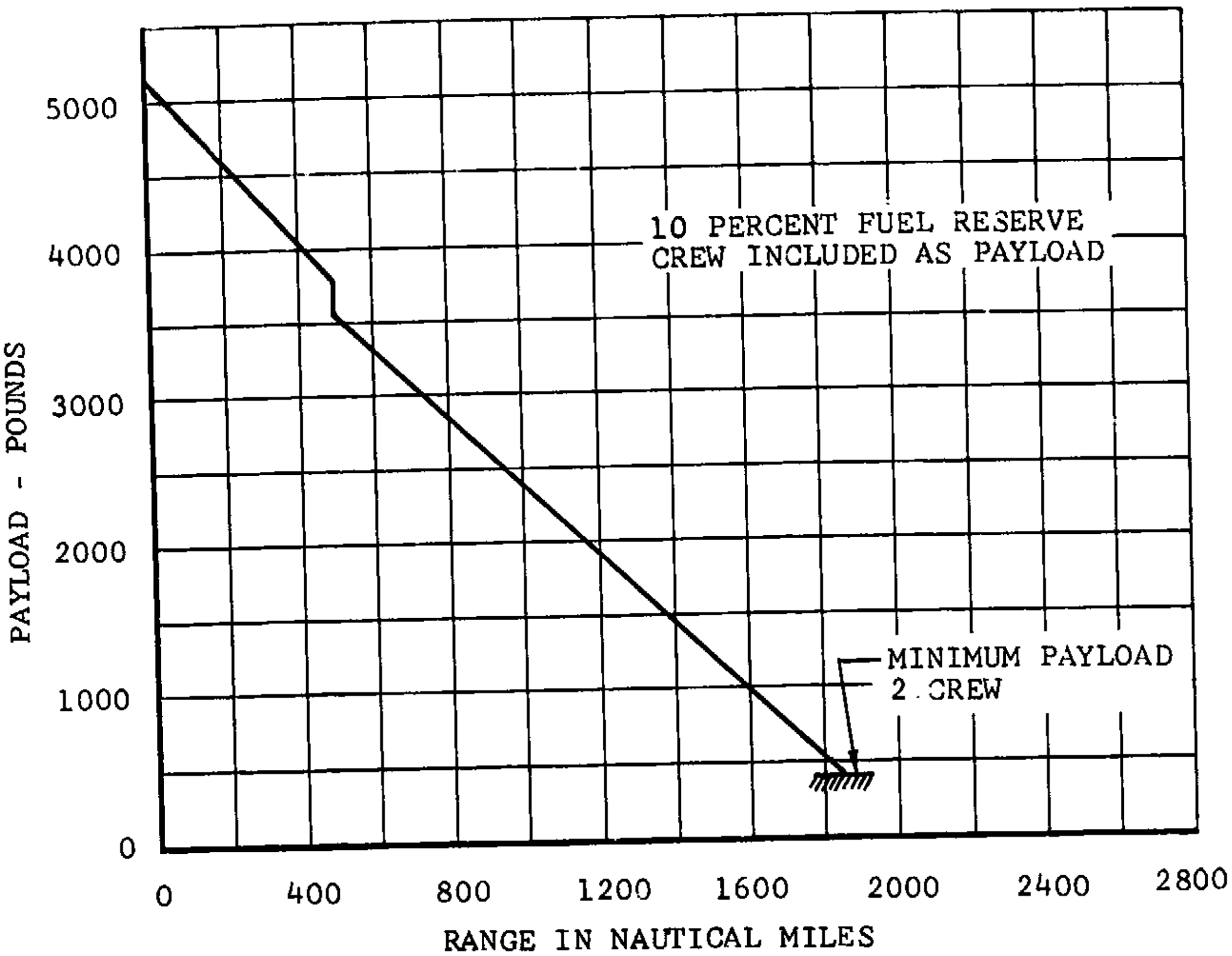

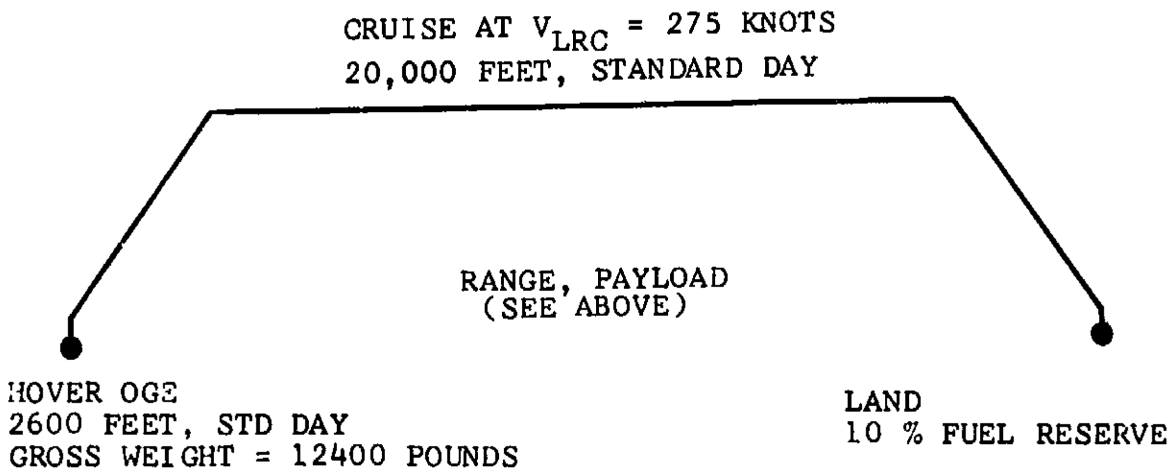

Figure V-41. Ferry Mission.

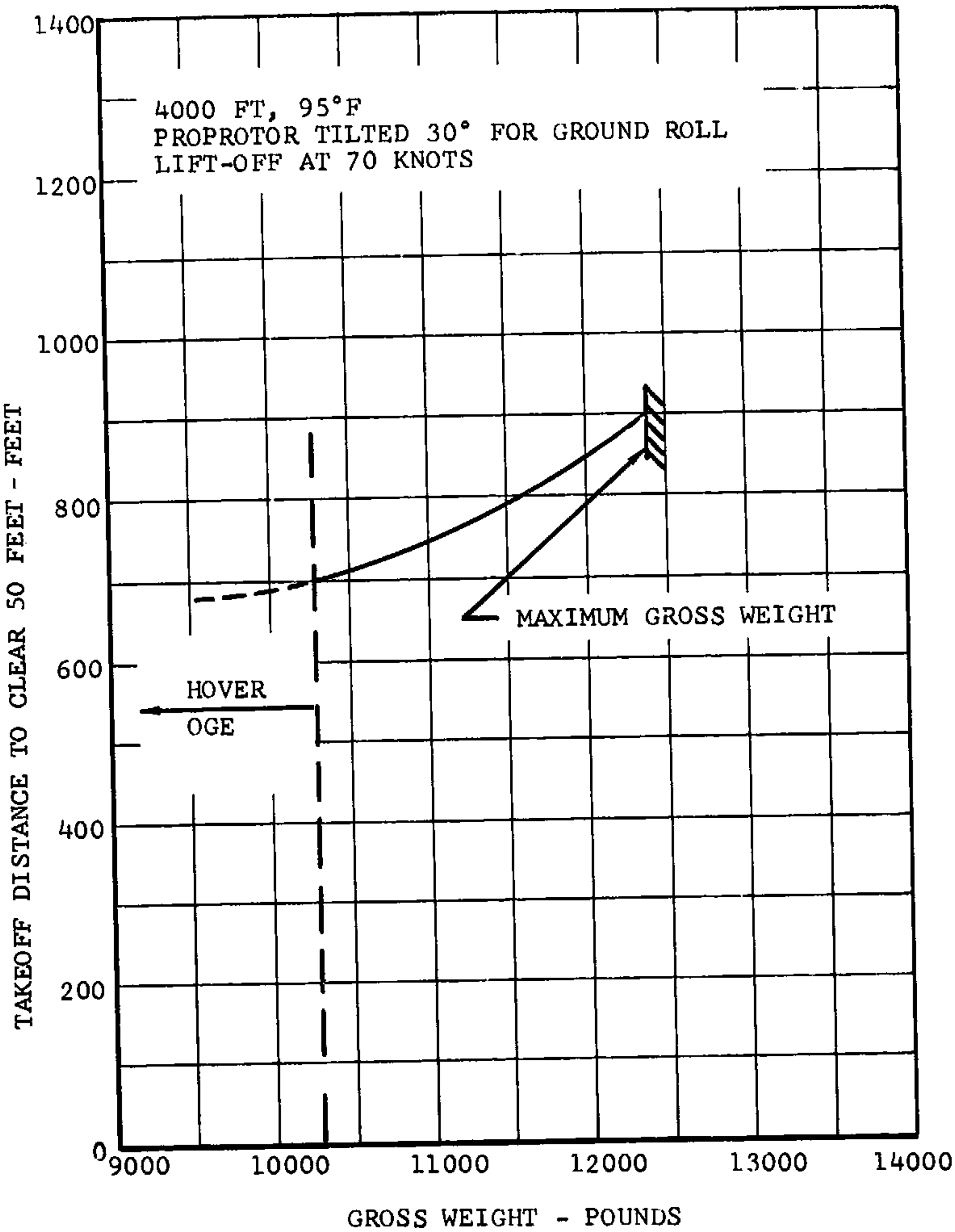

Figure V-42. STOL Takeoff Performance.

VI. DYNAMIC STUDIES

Dynamic characteristics have been carefully considered in the design of the Model 300. Particular attention has been paid to the proprotor and flight mode stability in airplane mode. The structural dynamics analysis has also included investigation of airframe vibration and proprotor dynamic loads. Proven analytical methods and computer programs were used for the study. All have been correlated with either flight-test or wind-tunnel test results to verify their accuracy.

A. Proprotor Dynamics

1. Natural Frequencies

The proprotor blade natural frequencies were calculated using BHC Computer Program C02. C02 is a Myklestad-type analysis for a rotating, twisted beam. It includes the coupling between beamwise and chordwise deflections resulting from built-in twist and collective pitch. Good correlation has been achieved with the measured frequencies of many rotor designs, including three-bladed semi-rigid rotors, such as those of the Model 300. The blade mass and stiffness distributions shown in Figure III-2 were obtained from the detail drawings released for the fabrication of the 25-foot proprotor. Particular care was taken in representing the stiffness of yoke and spindle regions as they were essential in determining the frequency of the fundamental modes.

Figures VI-1 and VI-2 show the calculated natural frequencies as a function of rpm and collective pitch. The frequencies are presented in terms of collective and cyclic modes. The collective modes are those excited by airloads whose frequency per revolution is an integer multiple of the number of blades (i.e., 3, 6, and 9 per rev). Nonmultiple harmonic airloads excite the cyclic modes (i.e., 1, 2, 4, and 5 per rev). The coupled natural frequencies shown in Figures VI-1 and VI-2 are noted by crosses (X) and diamonds (◊). The crosses denote modes whose largest deflection is normal to the plane of rotation; the diamonds are those whose largest deflection is in the plane of rotation. The solid lines denote the frequency of an untwisted blade at zero collective pitch (i.e., uncoupled beamwise and chordwise frequencies) and are provided for reference. By comparing the uncoupled and coupled frequencies, the effect of built-in twist on the blade frequencies is apparent.

Past experience with the design and testing of three-bladed semi-rigid rotors has shown that frequency placement of the first inplane (cyclic) and second beamwise (collective) modes poses the most severe requirements. The first inplane mode must be sufficiently removed from 1 per rev to avoid high 1-per-rev loads but cannot be too high or 2-per-rev loads will be a problem; a frequency of 1.5 per rev results in the lowest oscillatory loads. However, frequencies as low as 1.25 per rev are acceptable. Meeting this requirement can be difficult in proprotors because of the wide rpm and collective range required for efficient operation. The second beam mode must be located above 3 per rev to avoid high loads and airframe vibration. In helicopters, keeping this mode out of resonance has been a problem; however, it is less of a problem with proprotors because of the thicker and stiffer blade root sections required for static strength.

The frequency location of the major blade modes is as follows: The first inplane mode varies from 1.42 per rev to 1.27 per rev in helicopter and conversion modes (565 rpm), and from 1.44 per rev to 1.3 per rev in airplane mode (458 rpm). The second beam mode is 3.7 per rev at high collective pitch in helicopter mode and 3.92 per rev in airplane mode. Close proximity to 4-per-rev resonance is indicated for the second cyclic mode at high pitch in airplane mode (458 rpm) and 6-per-rev resonance for the third collective mode at high pitch in helicopter mode (565 rpm). Very low airload excitation in these resonant harmonics is anticipated. However, should these resonances pose a problem in wind-tunnel or flight testing, tuning weights can be used to raise or lower the mode's frequency, as required.

The first torsional natural frequency is located at 4.5 per rev. This mode is rigid-body blade pitching on the control system stiffness. The second torsional frequency, which involves blade torsional deformation, is much higher.

2. Loads

Past Bell studies of proprotor loads have shown that two flight conditions impose the most severe blade loads. For oscillatory loads, the maximum level flight airspeed in helicopter mode is the most severe; this is also true for conventional helicopters. For design limit loads, the maximum results from a gust encounter in airplane mode. Several other flight conditions, including maneuvers in all modes were examined. In all cases the blade loads were less severe than those of the two abovementioned flight conditions.

Blade loads were calculated using Bell's "Rotorcraft Maneuver Program," Computer Program C-81. In C-81, a trim condition of the aircraft is established by balancing the forces and moments acting at the aircraft's cg. Rotor collective and cyclic pitch and the aircraft fixed controls are adjusted to obtain the trim condition. Once trim is achieved the blade air loads are harmonically analyzed and used, with the blade frequency response, to calculate the bending moments. They are presented in harmonic format. If desired, a maneuver may be entered and loads calculated at specified time intervals during the maneuver. At any time during the maneuver a discrete gust encounter may be simulated with the gust's shape and magnitude specified as required.

Figure VI-3 shows the calculated blade oscillatory stress for V_{max} in helicopter mode, sea level, 140 knots TAS, 9500 pounds gross weight, 30-degree flaps, and 0-degree conversion angle. The peak stress of 22,500 psi occurs at 60 percent radius; the allowable design endurance stress for the 17-7PH, condition TH1050, steel used for the Model 300 blades is 30,000 psi. The principal frequency of the loads is 1 per rev with a small amount of 2 per rev and 3 per rev. The peak stress in the titanium yoke/spindle is calculated to be 7400 psi; the endurance limit is estimated to be 15,000 psi.

The limit blade stress resulting from a 50 foot-per-second vertical gust encounter at V_H, in airplane mode, is shown in Figure VI-4. The limit stresses are obtained by applying the 80-percent alleviation factor specified for the aircraft gust load factor determination. The (1- cosine) gust shape was also examined and produced the same magnitude of loads as the sudden gust with an 80-percent alleviation factor. The compressive buckling stress is also shown in Figure VI-4 as blade buckling is more marginal than a tensile failure. By conservatively assuming that the blade buckling strength is equivalent to the initial buckling stress, it is seen from Figure VI-4 that the blade can carry an ultimate load 1.5 times the limit loads resulting from the 260-knot gust encounter. This condition is also a critical one for the hub. The critical stress occurs in the titanium hub yoke, at the junction of the spindle and yoke ring. The calculated limit stress for this region is 89,000 psi. With an allowable ultimate bending stress of 130,000 psi, the margin of safety is slightly negative.

Control loads were also calculated. Several methods were used to determine loads in the helicopter mode including an empirical method based on measured loads for a number of helicopter rotors. The results are tabulated below. The design pitch-link load was conservatively established as 345 ± 345 pounds.

TABLE VI-1

MODEL 300 PITCH LINK LOADS

Condition/Method	Steady	Oscillatory
V_{max} - helicopter mode		
Calculated	180 lb	± 130 lb
Empirical	-	± 175 lb
Model test	-	± 120 lb
V_{max} - airplane mode		
Calculated - power on	163 lb	-
- power off	332 lb	-

3. Blade Flapping

Proprotor flapping in airplane mode is summarized in Figure VI-5. Flapping was calculated using several methods. For steady-state maneuvers, the mast angle of attack and pitch rate were multiplied times the respective flapping derivatives ($\partial\beta/\partial\alpha$ and $\partial\beta/\partial q$). The gust response was calculated with Computer Program C-81.

It should be pointed out that the Model 300's flapping sensitivity to angle of attack in airplane mode is only about 40 percent that of the XV-3. This is due to the higher tip speed (600 fps versus 356) in airplane mode. The rotor following time is also less since the Model 300 blade lock number is 3.76 compared to 2.1 for the XV-3.

Flapping in helicopter and conversion modes is always less than the 12 degrees allowed by the mechanical flapping stops.

B. Airframe Dynamics

1. Natural Frequencies

Placement of the wing-pylon-fuselage natural frequencies has been guided by two considerations: first, resonance of the coupled airframe modes with proprotor 1, 3, and 6 per rev must be avoided to have satisfactory vibration characteristics; second, the frequencies must be adequately separated to avoid aeroelastic instability. For the Model 300 the wing beam and chord and the fuselage bending stiffness resulting from strength requirements provide for satisfactory location of these fundamental modes. However, the wing torsional stiffness for strength requirement resulted in a wing torsion resonance at airplane mode rpm. The wing torsional stiffness was increased by 60 percent to provide adequate separation from 1-per-rev resonance. The resulting frequency locations provide good vibration isolation and as a result of the torsionally stiff wing, a high level of proprotor-pylon stability.

The symmetric and asymmetric airframe natural frequencies are shown in Figures VI-6 and VI-7 as a function of pylon conversion angle. The range of frequency of the symmetric modes with gross weight is indicated by shadings; the higher frequency corresponding to minimum operating weight and the lower to the 12,400-pound maximum gross weight. The asymmetric modes do not vary significantly with gross weight. The frequencies shown are for the 9500-pound design gross weight.

Two resonance conditions are passed through during conversion. The asymmetric wing chord mode is in 1-per-rev resonance at partial conversion angles. This mode has relatively low fuselage response to hub shears. Also, the second wing beam modes (symmetric and asymmetric) are in resonance at 3 per rev for certain fuel loadings and conversion angles. These modes have nodes near the rotor hub and consequently low response to hub shears. They are also damped by fuel sloshing under partial fuel loading conditions and by aerodynamic damping. The pylon yaw natural frequency is above 4 per rev.

A preliminary frequency analysis has also been made for the empennage. The cantilever frequencies for the fin and tail plane are tabulated below: (The tail plane mass and inertia in the fin calculation.)

TABLE VI-2

EMPENNAGE NATURAL FREQUENCIES

Mode	Frequency
1st fin bending	8.0 cps
1st fin torsion	27.0 cps
1st tail plane bending	13.0 cps
1st tail plane torsion	68.7 cps

The airframe natural frequencies were calculated using two analyses. Program A75D, a state-vector, crossed beam analysis was used to determine the coupled fuselage-wing-pylon frequencies. A75D includes coupling between beam and torsion deflections, and shear deformation and rotary inertia. It has been correlated with shake tests of the XV-3 convertiplane and several helicopters. The empennage frequencies were calculated with Program DF1789, which uses the finite element method. The mass and stiffness distributions used in the airframe vibration analysis are shown in Figures III-3 through III-6.

2. Vibration Levels

Estimated 3-per-rev vibration levels for helicopter, conversion, and airplane modes are shown for the pylon and crew station in Figure VI-8. For the helicopter and conversion modes the vibration level was derived by using calculated hub shears from the blade load calculations and the airframe frequency response. The calculated 3-per-rev vertical response at the pylon and crew station and airplane modes is shown in Figures VI-9 and VI-10 respectively.

The airplane mode vibration level was estimated by using pylon 3-per-rev vibrations measured in the August wind-tunnel test of the Model 300 aeroelastic model. The pylon vibration is scaled directly from the measured data (the model is Froude scaled). The crew station level was determined by multiplying the ratio of the response level at the crew station to that of the pylon, times the measured pylon acceleration.

C. Stability

1. Proprotor

The high wing torsional stiffness, resulting from the requirement for avoiding wing torsional resonance, provides a high level of proprotor/pylon stability for the Model 300. Other factors are the moderate value of pitch-flap coupling and the hub restraint. A summary of the proprotor stability is given in Figure VI-11.

At sea level the margin is over 170 percent of the 350-knot dive airspeed, V_D, far in excess of the 120 percent V_D flutter-free requirement. For reference the airspeed above which the blade tip helical Mach number exceeds 0.9 is shown. In the following paragraphs the analytical methods used to determine the Model 300 proprotor stability boundary and the sensitivity to rotor rpm and wing stiffness are discussed.

Proprotor/pylon stability characteristics were determined with BHC Computer Program DYN4. DYN4 is a linear, twenty-one degree-of-freedom proprotor stability analysis. It is capable of determining the proprotor/pylon, blade motion, and flight mode stability characteristics of tilt-proprotor aircraft. A tip-path-plane representation is used for the proprotor dynamics and linear aerodynamic functions (C_L, C_D) are assumed.

Ribners' method (Reference 34) for propellers is used to correct for compressibility effects. Rotor details such as pitch-axis preconing, underslinging, pitch-flap coupling and flapping restraint are included. The first inplane blade mode is represented and control system flexibility may be simulated quasi-statically. Five coupled wing/pylon elastic modes are represented; wing beam, chord, torsion, pylon pitch and yaw. The six fuselage rigid body degrees of freedom are included which allows the symmetric and asymmetric free-free modes and the aircraft flight modes to be simulated. Input to DYN4 is in terms of lumped parameters describing the dimensions, inertias, stiffnesses, and kinematics of the aircraft being simulated. Standard aircraft stability derivatives are input to represent the airframe aerodynamics. Output of DYN4 is in terms of the system eigenvalues and eigenvectors. DYN4 has been extensively correlated with small-scale model test stability data. The correlation is good, even for complex aeromechanical types of instability. An example of the degree of correlation is shown in Figures VI-12 and VI-13. The measured data in Figure VI-12 are from a joint NASA-Bell proprotor test of the Model 266 aeroelastic model, that of Figure VI-13 are from the August tests of the Model 300 aeroelastic model. The Model 300 model is shown in Figure VI-14 in the semispan configuration tested in August. Figure VI-15 shows the full-span model as it will be mounted in the NASA Langley 16-foot transonic dynamics wind tunnel next spring.

Two other programs are used to investigate areas where DYN4 lacks capability. These are Programs DYN3 and DYN5. Program DYN3 combines the proprotor representation of DYN4 with a wing flutter analysis. This permits the influence of the proprotor on wing flutter characteristics to be determined, as well as to investigate the effect of wing aerodynamics on the proprotor/pylon stability. Bell studies of these effects have shown only small mutual effects of flutter on proprotor/pylon stability and vice versa. DYN5 is a nonlinear open-form proprotor aeroelastic analysis. It uses strip theory and a tabular representation of the blades aerodynamic functions (C_L, C_D, and C_m) to account for stall and compressibility. Tables for different profiles may be used to account for variation in the blade section from root to tip. Correlation of DYN4 and DYN5 is shown in Figure VI-16, and indicates only a small difference in the stability boundary. DYN5 does indicate a slightly lower level of damping than DYN4.

Figures VI-17 and VI-18 show the root loci of the Model 300 symmetric and asymmetric modes as a function of airspeed. Note that the symmetric wing chord mode becomes unstable first. In calculating these root loci the airframe structural damping was assumed to be zero. This is conservative since 1 to 1-1/2 percent of critical is inherent in the wing structure. The blade lift curve slope was corrected for compressibility for airspeeds up to 350 knots and the 350-knot value used at the higher speeds. Thus, the stability boundary shown in Figure VI-11 is representative of the stability margin at the 350 knot dive speed.

The Model 300 proprotor stability is not sensitive to rpm as evident in Figure VI-19. Also, a very large loss of wing structural stiffness or pylon attachment stiffness can occur before instability would occur inside the flight envelope. This is also shown in Figure VI-19. The principal dynamic problem that would result from the rotor rpm exceeding the ±10 percent rpm limit or from loss of structural stiffness would be a 1-per-rev resonance which could be avoided by changing rotor rpm.

Blade motion stability for the Model 300 proprotor is assured by the selection of rotor parameters that provide stable characteristics. The first inplane frequency is above operating speed which eliminates mechanical instability (ground resonance). The blade is mass balanced such that pitch-flap flutter or weaving will not occur. The blade effective center of gravity is at 24-percent chord with the effective aerodynamic center in hover being at 26.1 percent. The 2-1/2 degrees of pitch axis preconing and a stiff control system are used to prevent pitch-lag instability. Positive pitch-flap coupling of 0.268 ($\delta_3 = -15°$) is used to prevent flap-lag instability in airplane mode.

2. Flight Mode Stability

Stability characteristics in airplane mode were determined over a speed range from 150 knots to 350 knots and from sea level to

20,000 feet. The predicted stability characteristics are acceptable in accordance with MIL-F-8785(ASG).

The fin and tailplane of the Model 300 have been sized conservatively to over compensate for the destabilizing influence of the proprotors.

The flight-mode stability characteristics were calculated using Computer Program DYN4. The equations of motion for the basic airframe (less rotors) are those suggested by Etkin in Reference 35. The proprotor influence on the flight-mode stability is accounted for directly by using DYN4. This permits the coupling effects from the proprotor-pylon-wing and flight modes to all be treated simultaneously.

The stability derivatives for the basic airframe were obtained from the wind-tunnel test of the Model 300 1/5th scale force and moment model. Where necessary, estimates were made using the methods suggested in NASA TN-1098. The derivatives were corrected for Mach number effects using the Prantle-Glauret rule.

a. Longitudinal Modes

The root loci of the short period mode, as a function of airspeed and altitude, is shown in Figure VI-20. That of the basic airframe, less rotors, is shown in Figure VI-21 for comparison.

Analysis of the phugoid mode was made using Etkins method as shown in Reference 35. The frequency and damping are given by

$$\omega_n = \frac{C_{L_O}}{\sqrt{2}\mu} \quad \text{and} \quad \zeta = \frac{1}{\sqrt{2}} \frac{C_{D_O}}{C_{L_O}}$$

The time to half, conservatively estimated by assuming the proprotor forces to be zero, is presented in tabular form below:

TABLE VI-3

PHUGOID MODE CHARACTERISTICS

GW = 9500 lb
CG location 294 (aft)
Rotor rpm = 458

Flaps up
NASA standard day
Mast angle = 90°

Speed (kt)	Altitude (ft)	Period (sec)	Time to Half (Sec)	Magn	u/θ Phase (deg)
150	Sea level	35.1	52.0	0.706	-94.2
250	Sea level	58.6	59.1	0.706	-96.2
350	Sea level	83.5	42.3	0.704	-102.2
250	20000	58.4	100.6	0.706	-96.2
350	20000	81.8	79.2	0.708	-96.5

b. Lateral-Directional Modes

The root loci of the Dutch roll mode, as a function of airspeed and altitude is shown in Figure VI-22. That of the basic airframe, less rotors, is shown in Figure VI-23 for comparison. Note that at low airspeeds the damping is higher than that of the basic airframe. This is due to rotor thrust damping. At higher speed the thrust damping is less than the negative damping contribution of the proprotor side force and the Dutch roll damping is therefore lower than that of the basic airframe.

Mode shape and phase data for the Dutch roll at selected airspeeds and altitudes are presented below:

TABLE VI-4

DUTCH ROLL MODE SHAPE AND PHASE

Speed (kt)	Altitude (ft)	Period (sec)	Time to Half (Sec)	ϕ/β Magn	ϕ/β Phase (deg)	ϕ/ψ Magn	ϕ/ψ Phase (deg)
150	Sea level	4.73	0.885	1.187	3.7	1.190	188.7
250	Sea level	2.53	0.741	1.475	9.0	1.449	198.2
350	Sea level	1.66	0.655	1.820	14.4	1.740	204.9
250	20000	3.54	1.740	1.778	4.2	1.720	189.5
350	20000	2.32	1.680	1.190	7.2	2.070	194.9

The times to half for the spiral and rolling modes are tabulated below. For these modes the proprotor provides an increase in stability.

TABLE VI-5

TIME TO HALF FOR ROLLING AND SPIRAL MODES

Speed	Altitude (ft)	Time to Half Amplitude: Spiral Mode (sec)	Time to Half Amplitude: Rolling Mode (sec)
150	Sea level	5.96	0.773
250	Sea level	11.40	0.473
350	Sea level	147.00	0.363
250	20000	6.90	0.885
350	20000	10.95	0.695

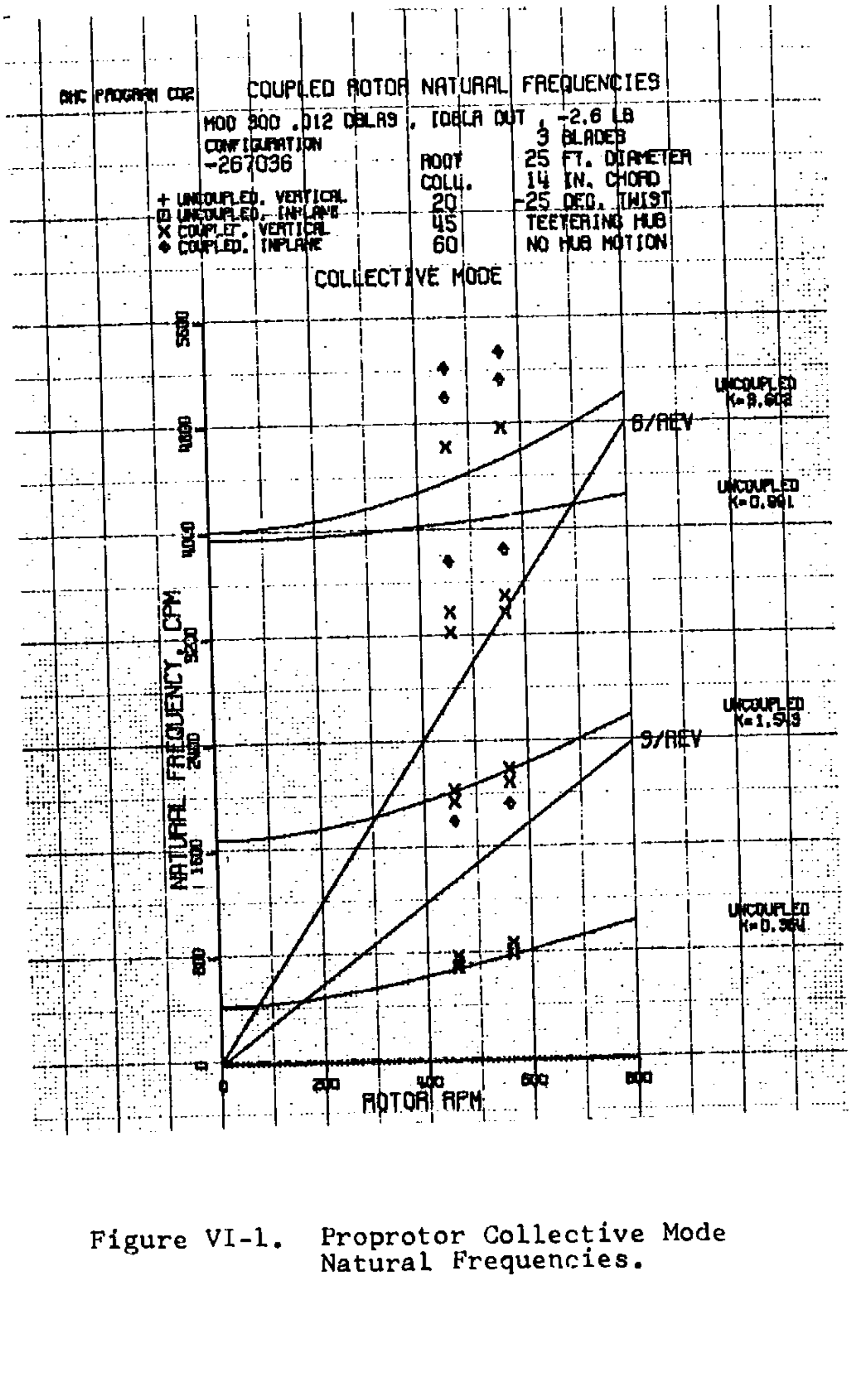

Figure VI-1. Proprotor Collective Mode Natural Frequencies.

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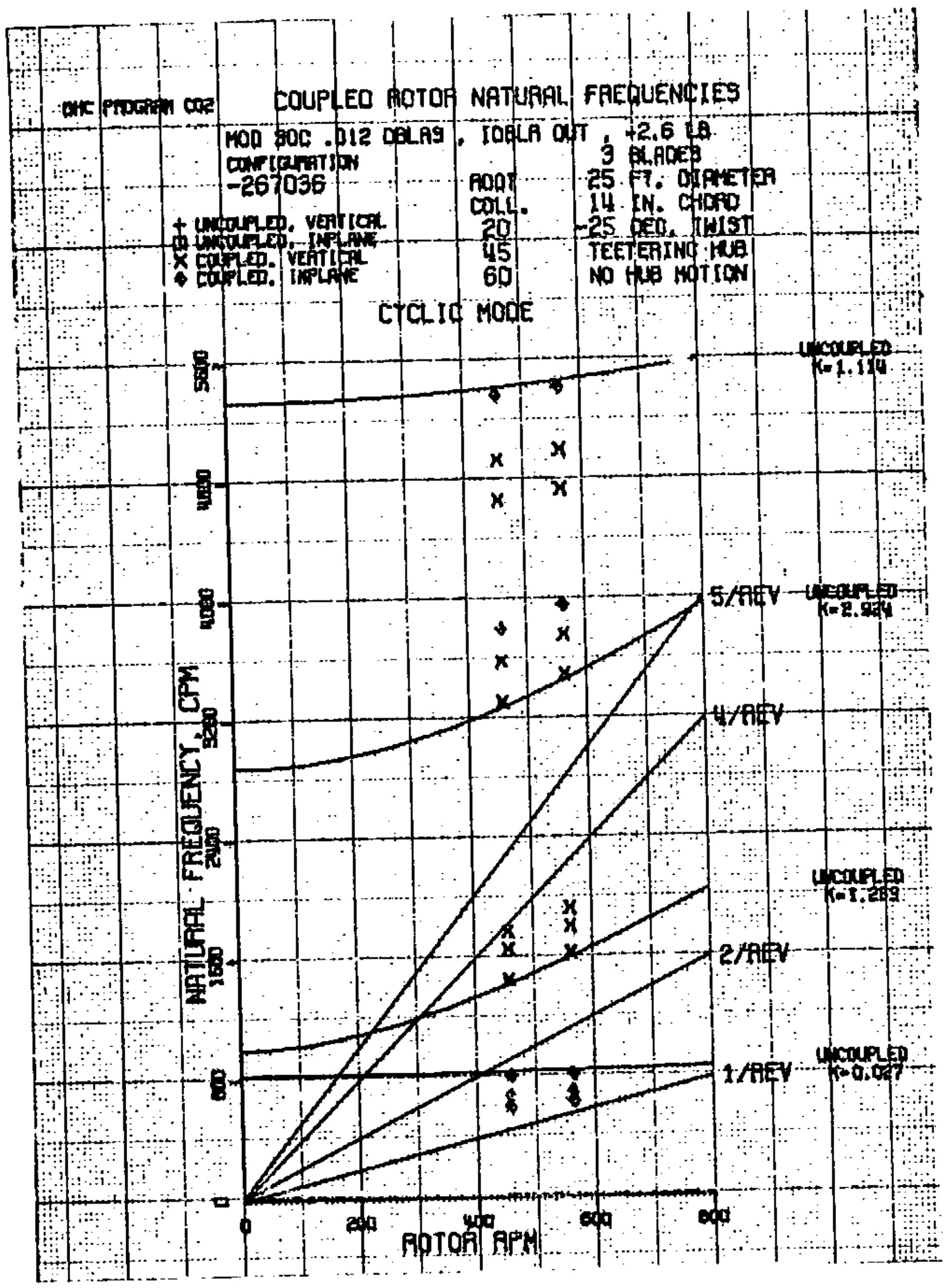

Figure VI-2. Proprotor Cyclic Mode Natural Frequencies.

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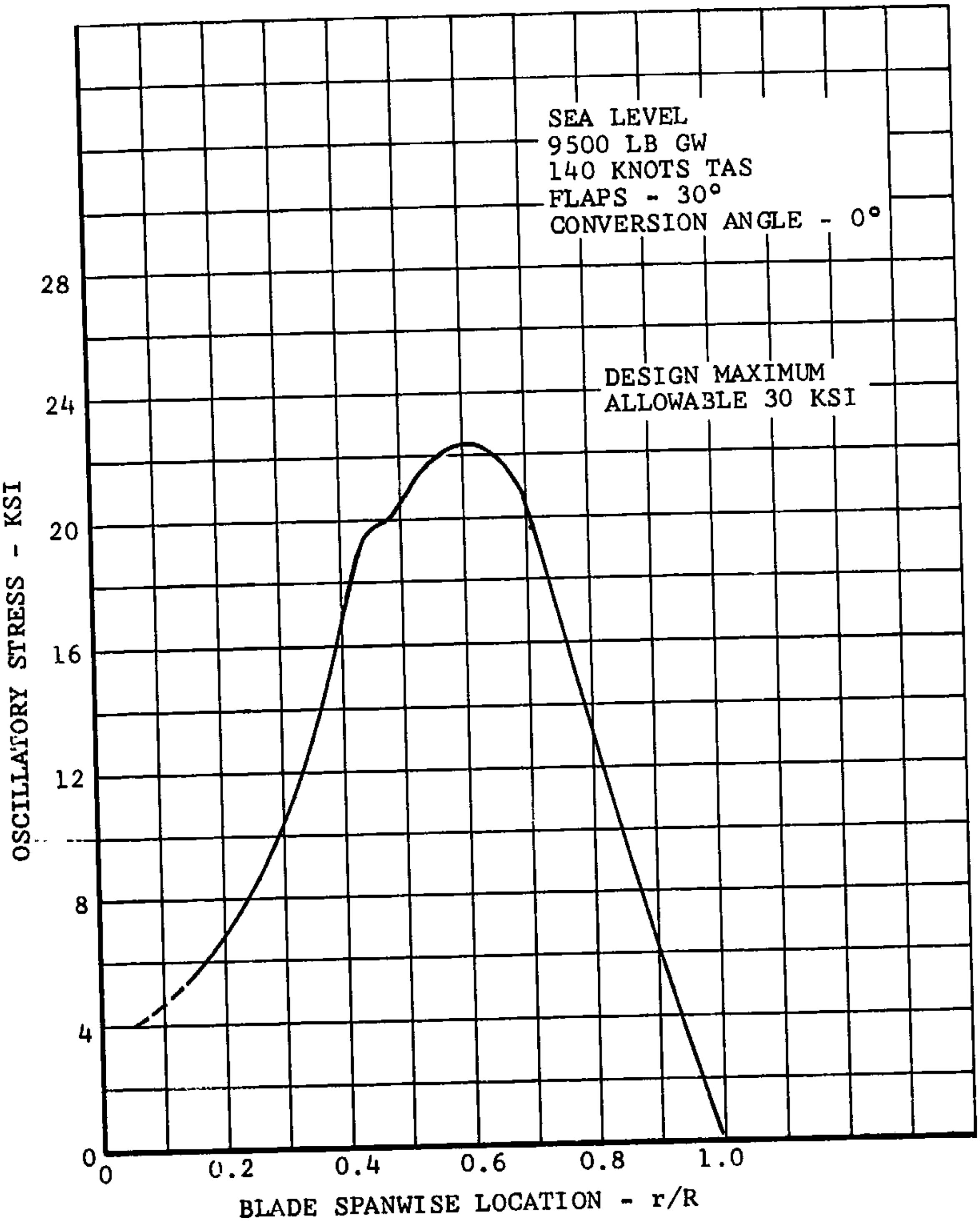


Figure VI-3. Blade Oscillatory Stress at V_H, Helicopter Mode.

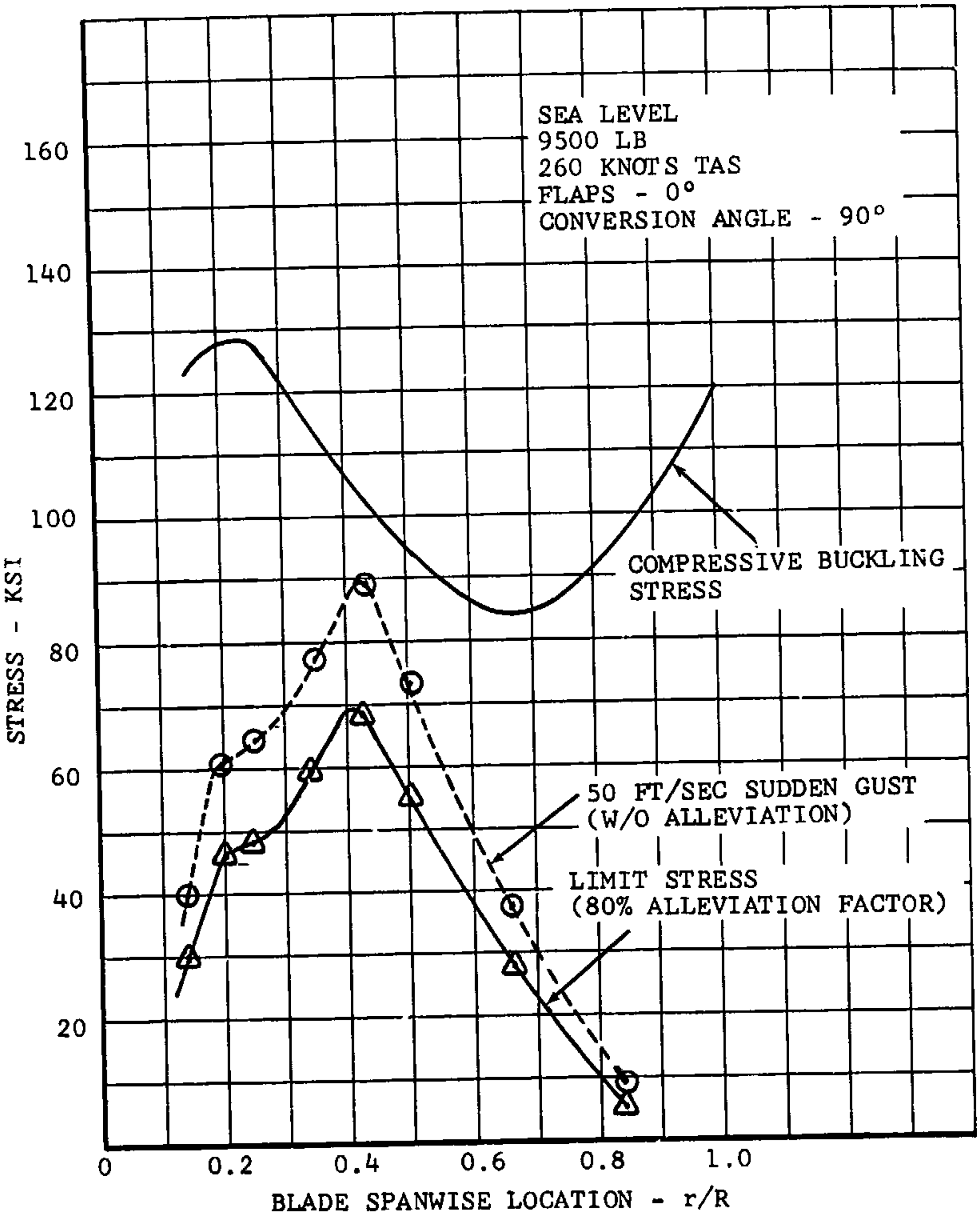


Figure VI-4. Blade Limit Stress at V_H, Airplane Mode.

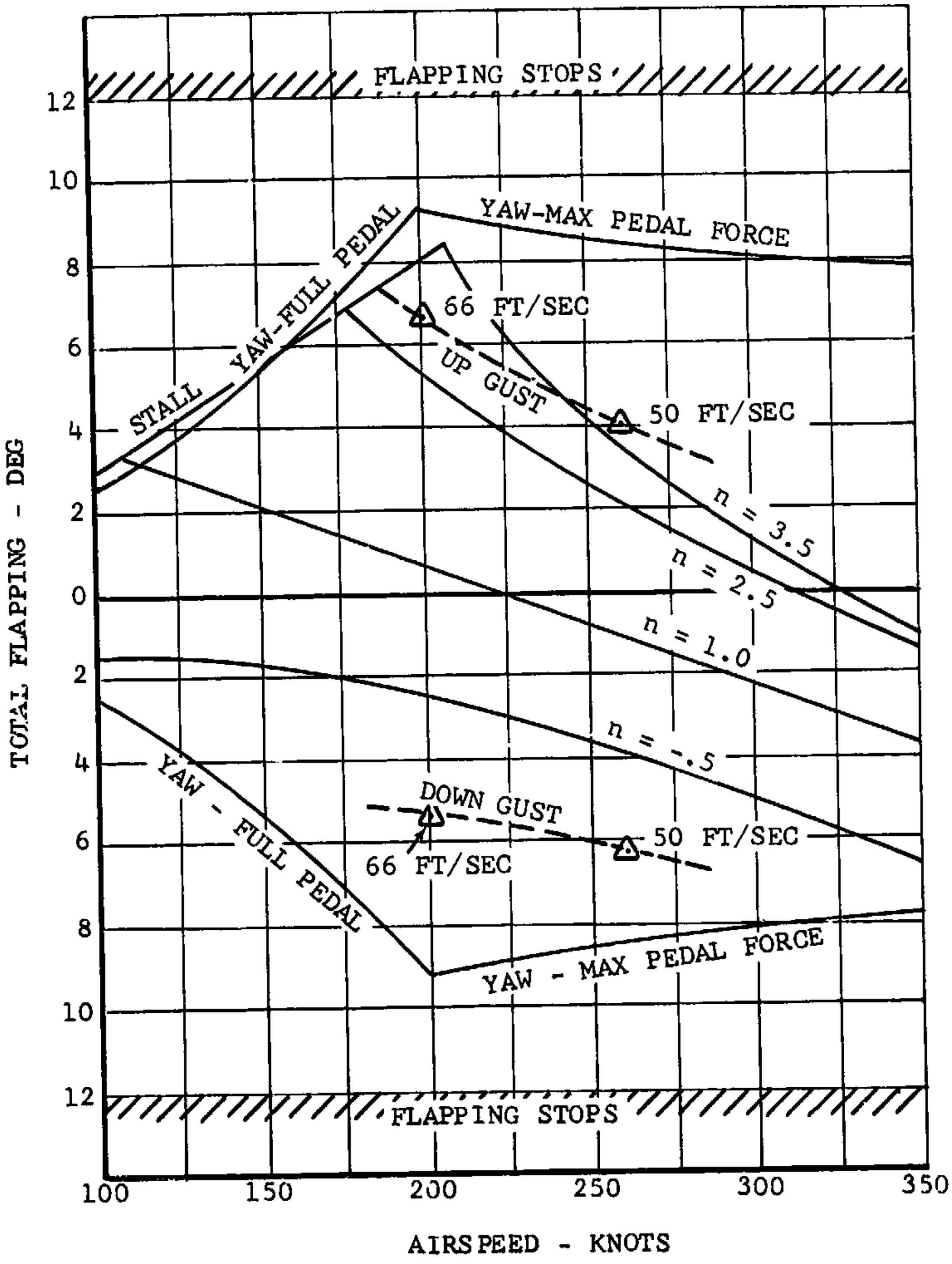


Figure VI-5. Flapping Envelope in Airplane Mode.

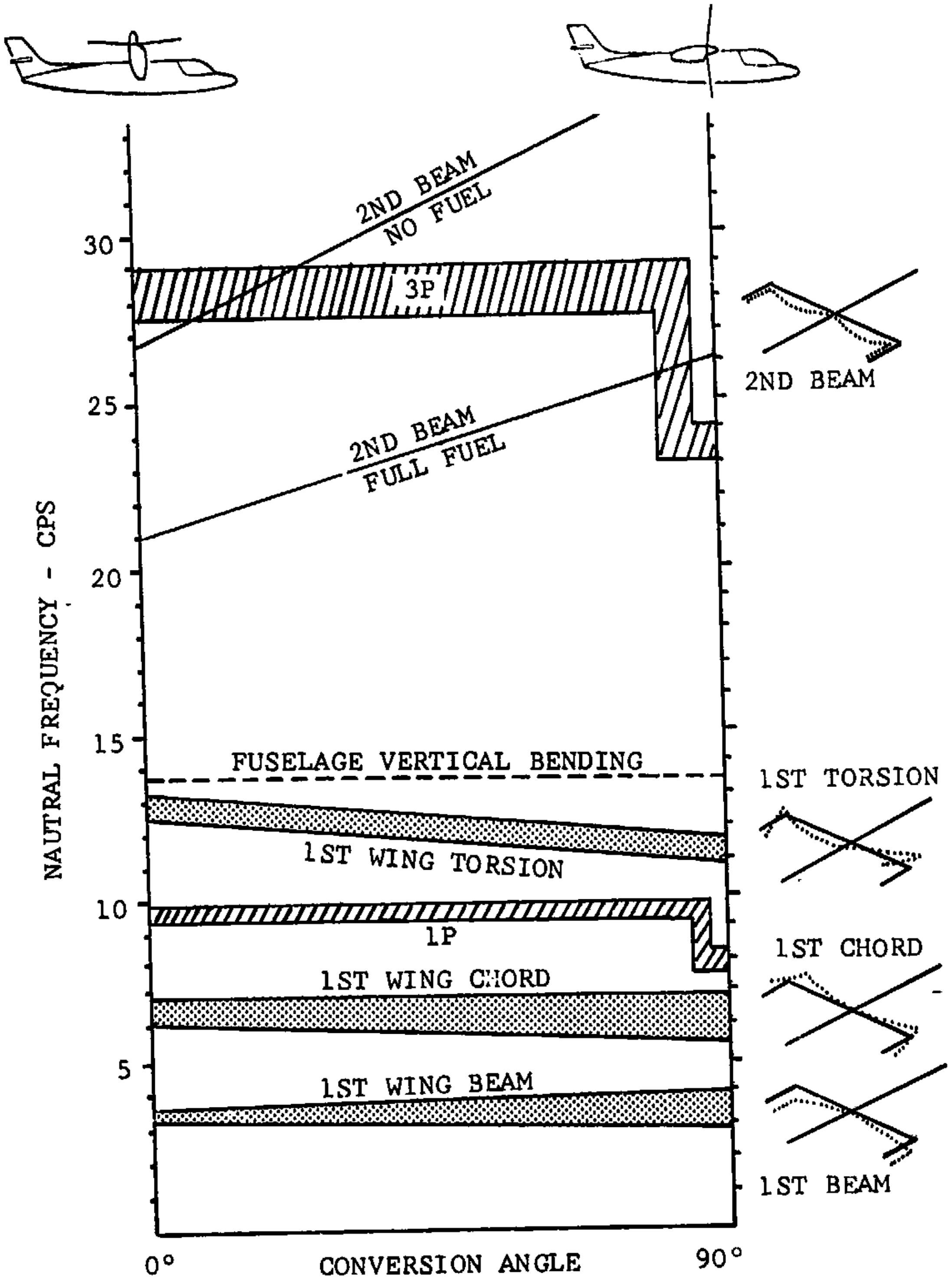


Figure VI-6. Airframe Symmetric Mode Natural Frequencies.

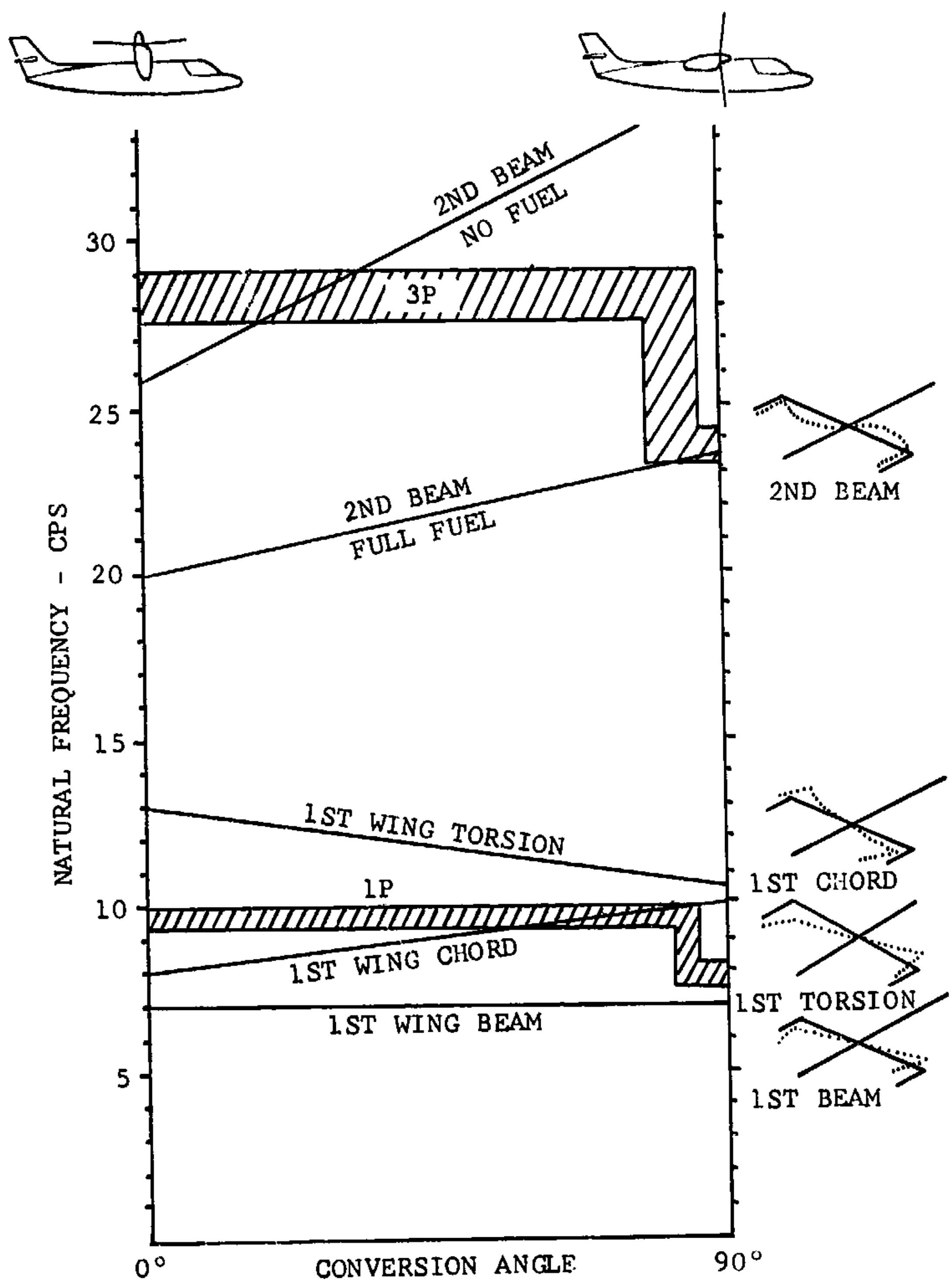


Figure VI-7. Airframe Asymmetric Mode Natural Frequencies.

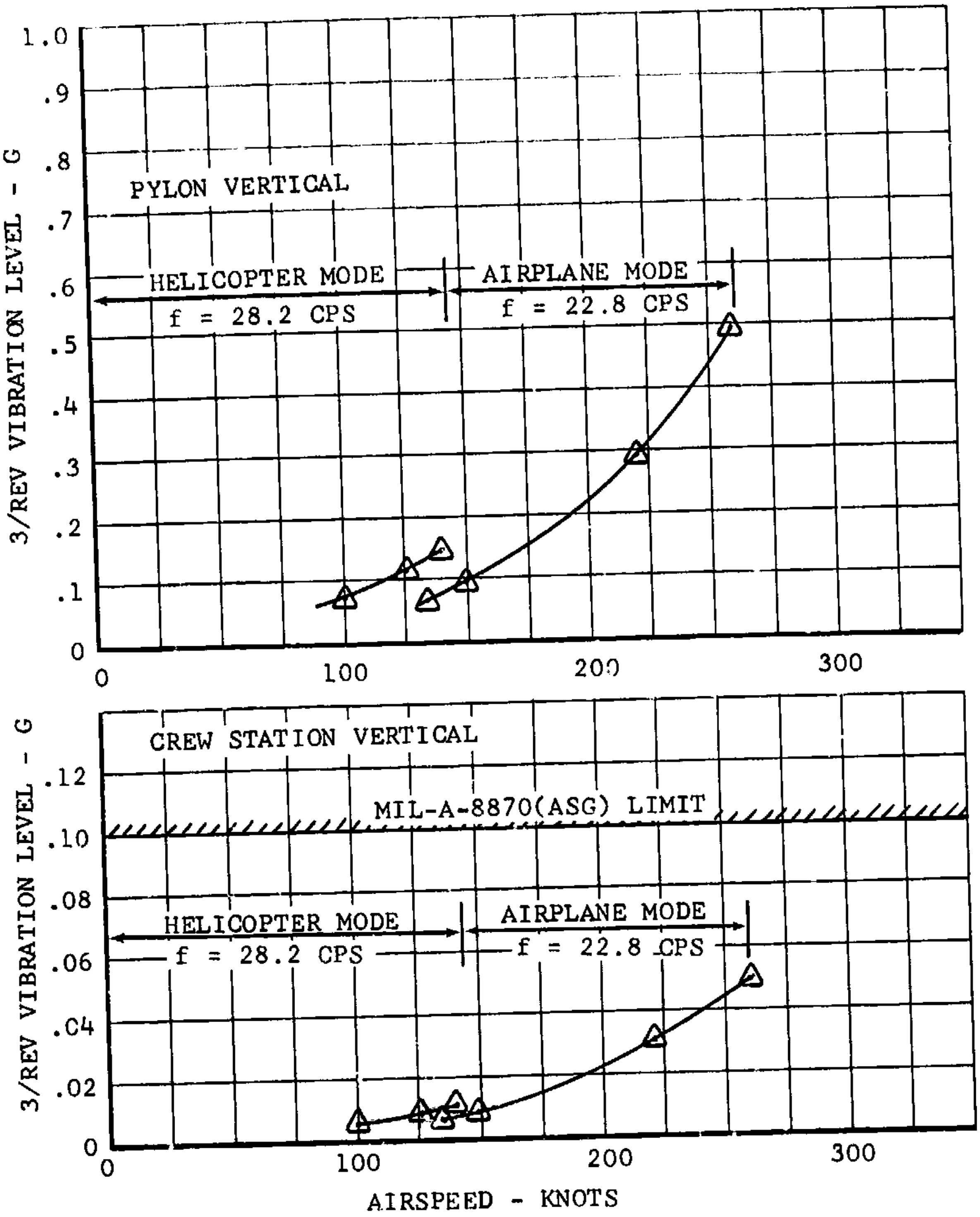
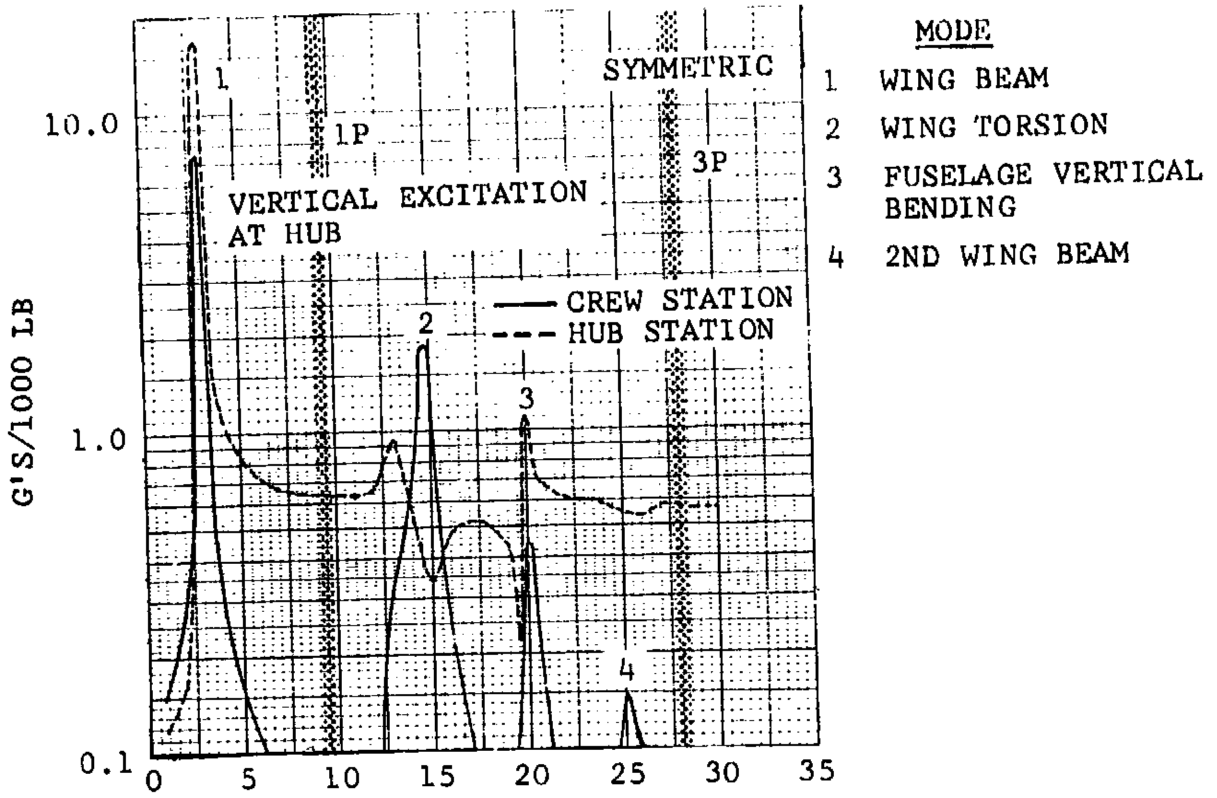


Figure VI-8. Pylon and Crew Station Vibration.



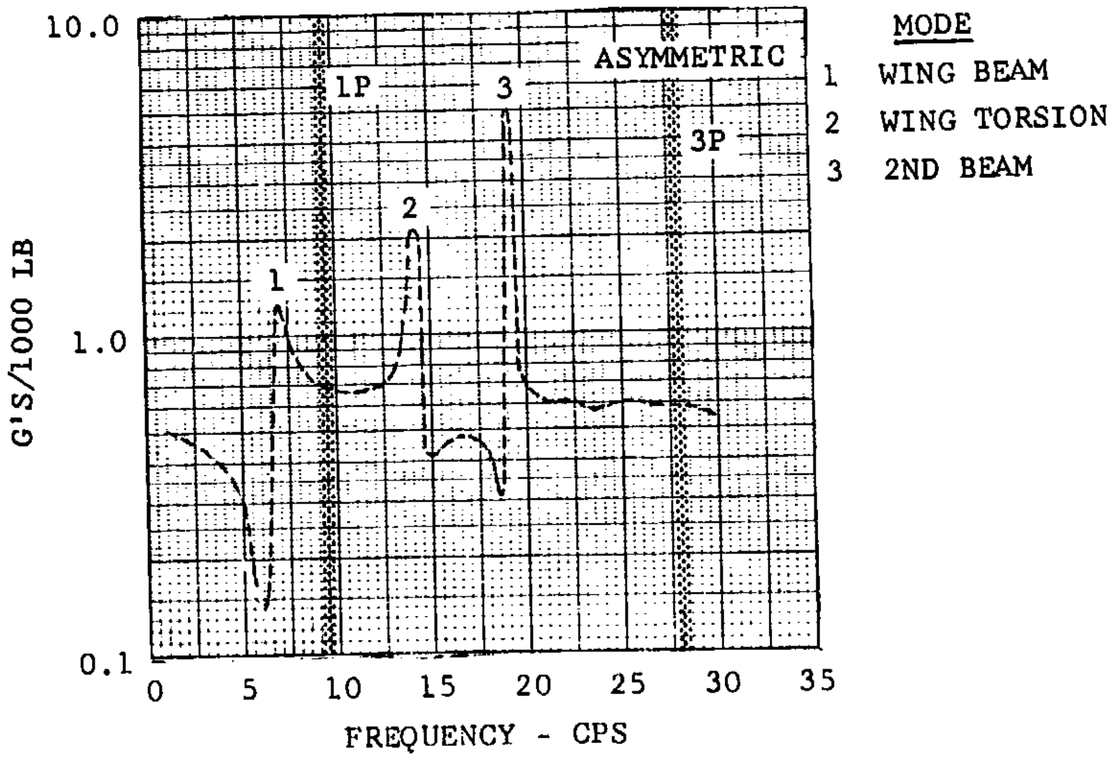


Figure VI-9. Pylon and Crew Station Frequency Response, Helicopter Mode.

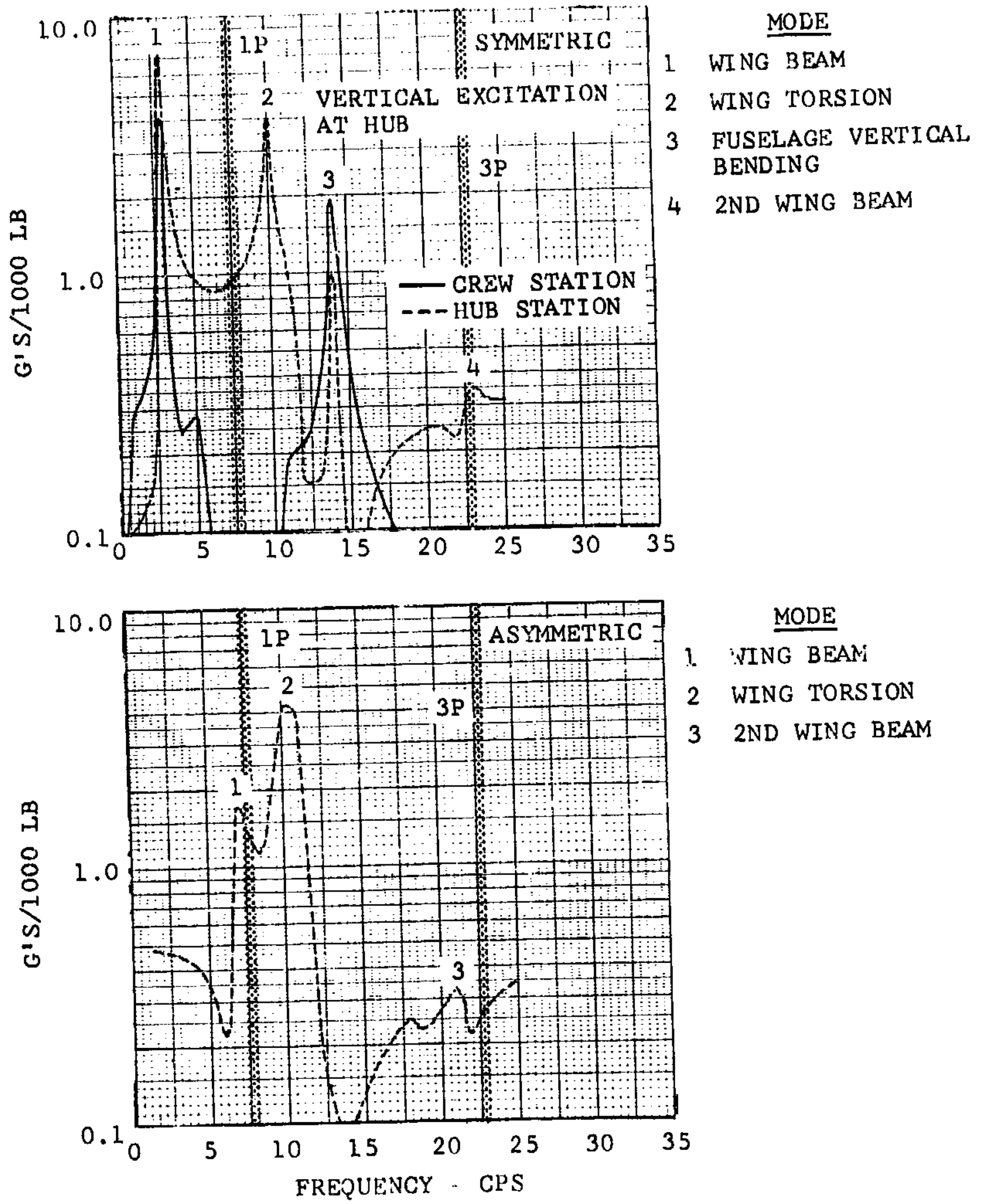


Figure VI-10. Pylon and Crew Station Frequency Response, Airplane Mode.

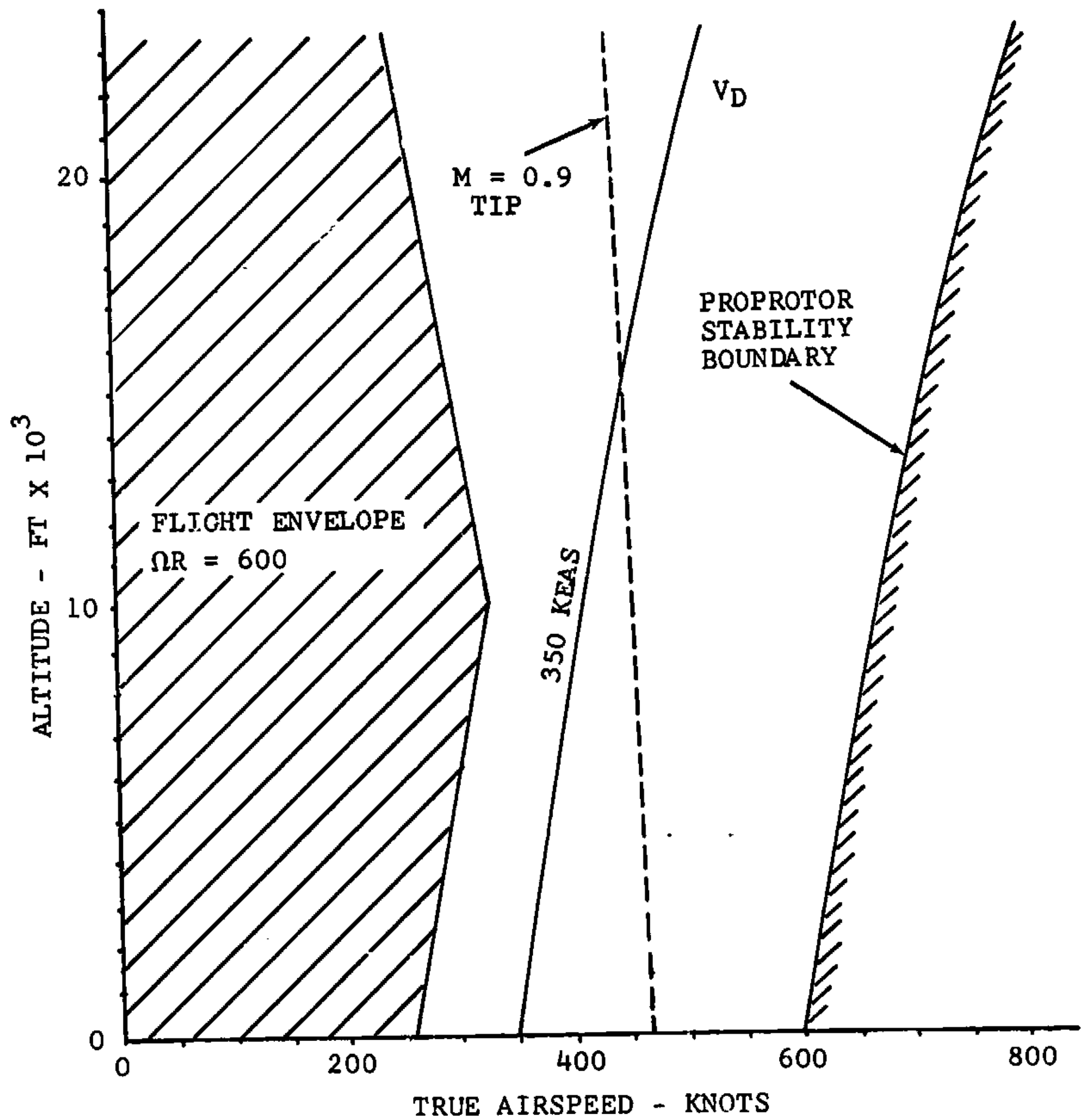


Figure VI-11. Model 300 Proprotor Stability Boundary.

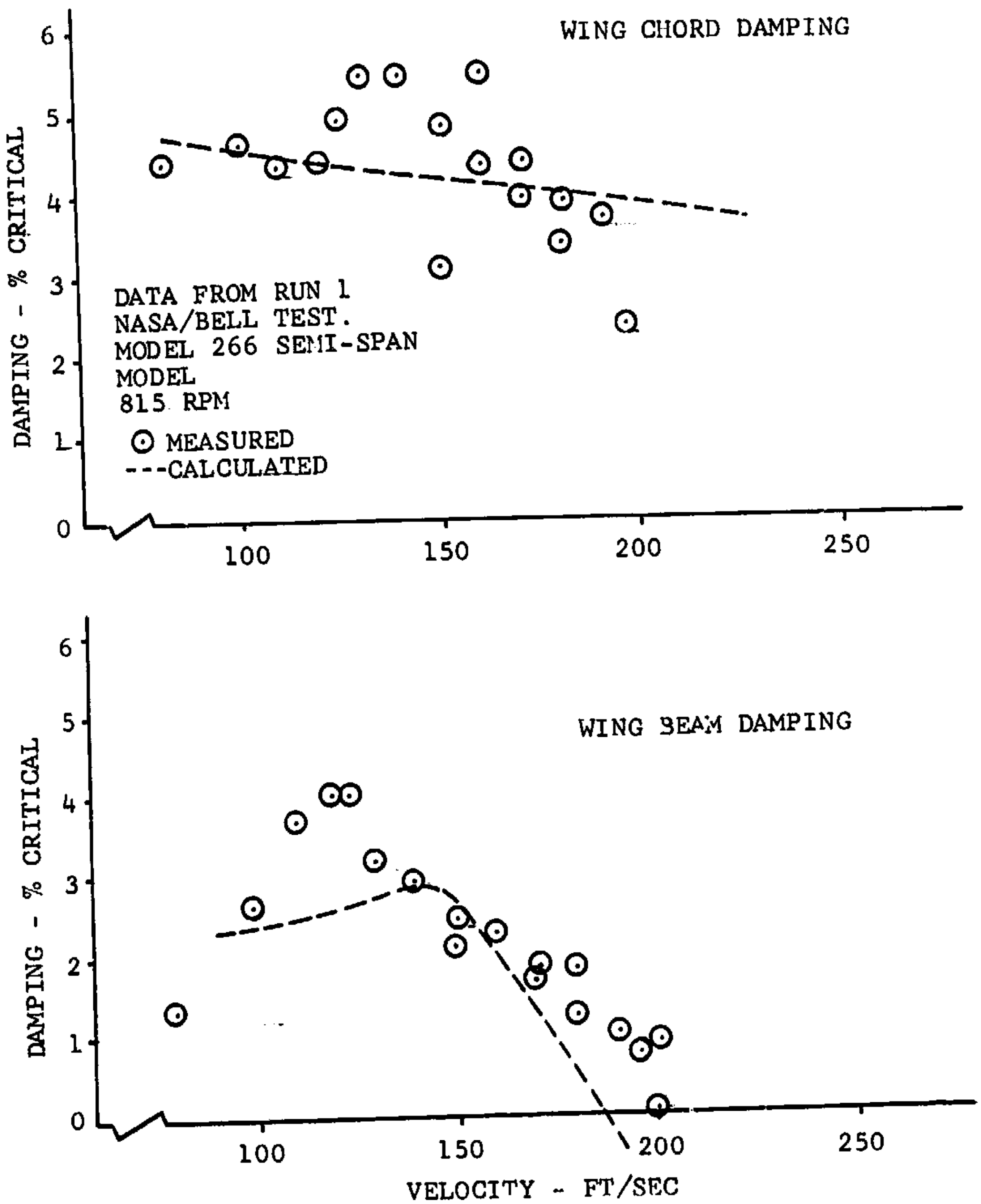


Figure VI-12. Correlation of Calculated and Measured Proprotor Stability - NASA-Bell Test of Model 266 Aeroelastic Model.

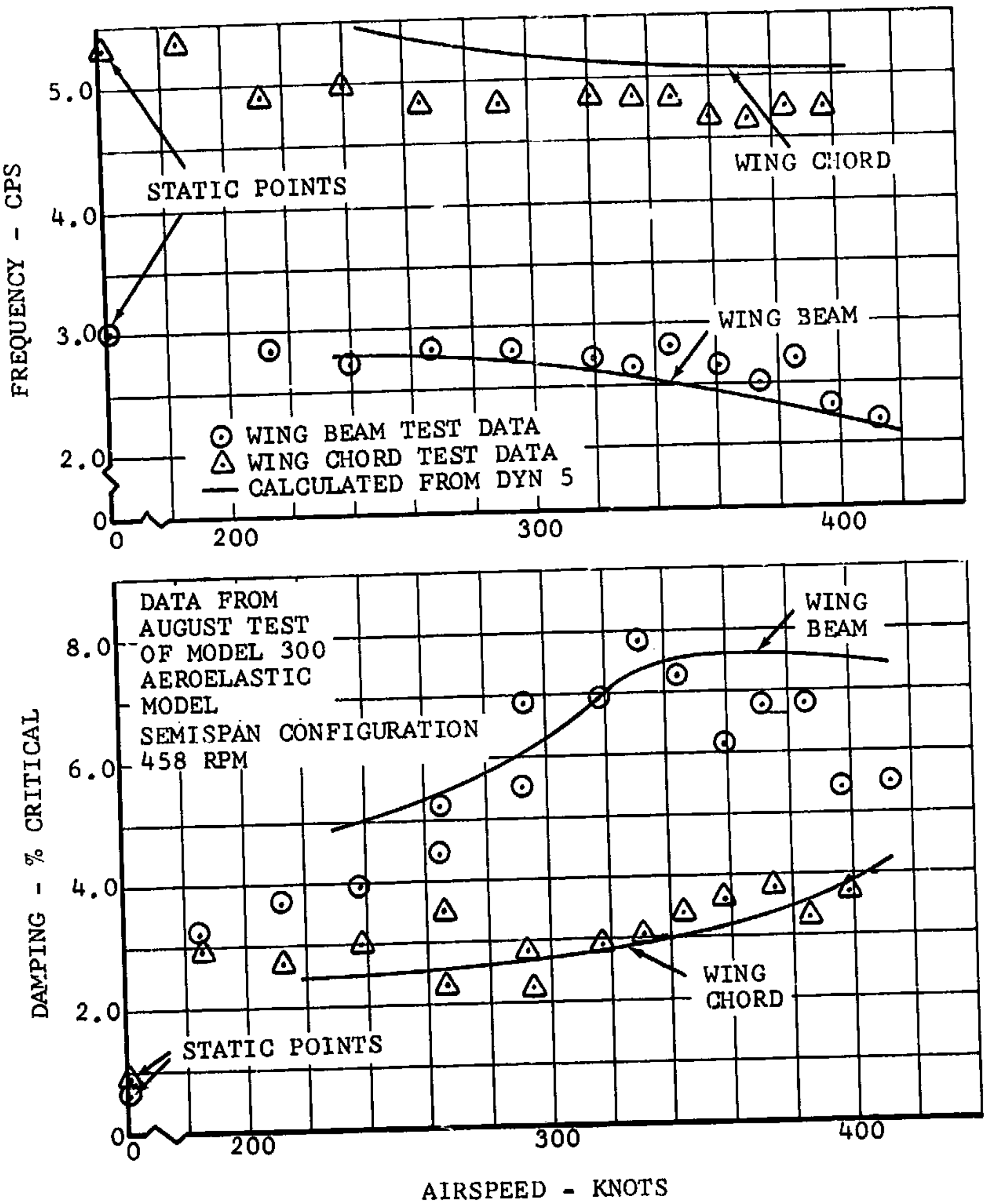


Figure VI-13. Model 300 Aeroelastic Model Proprotor Stability (Full-Scale Frequency and Airspeed).

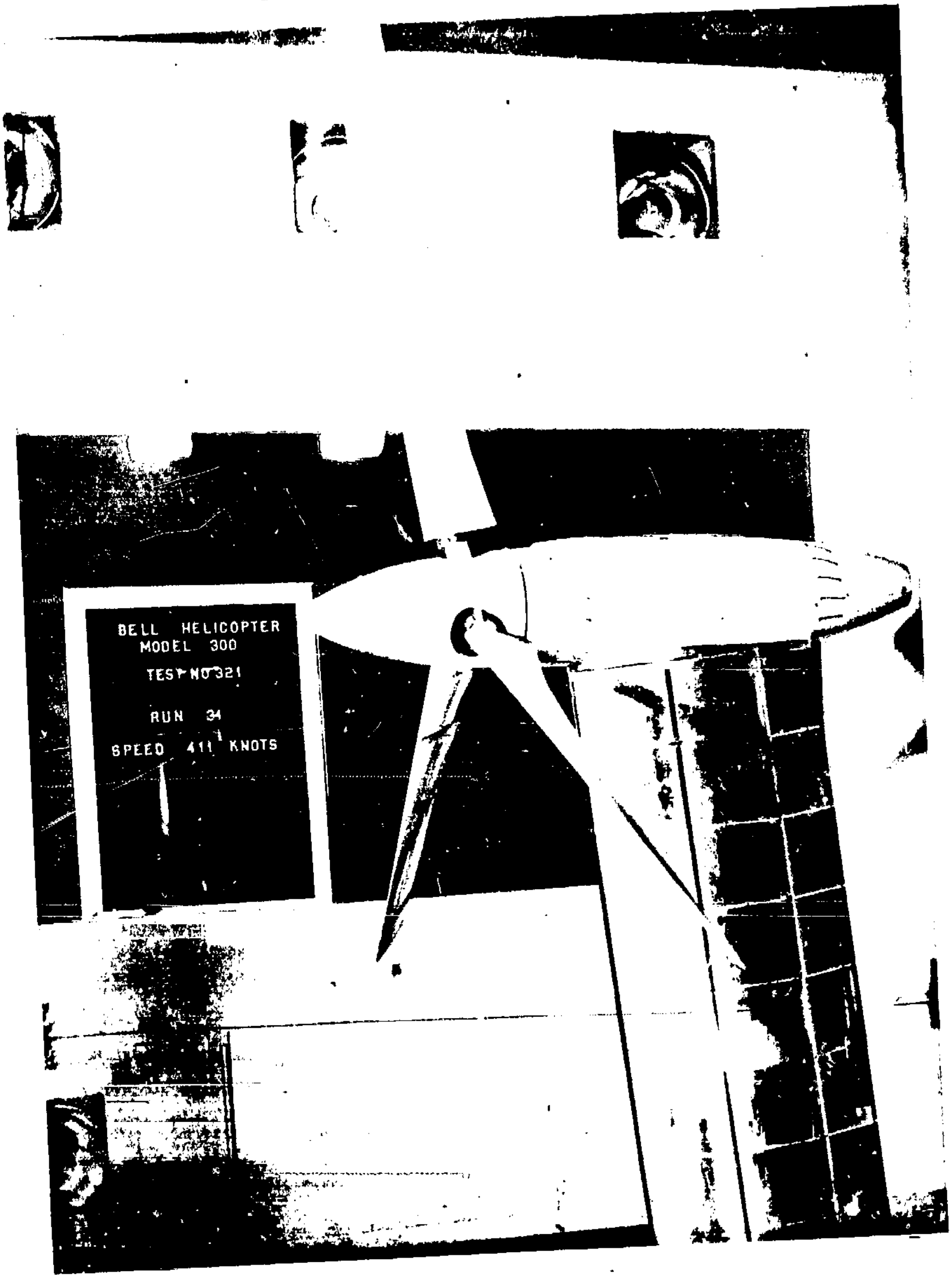


Figure VI-14. Model 300 Aeroelastic Model in 7-by-10-Foot Tunnel.

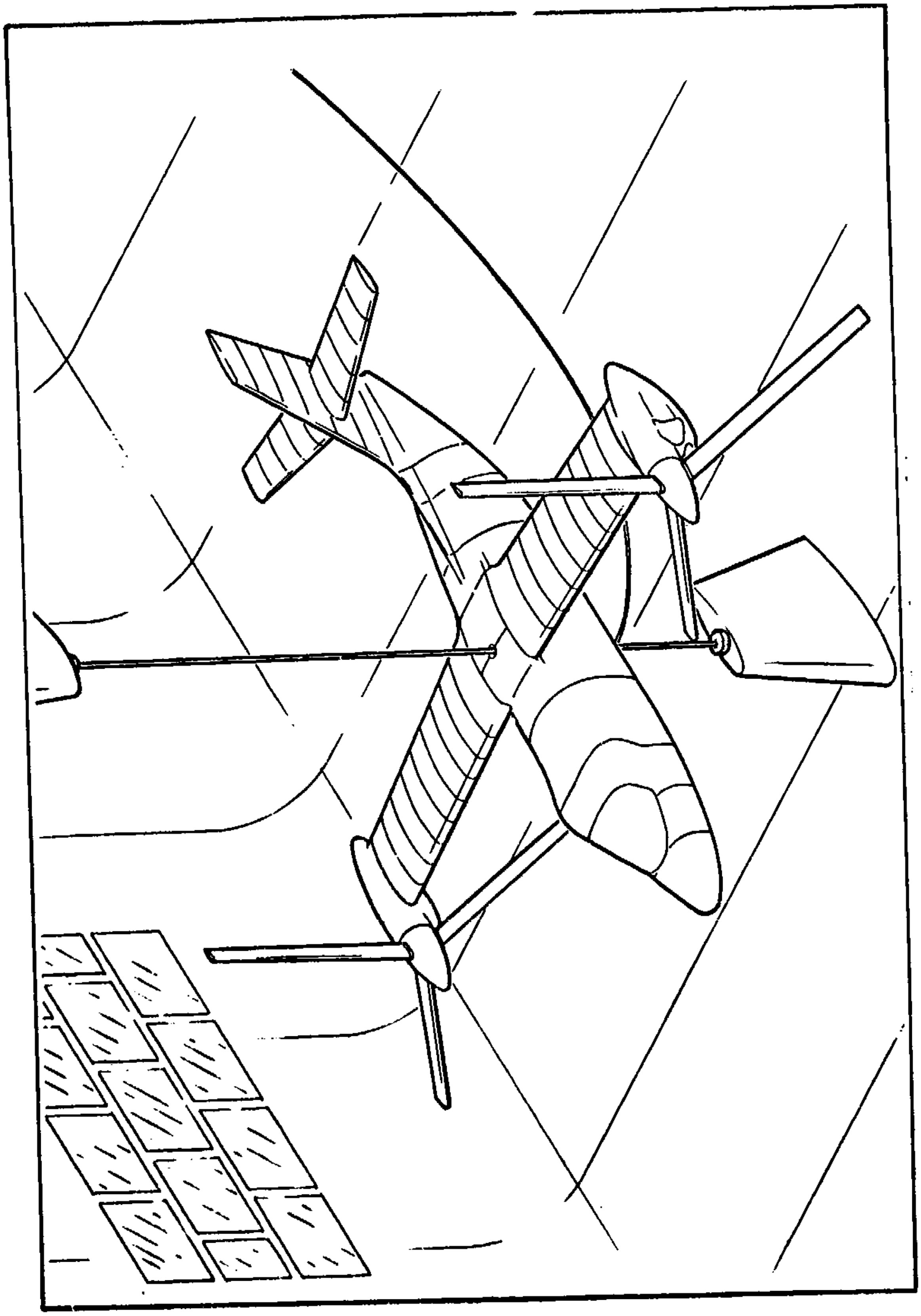


Figure VI-15. Model 300 Full-Span Model in NASA-Langley 16-Foot Transonic Dynamic Tunnel.

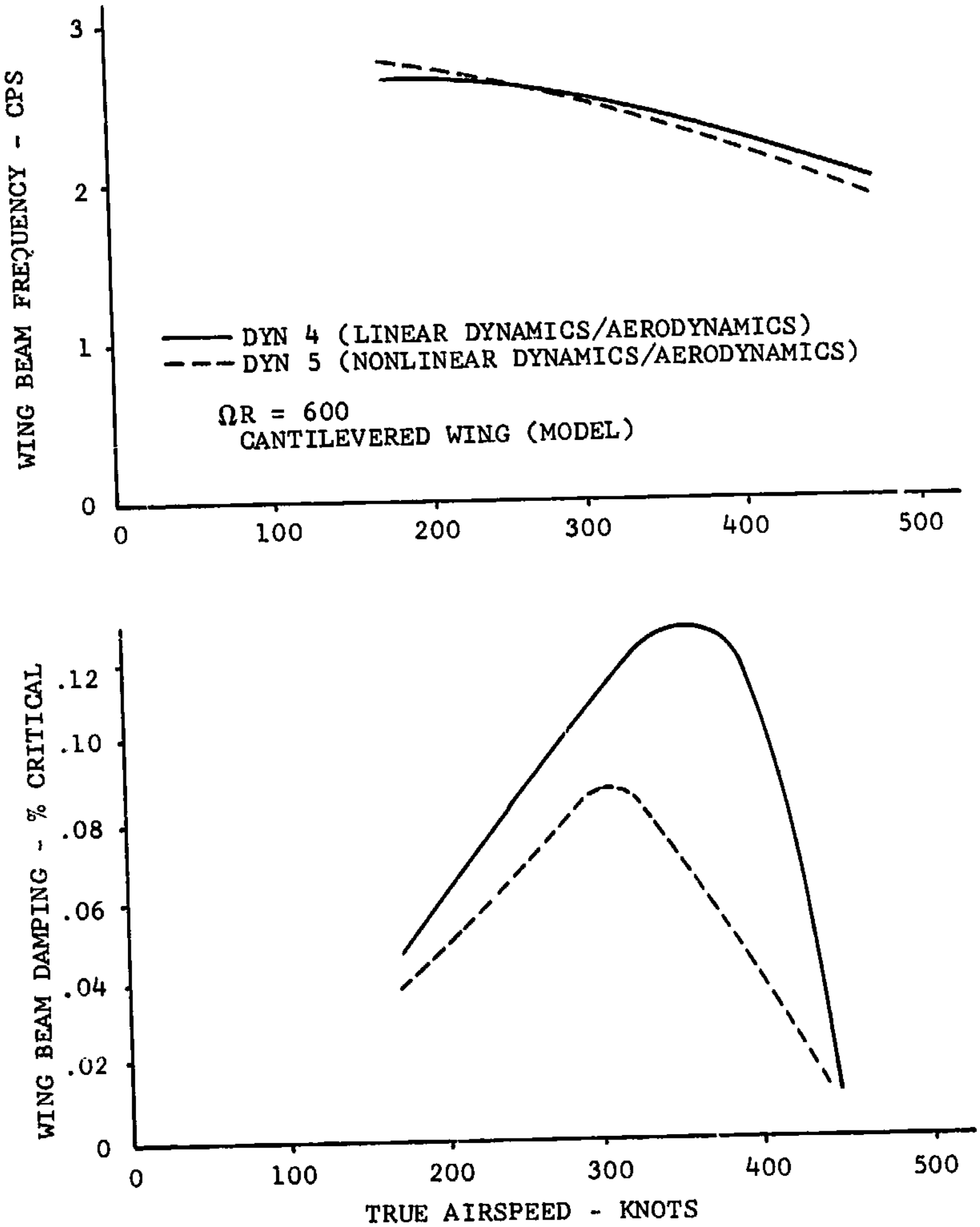


Figure VI-16. Comparison of Stability Analysis Method.

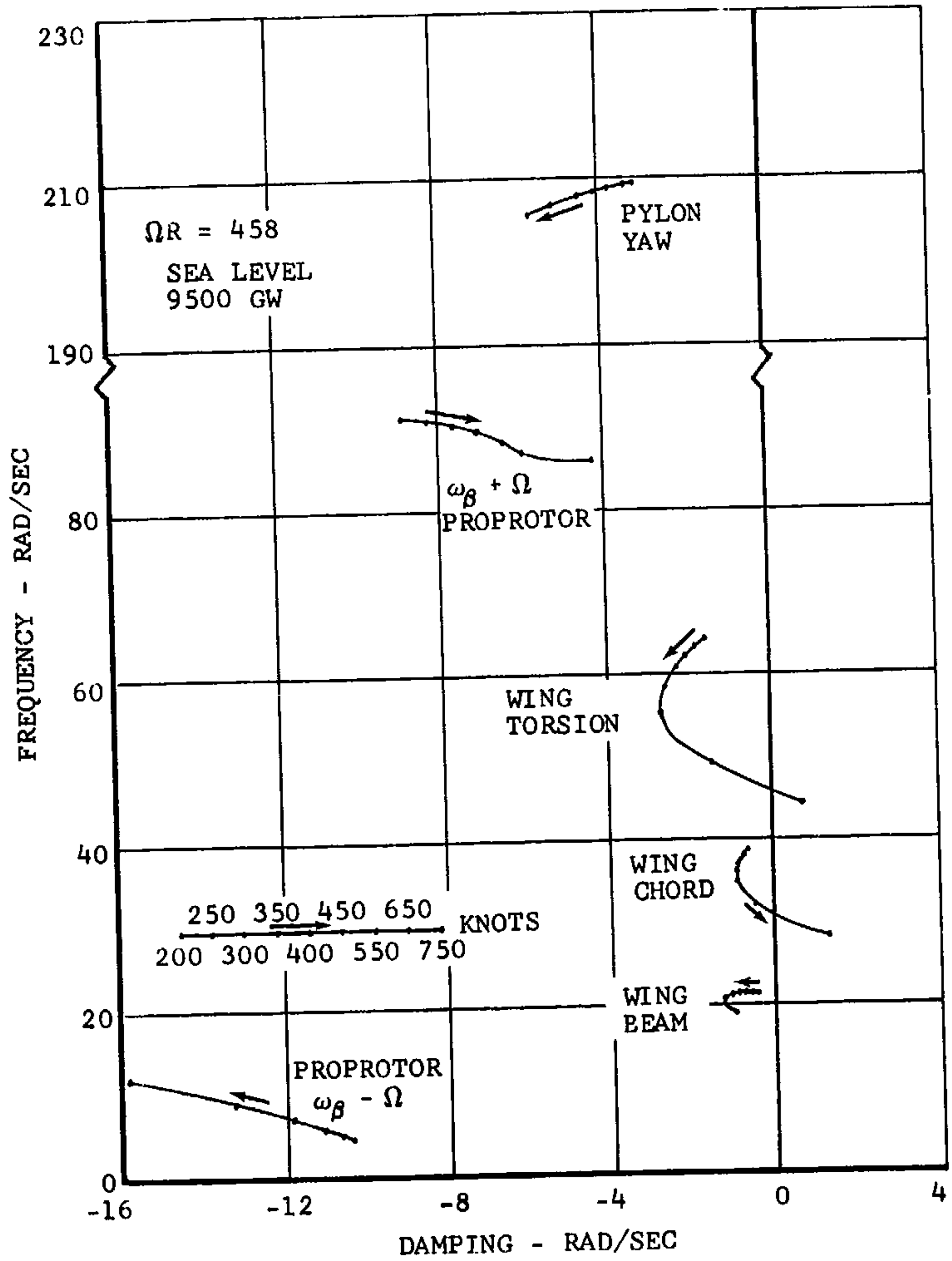


Figure VI-17. Root Locus of Symmetric Free-Free Modes.

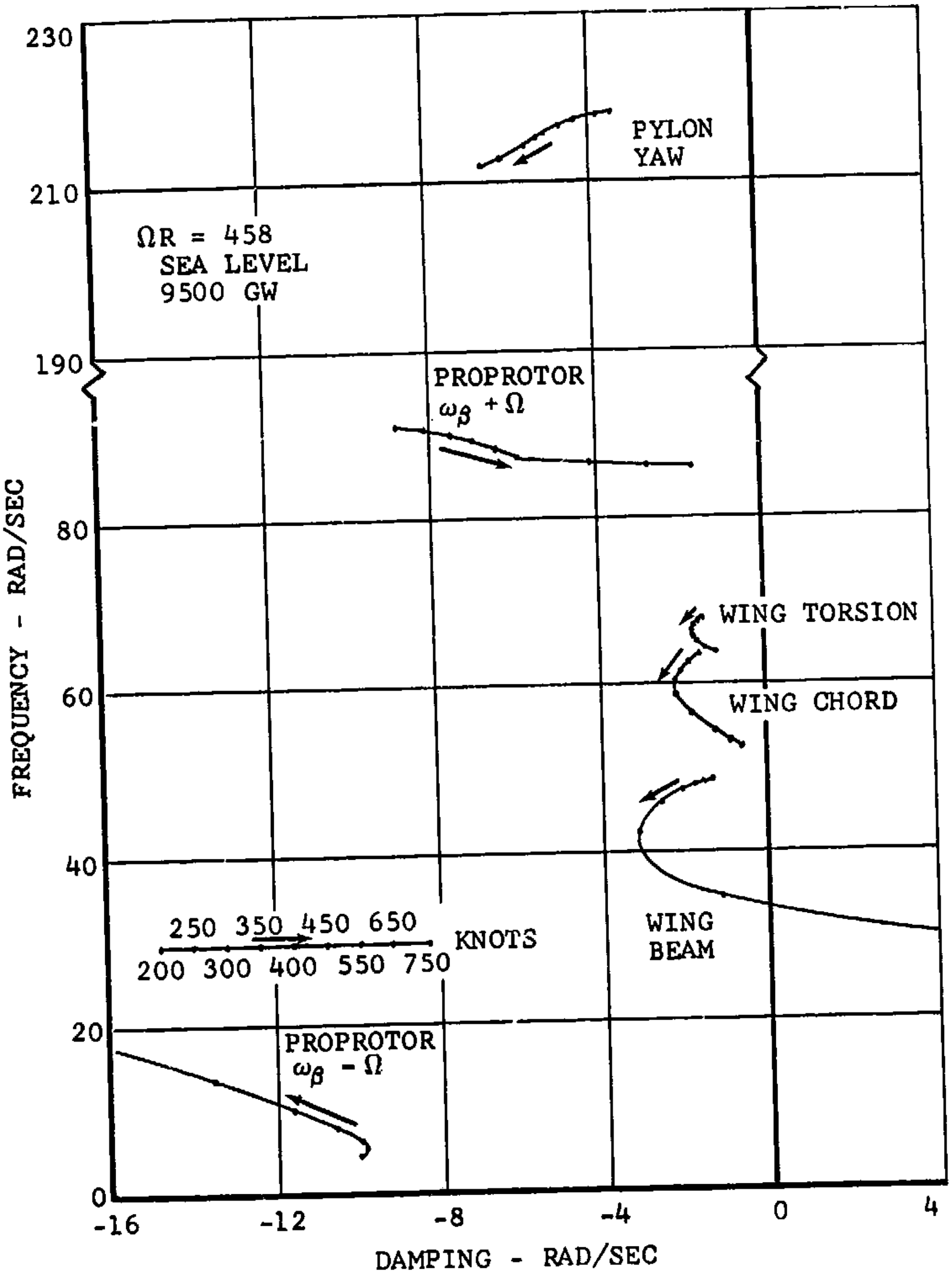


Figure VI-18. Root Locus of Asymmetric Free-Free Modes.

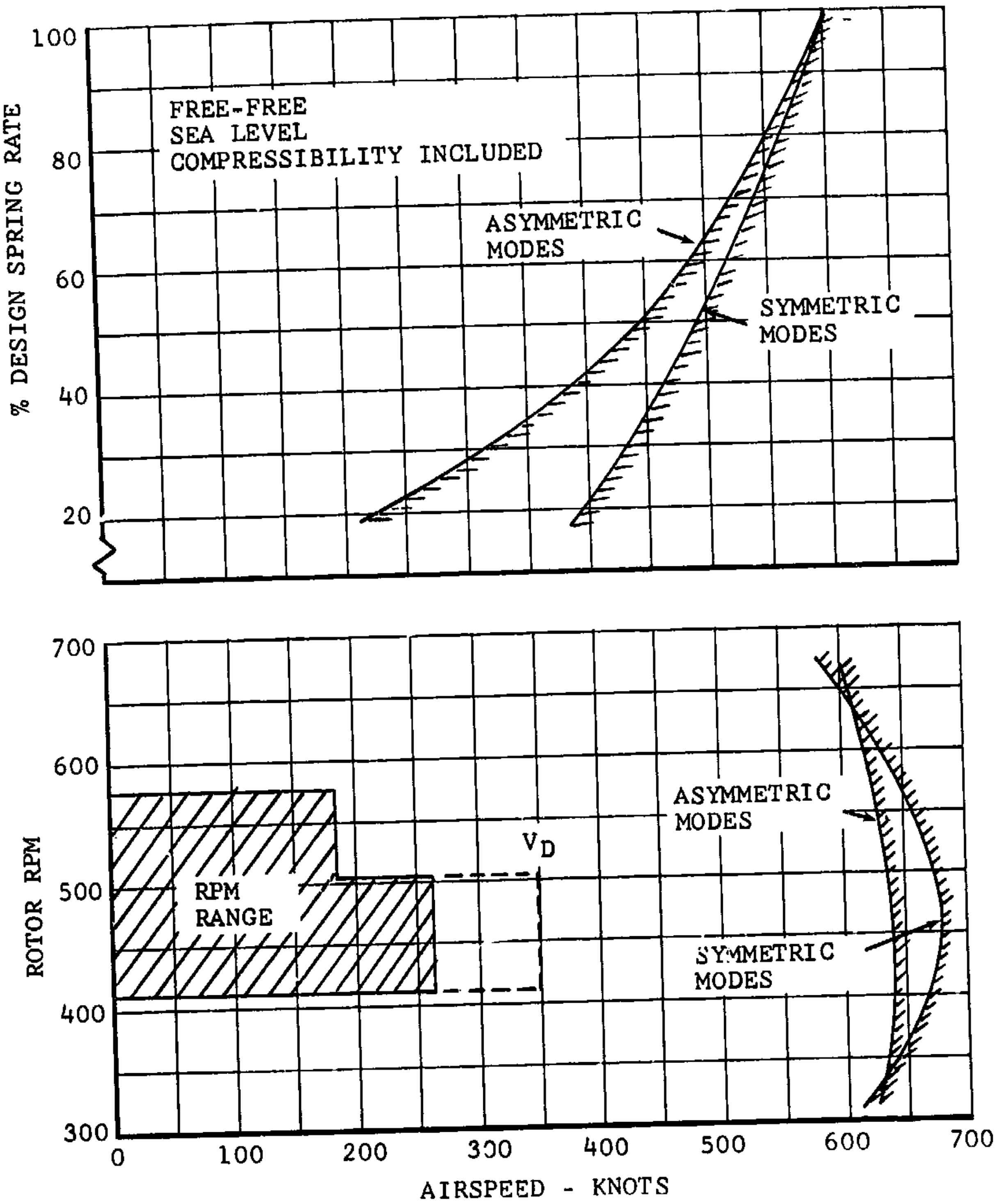


Figure VI-19. Proprotor Stability Sensitivity to Wing Stiffness and RPM.

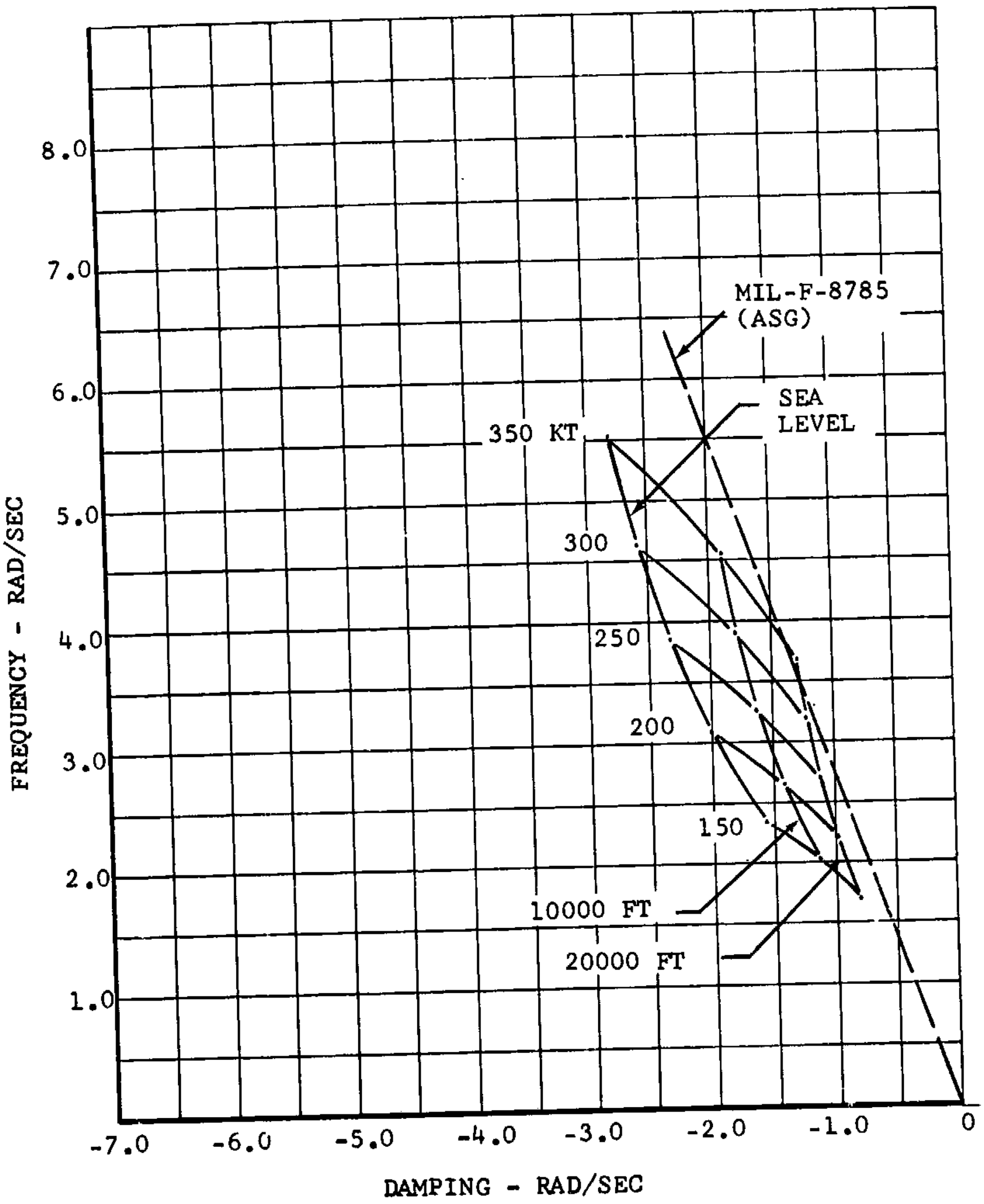


Figure VI-20. Root Locus of Longitudinal Short-Period Mode.

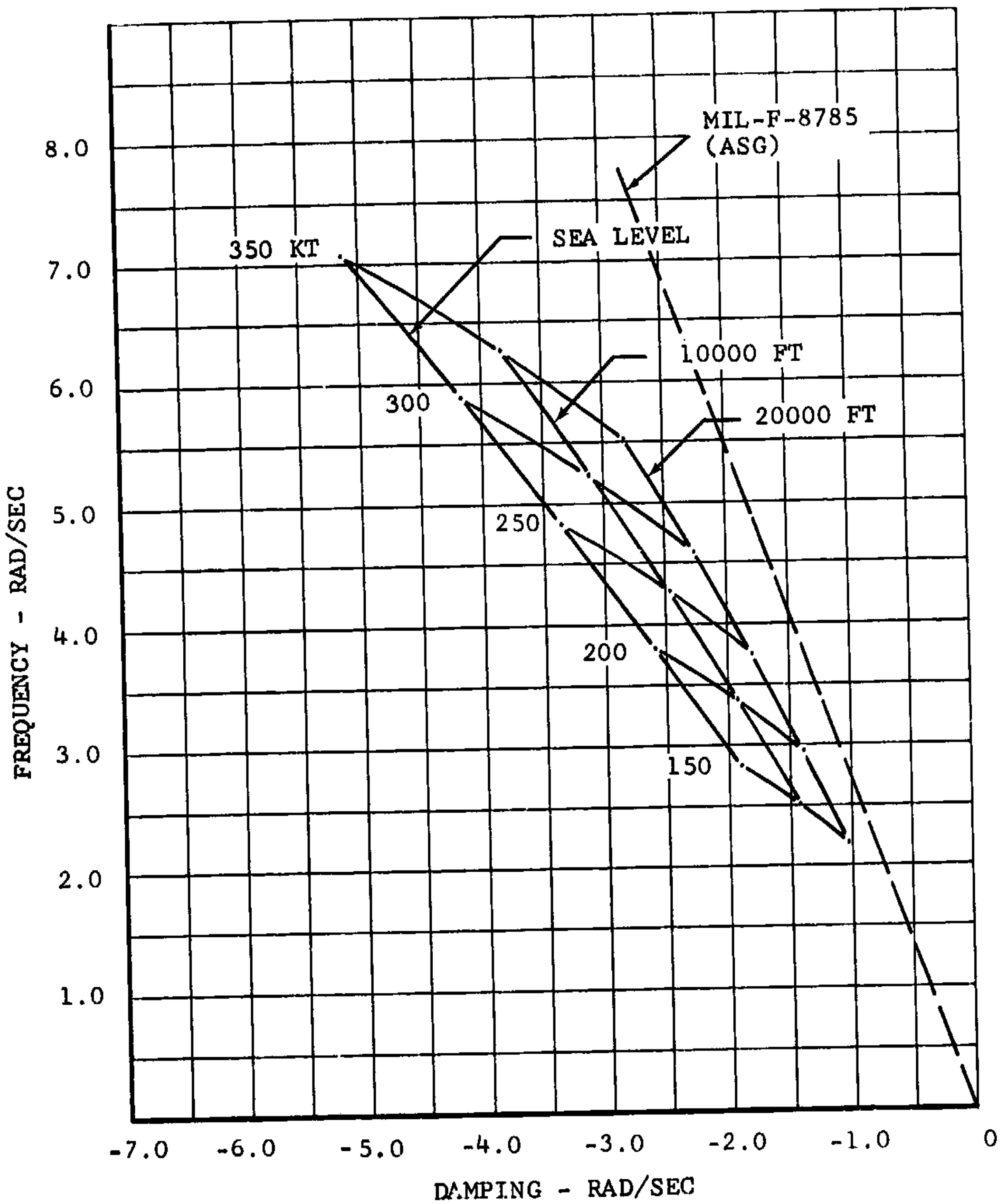


Figure VI-21. Root Locus of Longitudinal Short-Period Mode of Basic Airframe.

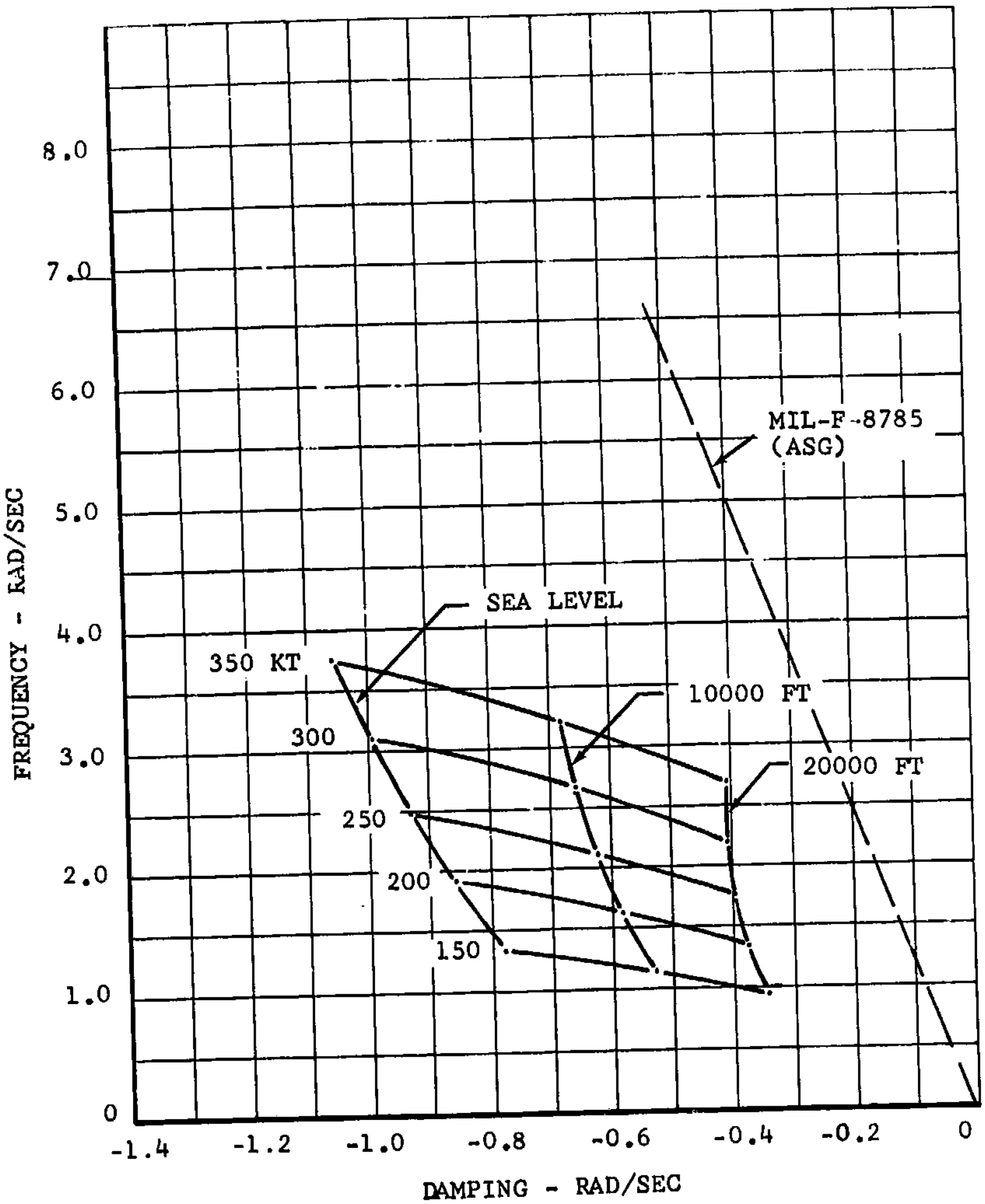


Figure VI-22. Root Locus of Dutch-Roll Mode.

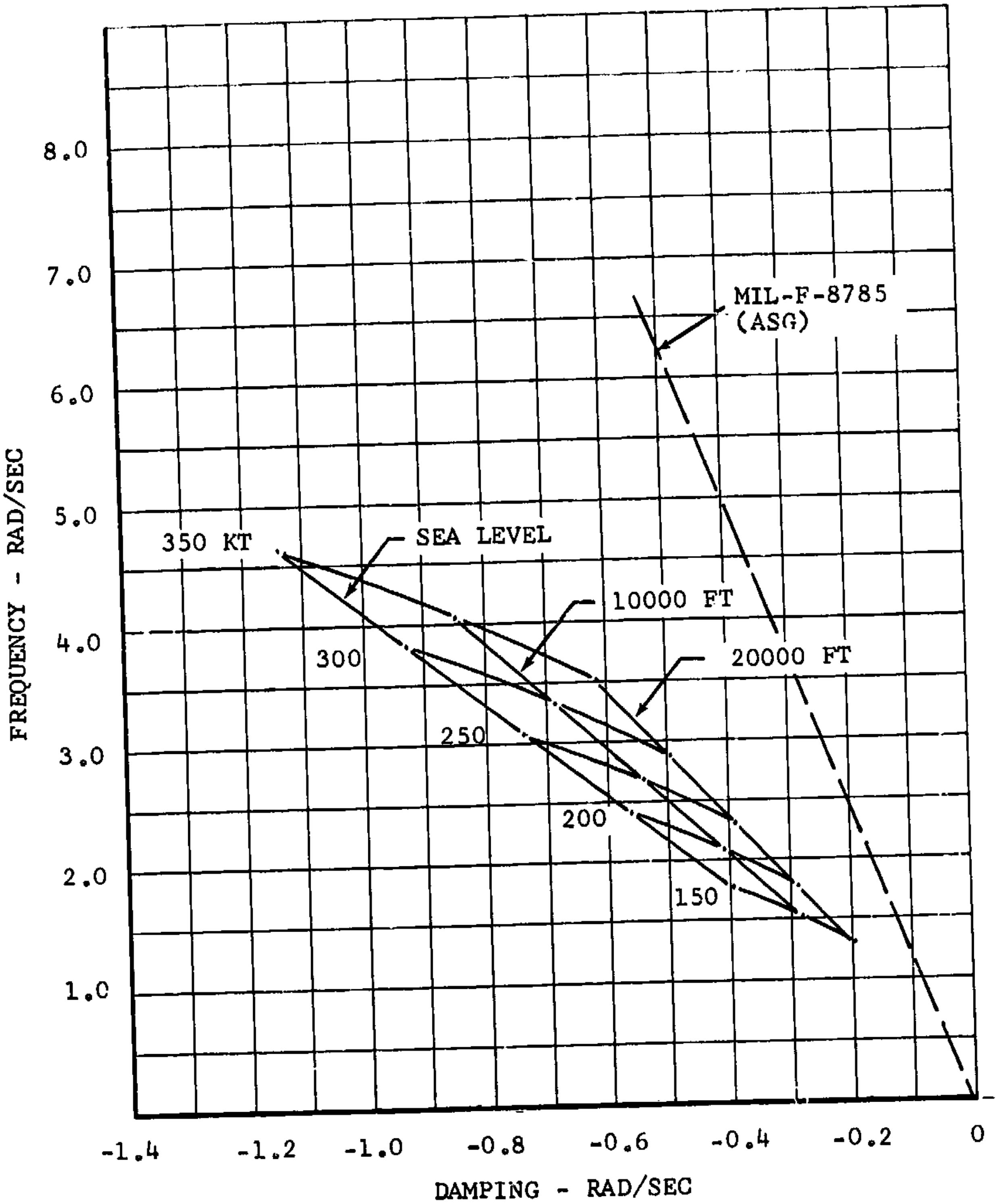


Figure VI-23. Root Locus of Dutch-Roll Mode for Basic Airframe.

VII. NOISE

An analysis of the merits of any particular VTOL design must include the consideration of noise radiated into communities and areas adjacent to VTOL sites. Close to such sites, the community noise problem will probably be worse than that experienced at our busiest airports today. Three factors tend to make the small downtown or suburban heliport more of a community noise problem than conventional airports: the relatively high power settings necessary during landing and takeoff, the small separation between the aircraft and exposed communities, and the low ambient noise environments at many of these locations.

Although considerable amount of experimental data and prediction methods exist for analyzing noise of fixed-wing aircraft and of conventional helicopters, the complex noise characteristics of future VTOL aircraft cannot be fully described for many of the less orthodox propulsive systems. However, sufficient information exists to permit a reasonable estimate to be made based on noise characteristics of individual components of any given propulsive system, i.e., propellers, jet engines, rotors, etc.

A measure of noise called the perceived noise level, which is expressed in units of PNdb, relates the measurements recorded by acoustical instrumentation to the subjective impressions people experience when they are exposed to sound. This measure rates the annoyance or noisiness of complex sounds and is used in this country and abroad for subjective-judgment studies of traffic, industrial and aircraft noise. Perceived noise level takes into account the distribution of sound energy over the audible frequency range and gives a direct and simplified measure of the subjective noisiness of different VTOL aircraft.

Perceived noise levels calculated for the Model 300 are compared in Figure VII-1 with estimated levels of VTOL aircraft proposed for (60-passenger) short-haul service and with measured levels of present-day helicopters. These curves show that rotor-type aircraft will be least noisy, while jet types will be the noisiest. The noise of the Model 300 will be 20 PNdb less than that for tilt-wing VTOL and 40 PNdb less than that for jet-lift aircraft. The Model 300 will also be quieter than conventional helicopters because it will not require a tail rotor, whose sound contributes to a helicopter's perceived noise. The maximum perceived noise level of the Model 300 in helicopter mode will be 93 PNdb at a distance of 300 feet.

Figure VII-1 also shows the range of perceived noise levels for heavy industrial areas. At a distance of 1000 feet, the noise of a hovering Model 300 will be no greater than that generated in heavy industrial areas. In addition, the Model 300 will be quieter when in airplane mode because the proprotor's tip speeds are substantially reduced. Cruising at an altitude of 1000 feet the noise of the Model 300 will be approximately 65 PNdb, a level comparable to ambients measured in commercial areas with light traffic. The Model 300 will not be loud enough to be heard in busier areas.

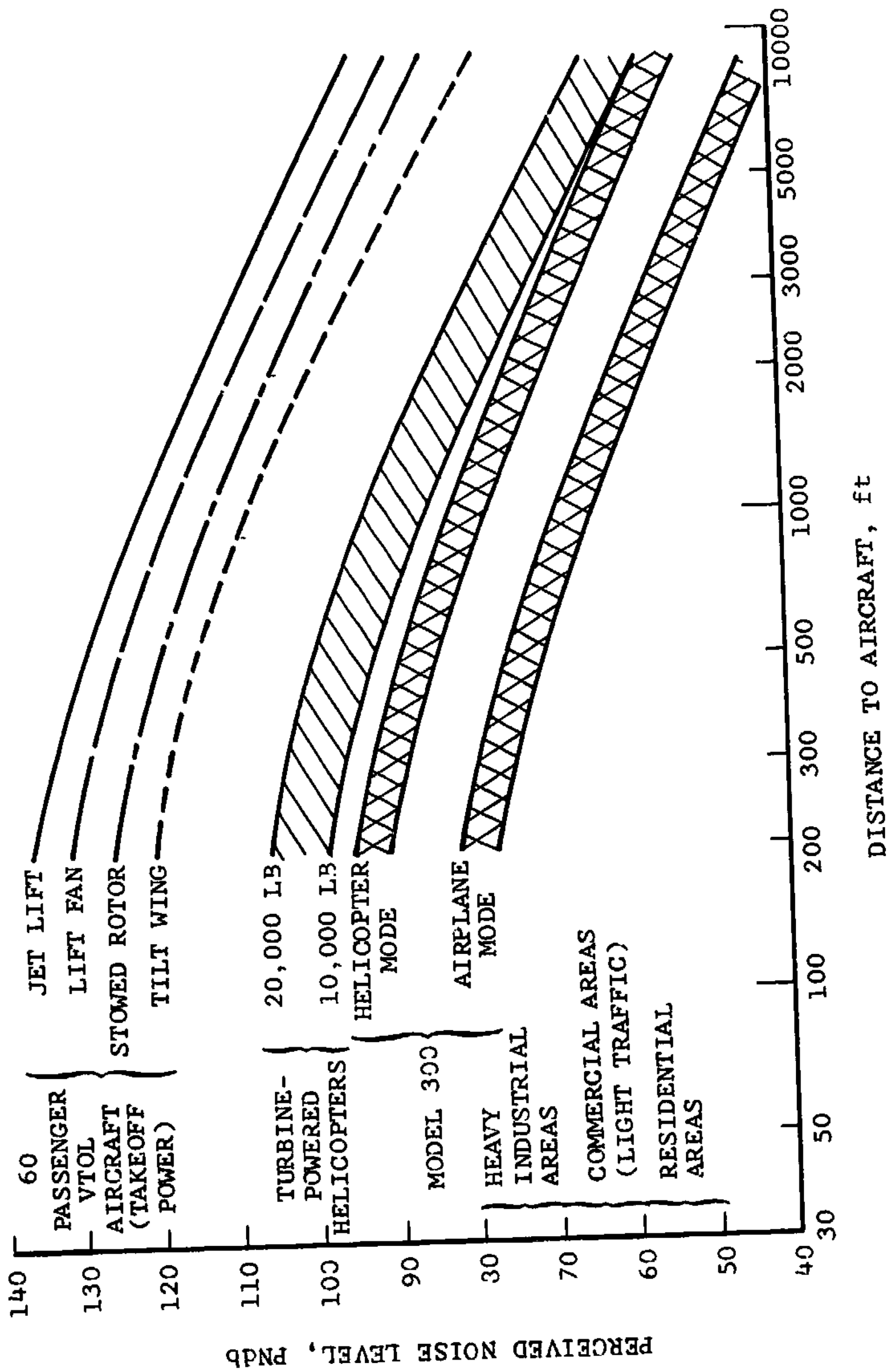


Figure VII-1. Calculated Noise Level of Model 300

VIII. RECOMMENDED PROOF-OF-CONCEPT PROGRAM

A. Objective and Purpose

The purpose of the tilt-proprotor proof-of-concept aircraft program is to extend and evaluate technology for this promising concept and to determine its suitability for future development and service. The ultimate objective is to enable industry, military and civil planners to proceed in a straightforward manner to develop operational aircraft and VTOL transportation systems.

The technical results from the program will establish that the concept is technically feasible and that technology is adequate for the successful development of operational aircraft. In addition to flight research to obtain technical data, tests will be conducted to evaluate economic feasibility, battlefield acceptability and social acceptance for a variety of civil and military missions. Measured performance on simulated missions will establish economic feasibility for the concept and will provide basic data to establish the cost effectiveness of this type of aircraft in a VTOL transportation system. Measurement of environmental effects on the aircraft and the aircraft's safety, noise and other characteristics will determine its acceptance to society and its ability to operate successfully in differing military roles and battlefield conditions.

B. Program Plan

The recommended program plan for the tilt-proprotor aircraft proof of concept is presented and discussed in the following paragraphs. This plan is based on the successful completion of the NASA tilt-proprotor technology program and assumes that the resulting technology will be available for the design of the proof-of-concept aircraft. It is also assumed that the proof-of-concept aircraft would be the same or very similar to the design presented in this report.

This same plan would be applicable for a folding proprotor aircraft. The design study for a folding proprotor proof-of-concept aircraft is in progress at BHC under a separate NASA contract, NAS 2-5461. The basis of that study is to utilize the Model 300 tilt-proprotor aircraft airframe with modifications as required to install the stop-fold provisions and the cruise propulsion system. Because these concepts are closely related, these programs could be combined. This is discussed in more detail under program schedule.

The program for the tilt-proprotor can logically be divided into three distinctive elements of work and/or phases. These are:

- Phase I - Design and Fabrication of Aircraft
- Phase II - Flightworthiness Tests
- Phase III - Proof-of-Concept Flight Research

The tasks to be covered in each phase are outlined below.

1. Phase I - Design and Fabrication

a. Design

The aircraft design approach will utilize the latest available technology in terms of the concept, materials, design, systems, engines, etc. However, items not fully developed or proven will be avoided as it is not the purpose here to develop supporting technologies. For instance, the development of engines or new material applications is not intended. The design will be reasonably refined so that the resulting aircraft will be representative of the state of the art. Close weight control will be maintained in the design stages so that the resulting empty weight to gross weight ratio of the proof-of-concept aircraft will be indicative of useful load ratios for future operational aircraft. The aircraft design criteria will be selected to permit the aircraft to explore safely the extremes of the aircraft's performance envelope. In addition, design maneuver and dive conditions should permit thorough flight testing to investigate the technical problem areas of the concept.

b. Fabrication

Fabrication of the aircraft, test items, and spares will be done with prototype or soft type tooling except for components such as blades and transmissions where hard tooling will be required to maintain quality. Quantities of components to be manufactured will be established to satisfy the test requirements of Phase II and III and to provide adequate spares to support these programs.

2. Phase II - Flightworthiness Tests

a. Component Qualification Tests

Critical components which are not off the shelf and have been designed or repackaged for the proof-of-concept aircraft will undergo tests to qualify them for flight research. This will include functional tests, proof tests, load-cycling tests, life tests, etc. for such components as conversion, flap and control phasing actuators, hydraulic boost cylinders and ejection seat installation. Drive system bench tests will include engineering development testing as well as green running of the components for the flightworthiness and flight-test articles.

b. Fatigue Tests

A minimum amount of fatigue testing will be conducted on critical blade, hub and proprotor control components to establish a safe life for at least 200 hours of flight testing. Typically, four specimens will be tested of each component. Two will be tested prior to flight based on estimated loads and the remaining two after loads have been measured in exploratory flight test. Fatigue failures will be used to establish S-N curves and a safe test life.

c. Propulsion System Tests

Development and endurance testing will be accomplished in a ground test rig to qualify the propulsion and related systems for flight. This will include the proprotors, the drive system with interconnect, powerplant installations, conversion actuators, and the associated hydraulic and mechanical control systems for the proprotors and engines. This type of testing can be done with a flight article, but is sometimes done on an "iron bird". A combination of flight article and iron bird is recommended for this program. All components and systems will be designed and fabricated as flightworthy systems and installed in a flightworthy wing. The complete package of wing and systems will comprise a flightworthiness test article. The test article will be supported in a test stand which will permit operating the proprotors in propeller mode as well as in helicopter mode. After completing functional and operational checks of all systems, a 150-hour qualification run will be made. This will consist of running at specified combinations of power, rpm, mast angle and control setting. Some of the time will be with only one engine operative to qualify the interconnect drive system.

d. Full-Scale Wind-Tunnel Tests

Wind-tunnel tests will be conducted in the NASA-Ames Full-Scale Wind Tunnel to identify deficient areas and establish a safe flight envelope (within the speed range of the tunnel) thereby permitting a more rapid and safe flight test program in the helicopter, conversion and low airplane speed ranges. Rather than use the flight-test article, it is recommended that the flightworthiness test article be used for these tunnel tests. The same remote control and instrumentation systems used for the flightworthiness ground tests would be used in the tunnel tests. The hard points installed on the wing could be designed so that they would also serve to mount the test article in the tunnel. The flightworthiness article is shown in Figure VIII-1 as it would appear mounted in the 40-by-80-foot wind tunnel.

Tunnel tests will be conducted to determine that the steady and oscillatory proprotor and control system loads are within design limits. Aircraft forces and moments and control positions will be recorded and analyzed to determine if the aircraft's flight characteristics are safe and that the conversion control phasing relationships designed into the control system are correct. Boundaries of the conversion corridor will be explored and established by testing at various combinations of wing angle of attack, conversion angle and power. Wing buffet the rotor-blade stall and oscillatory loads will be used to establish limitations and boundaries for safe conversion.

e. Aircraft Ground Tests

The flight aircraft will be thoroughly checked to determine proper functioning and operation of its electrical, hydraulic and mechanical systems. The control system will be rigged and proof loaded. A ground vibration frequency survey will be conducted to determine the frequency location of the major air-frame modes. If any frequencies are found to be significantly different from predictions, stability analyses will be redone to evaluate these differences and to determine if any corrective action is necessary.

The aircraft will be run on tiedown to further check all systems and to check the dynamic components. Before releasing the air-craft for flight, a minimum of fifteen hours of running time will be accomplished. This will include running at various combina-tions of conversion angle, control position and power settings.

f. Exploratory Flight Test

Exploratory flight tests will be conducted in a cautious manner and will be guided by the results of the wind-tunnel test pro-grams. Conversion would not be attempted prior to conversion tests in the tunnel. Expansion of the flight envelope beyond the wind-tunnel envelope will be done in a build-up manner by extrapolating load and stability data to the next speed increment prior to flight at that speed. Analysis of telemetered data will also be used to monitor each flight. Inflight shakers on the aircraft and appropriate controls will be utilized to determine damping of the proprotor, pylon, wing and empennage modes of vibration. Sufficient powerplant measurements will be taken to verify proper engine operation.

A preliminary flight loads survey will be conducted to permit completion of the fatigue test of the dynamic components and establishment of a safe component life for the flight-research program.

A safe nominal flight envelope in terms of altitude, gross weight, speed and load factor will be established. As necessary, adjust-ments or modifications will be made to the aircraft such that this envelope will be sufficiently large to demonstrate the performance and general characteristics of the aircraft. Upon completion of the exploratory flight-test program, the aircraft proprotors and transmission will be disassembled and inspected.

Parts will be replaced as required and the aircraft reassembled and readied for proof-of-concept flight research.

3. Phase III - Proof-of-Concept Flight Research

The test program is directed at establishing proof of concept in three areas; technical proof of concept, economic proof of concept, and social proof of concept. Dividing the test program into phases would not be required, but the effort will logically proceed with the emphasis being placed on the technical areas early in the program with emphasis switching later to tests directed at establishing economic feasibility and environmental acceptability. The following paragraphs briefly discuss the type of testing that is anticipated in the three proof-of-concept areas.

a. Technical Proof of Concept

The aircraft's performance, flight characteristics and problem areas will be investigated and evaluated to determine if technology is in hand for the successful development of an operational aircraft with desirable technical characteristics. Engineering flight tests will be conducted to document the aircraft's flight-handling characteristics, proprotor and airframe dynamic stability, airframe and proprotor loads and aircraft performance. The flight envelope as defined by the exploratory flight-test program will be expanded to define fully the aircraft's capability and limitations.

Evaluation of proprotor-pylon wing dynamic stability and the effects of compressibility on proprotor performance will be extended to the vicinity of 400 knots by using maximum power and diving the aircraft.

b. Economic Proof of Concept

Specific data will be recorded to establish mission effectiveness and economic feasibility. This would include data on payload/lift capability, single engine performance, hover fuel consumption, range and endurance. Selected mission profiles will be flown to demonstrate the capability of the aircraft to perform various civil and military missions. The flight-test data along with operational analysis inputs will be used to determine economic feasibility of the concept.

c. Environmental Proof of Concept

The desirability of using tilt-proprotor aircraft for particular civil and military missions will be established. The aircraft will be operated in and from various environments to determine environmental effects on the aircraft and the aircrafts effects on its surroundings. Engine failure and other emergency situations will be simulated. These tests will provide data on flight safety and safety to ground personnel, internal and external noise levels, and the effects of downwash and recirculation on engine ingestion, visability and dust signatures. Approach and landing procedures will be evaluated for operation from airports and heliports as well as undeveloped areas to provide data for the development of navigational systems and to determine real estate requirements for future VTOL ports and to permit definition of slow-speed maneuver requirements for military aircraft.

The proof-of-concept program results will define the capabilities and characteristics of the tilt-proprotor aircraft and will enable realistic VTOL operational specifications and requirements to be prepared. Civil, military, and industry planners will then be able to move swiftly to develop the needed operational aircraft and VTOL transportation systems.

C. Program Schedule

A suggested schedule for the tilt-proprotor aircraft proof-of-concept program is shown on Figure VIII-2. The current NASA tilt-proprotor technology program is shown as the first item on the schedule. It is assumed that the next step, the design and fabrication of the aircraft and test articles, would commence on completion of the 25-foot proprotor full-scale wind-tunnel tests which are scheduled for the first half of CY 1970 in the technology program.

This schedule shows the flightworthiness test article complete at the end of 1971 and the flight research aircraft roll out in the second quarter of CY 1972. Exploratory flight test would commence while the flightworthiness test article was in the full-scale tunnel. The flight program would be paced by the progress in the tunnel. The first in-flight conversion should occur in the third quarter of 1972 which would permit the proof-of-concept flight research program to start early in CY 1973.

The same program plan and type of schedule would also be appropriate for a folding proprotor proof-of-concept aircraft program. The schedule would be essentially the same with events in the folding-proprotor aircraft program occurring twelve months later than the schedule shown for the tilt-proprotor program. The phasing of the two programs could be arranged to permit the tilt-proprotor flightworthiness test article to be modified for flightworthiness and wind-tunnel tests of the folding-proprotor components.

An alternate program plan and schedule is shown on Figure VIII-3 This plan combines the tilt-proprotor and the folding-proprotor programs into one proprotor proof-of-concept program. This should make the overall program more economical than the two separate programs and permit the proof-of-concept flight research to be conducted simultaneously with the two research aircraft. Since the single flightworthiness article must now be capable of testing both types of proprotor systems without a long delay for modifications, the alternate schedule shows the design and fabrication of the flightworthiness article as the first step with the design and fabrication of the ships following. This phasing should permit the first conversion to occur at the end of CY 1972 and the first stop-fold to follow three months later. Proof-of-concept flight research would start in the middle of CY 1973.

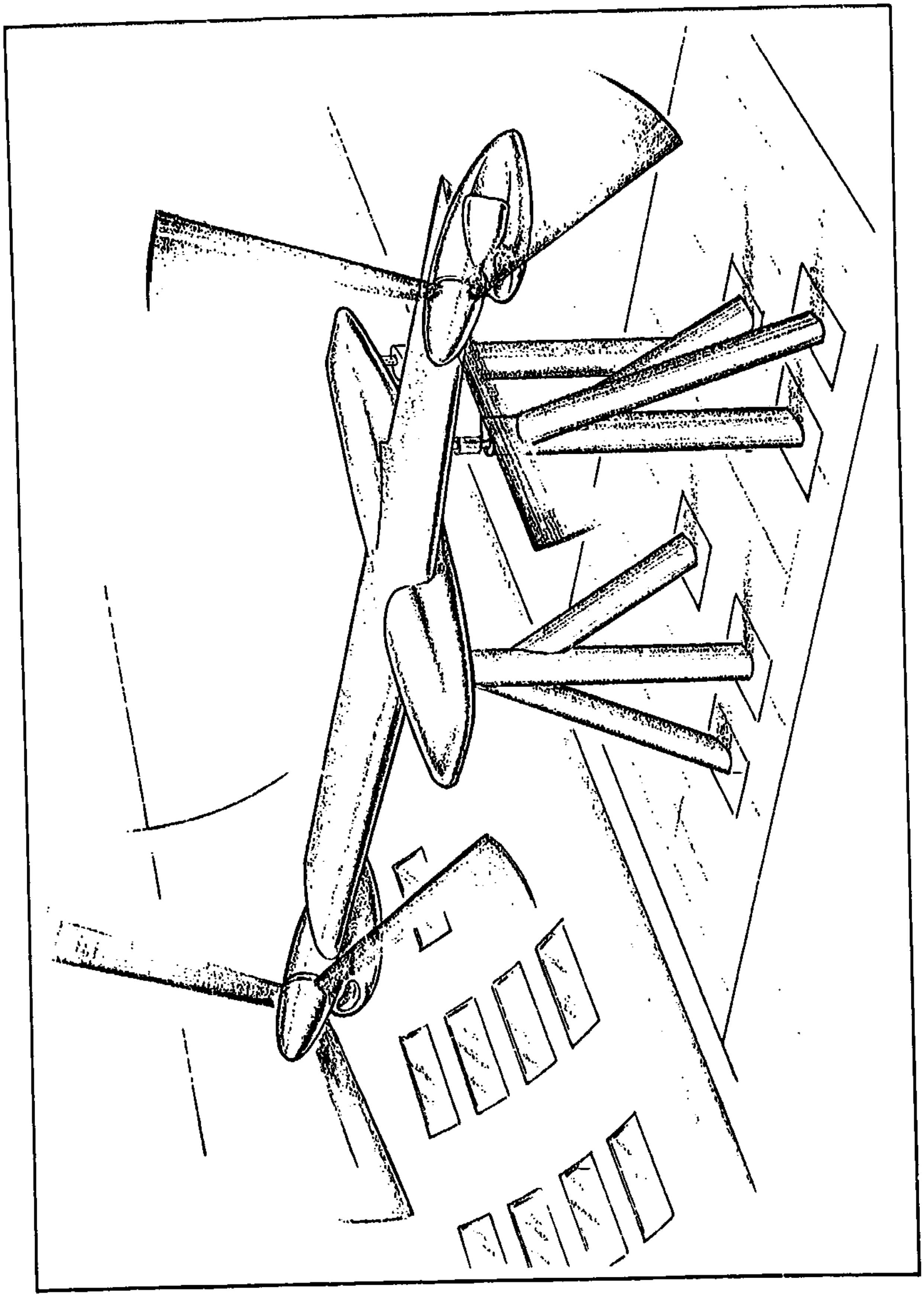


Figure VIII-1. Flightworthiness Test Article in the Ames 40-by-80-Foot Tunnel.

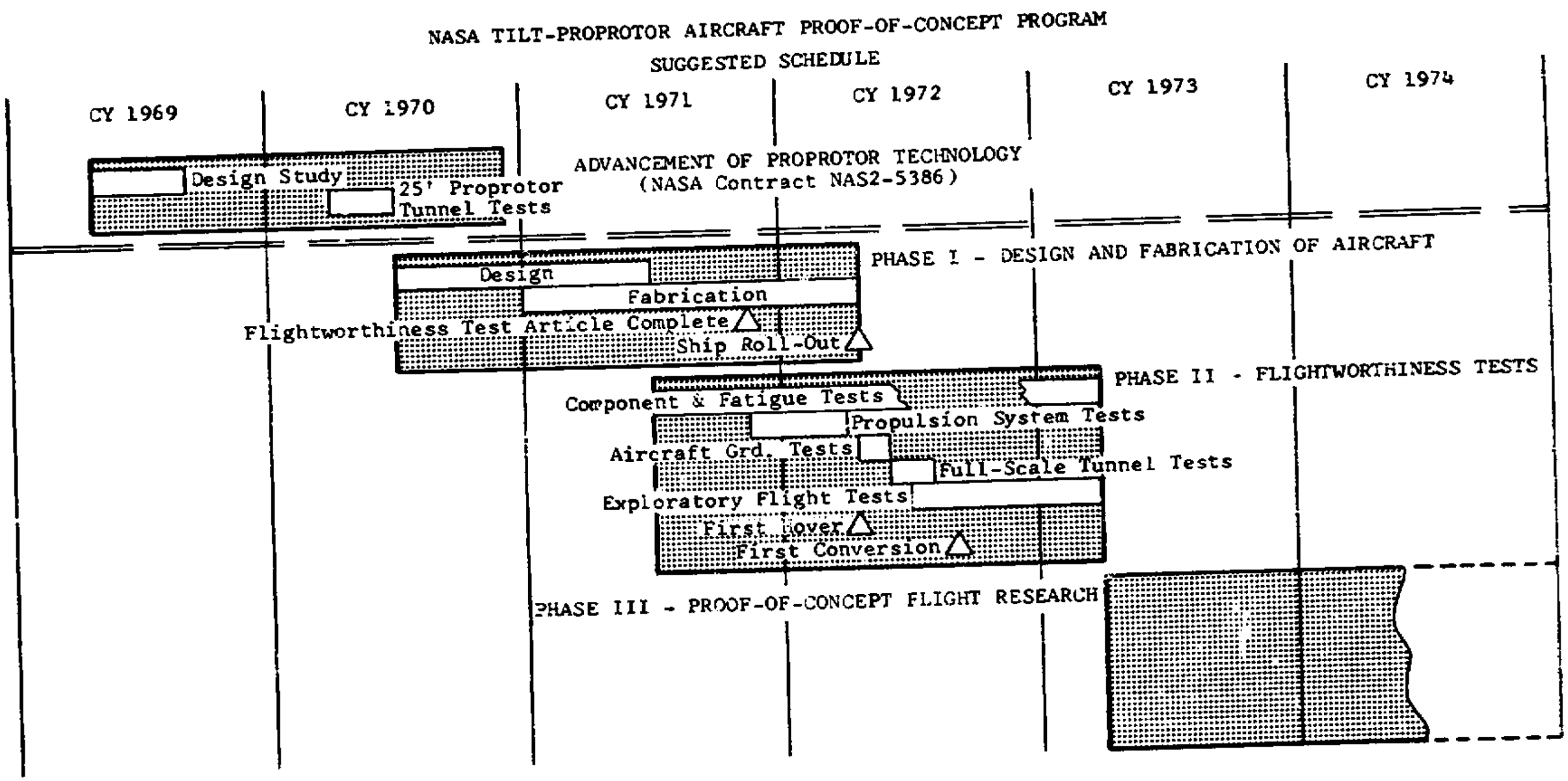


Figure VIII-2. Proof-of-Concept Program - Suggested Schedule

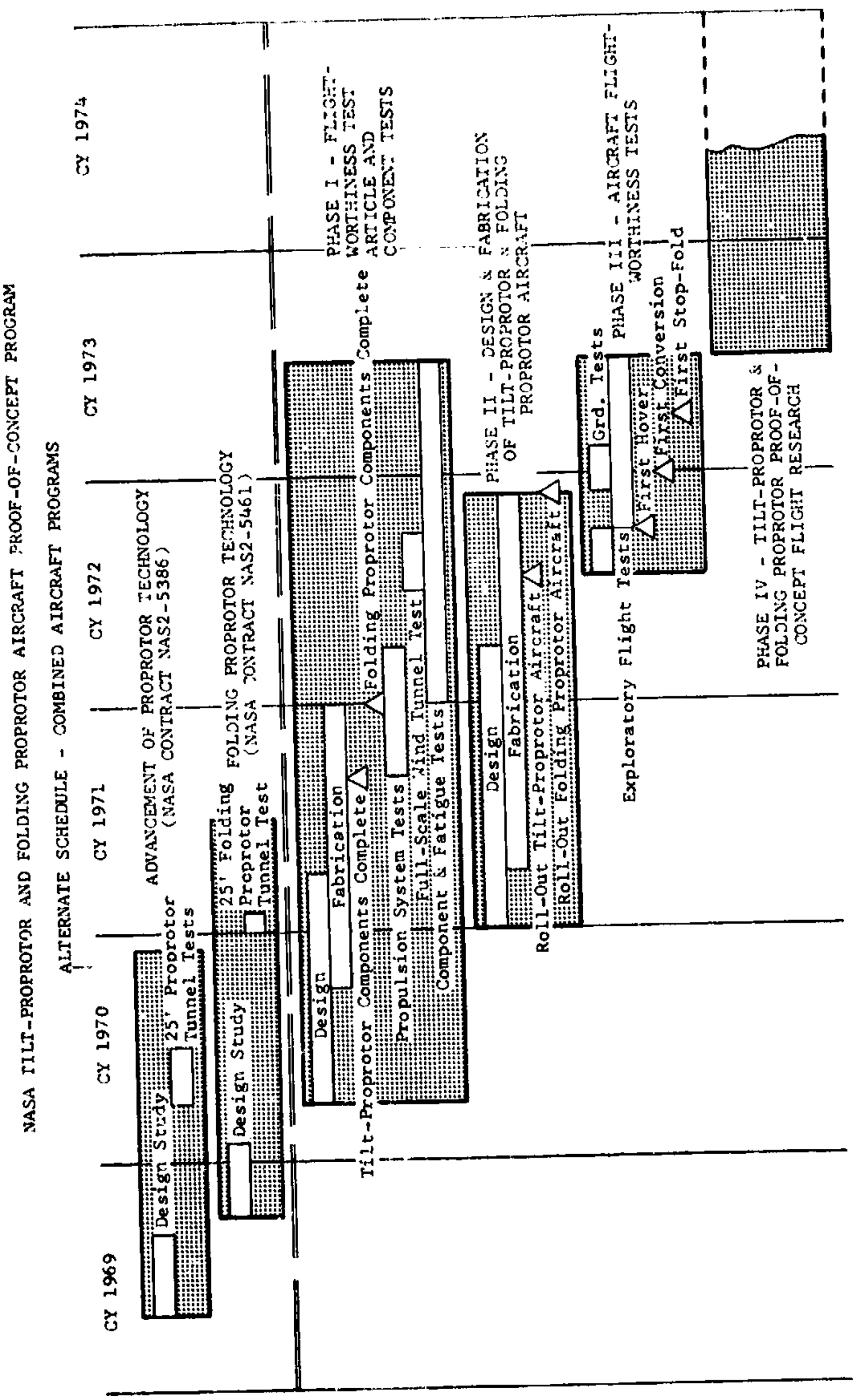


Figure VIII-3. Proof-of-Concept Program - Alternate Schedule

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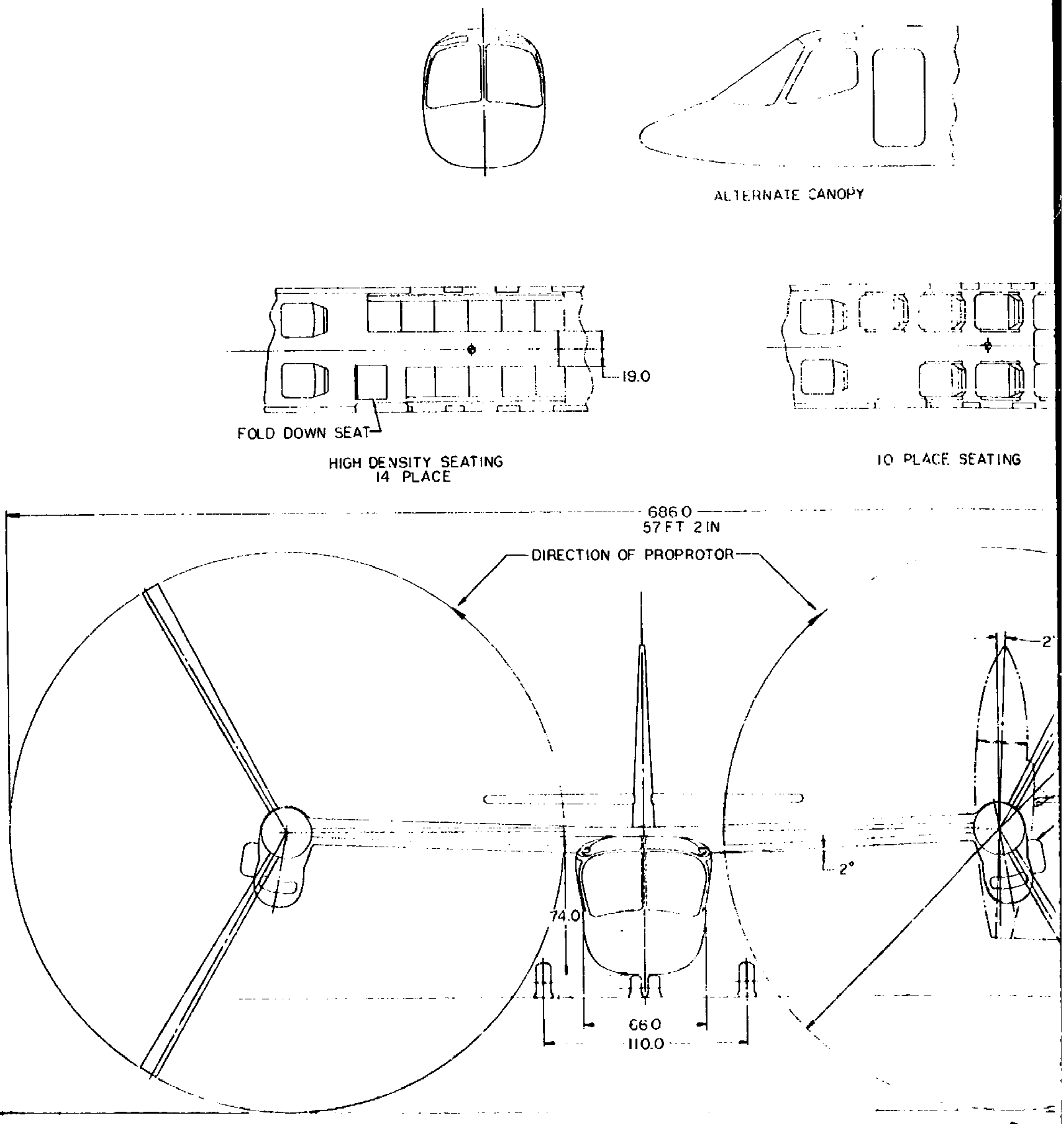
X. DESIGN LAYOUTS


ALTERNATE CANOPY
19.0
FOLD DOWN SEAT
HIGH DENSITY SEATING
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57 FT 2 IN
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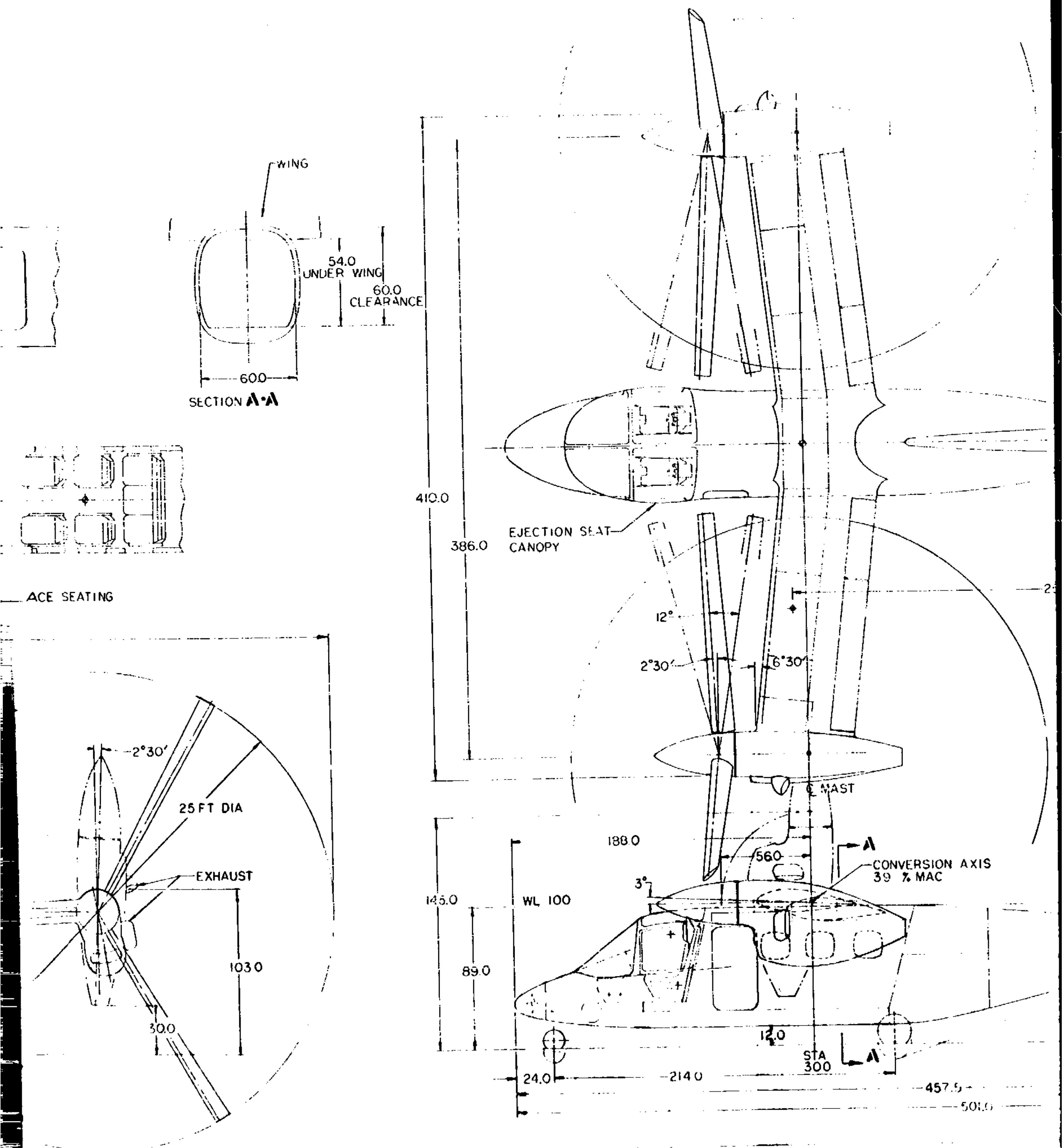
110.0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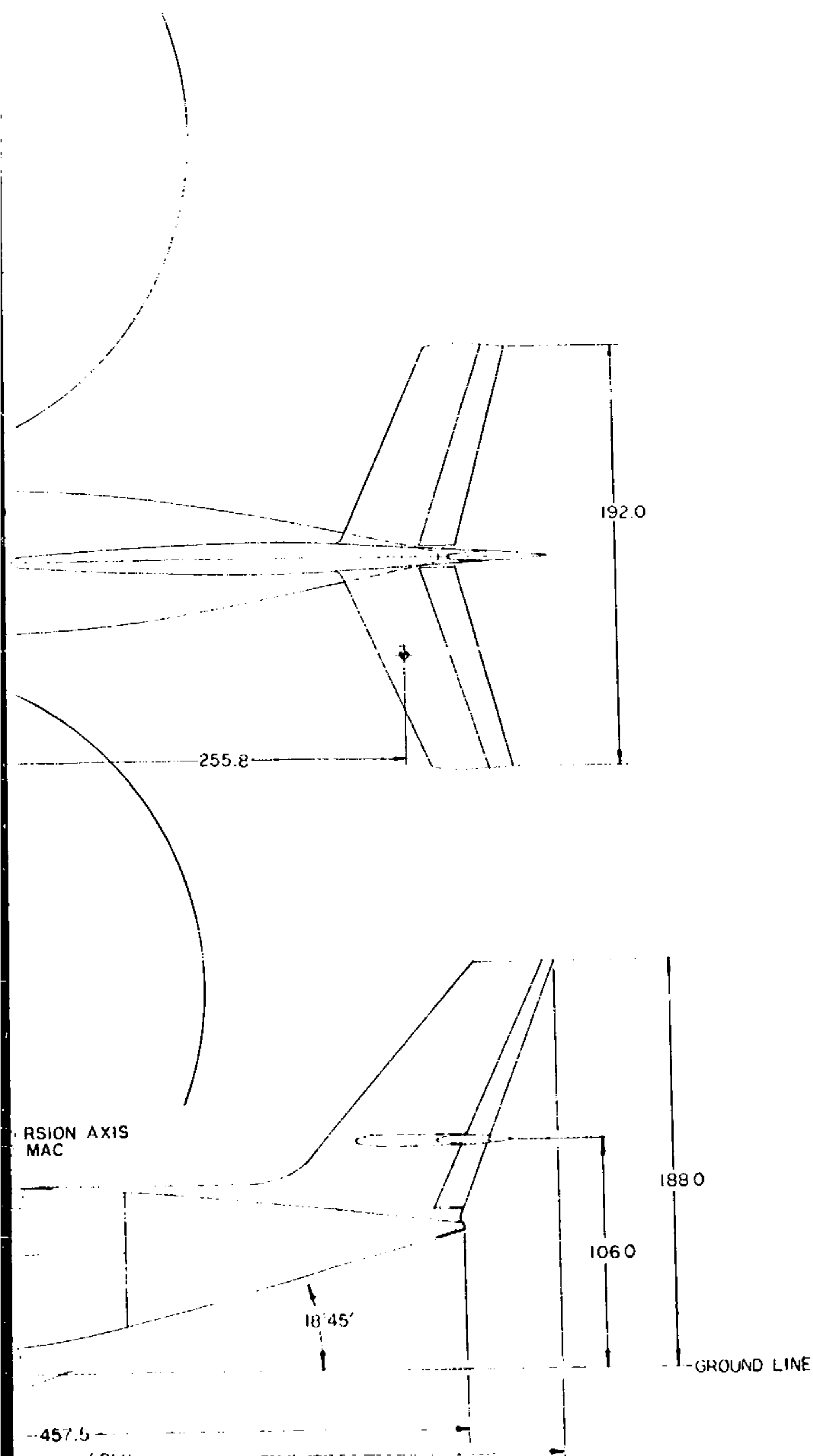
SECTION A-A
410.0
386.0
EJECTION SEAT
CANOPY
12°
2°30'
6°30'
25 FT DIA
EXHAUST
2°30'
103.0
30.0
MAST
188.0
56.0
A
CONVERSION AXIS
39 % MAC
3°
145.0
WL 100
89.0
12.0
STA
300
A
24.0
214.0
457.5
501.0

FOLDOUT



CHARACTERISTICS

WEIGHTS		
NORMAL GROSS	(RESEARCH MISSION)	95
EMPTY		68
EJECTION SEAT INCREASE		1
ENGINE OIL		4
CREW		16
FUEL		4
TEST EQUIPMENT		
MAXIMUM GROSS		124
POWER PLANT		
MANUFACTURER & MODEL	PRATT WHITNEY	PT
MAX CONT POWER	(2×995)	19
30 MINUTE POWER	(2×1150)	23
POWER LOADING	(30 MINUTE)	4
ROTOR		
DIAMETER		25
DISC AREA/ROTOR		49
DISC LOADING	(NORMAL GROSS WEIGHT)	9E
BLADE AIRFOIL	THEORETICAL ROOT	NA
	TIP	NA
BLADE CHORD		14
SOLIDITY		.0
BLADE TWIST EFFECTIVE		45
TIP SPEED	HELICOPTER MODE	7
	AIRPLANE MODE	6(
WING		
SPAN		34
AREA		17
WING LOADING	(NORMAL GROSS WEIGHT)	5
ASPECT RATIO		6
AIRFOIL	TIP/ROOT	N
FLAP AREA/SIDE	AFT OF HINGE	5
AILERON AREA/SIDE	AFT OF HINGE	1(
EMPENNAGE		
HORIZONTAL TAIL	AREA	6
	ASPECT RATIO	4
ELEVATOR AREA TOTAL	AFT OF HINGE	1
VERTICAL TAIL	AREA	5
	ASPECT RATIO	1
RUDDER AREA	AFT OF HINGE	

0 INCHES SCALE 100

CHARACTERISTICS

WEIGHTS		
NORMAL GROSS	(RESEARCH MISSION)	9500 LB
EMPTY		6876 LB
EJECTION SEAT INCREASE		114 LB
ENGINE OIL		35 LB
CREW		400 LB
FUEL		1600 LB
TEST EQUIPMENT		475 LB
MAXIMUM GROSS		12400 LB
POWER PLANT		
MANUFACTURER & MODEL	PRATT WHITNEY	PT6C-40
MAX CONT POWER	(2×995)	1990 SHP
30 MINUTE POWER	(2×1150)	2300 SHP
POWER LOADING	(30 MINUTE)	4.13 LB/HP
ROTOR		
DIAMETER		25 FT
DISC AREA/ROTOR		491 SQ FT
DISC LOADING	(NORMAL GROSS WEIGHT)	9.67 LB/SQ FT
BLADE AIRFOIL	THEORETICAL ROOT	NACA 64-935 a=0.3
	TIP	NACA 64-208 a=0.3
BLADE CHORD		14 IN
SOLIDITY		.089
BLADE TWIST-EFFECTIVE		45 DEG
TIP SPEED	HELICOPTER MODE	740 FT/SEC
	AIRPLANE MODE	600 FT/SEC
WING		
SPAN		34.3 FT
AREA		176 SQ FT
WING LOADING	(NORMAL GROSS WEIGHT)	54.0 LB/SQ FT
ASPECT RATIO		6.63
AIRFOIL	TIP & ROOT	NACA 64A223 MOD
FLAP AREA/SIDE	AFT OF HINGE	5.5 SQ FT
AILERON AREA/SIDE	AFT OF HINGE	10.4 SQ FT
EMPENNAGE		
HORIZONTAL TAIL	AREA	62.5 SQ FT
	ASPECT RATIO	4.1
ELEVATOR AREA TOTAL	AFT OF HINGE	17.6 SQ FT
VERTICAL TAIL	AREA	57.8 SQ FT
	ASPECT RATIO	1.84
RUDDER AREA	AFT OF HINGE	7.6 SQ FT

192.0

188.0

106.0

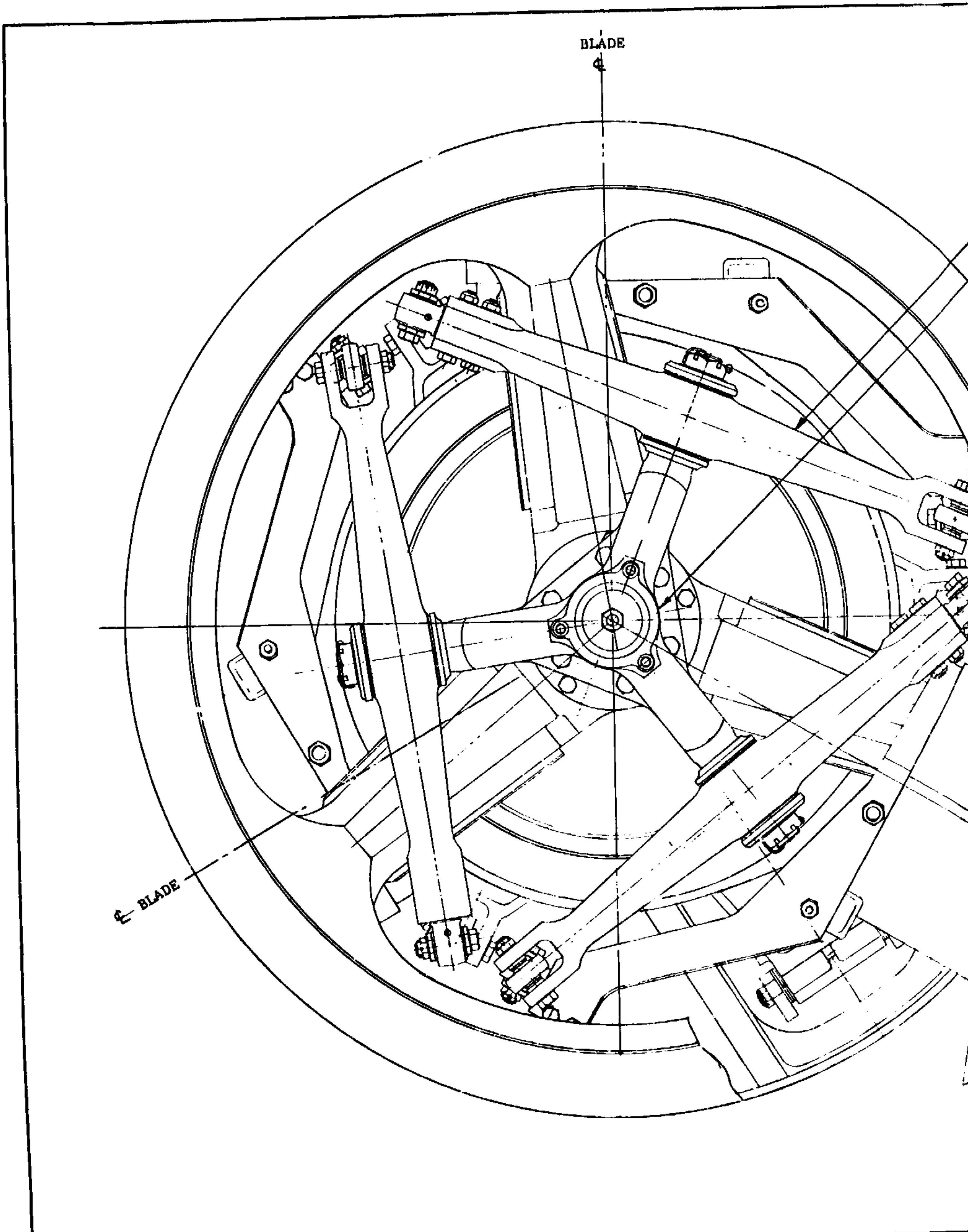
GROUND LINE

0 INCHES 100
SCALE

DESIGN LAYOUT

TITLE MODEL 300 TILTROTOR AIRCRAFT

G. SMITH | 8-63 | 1/20



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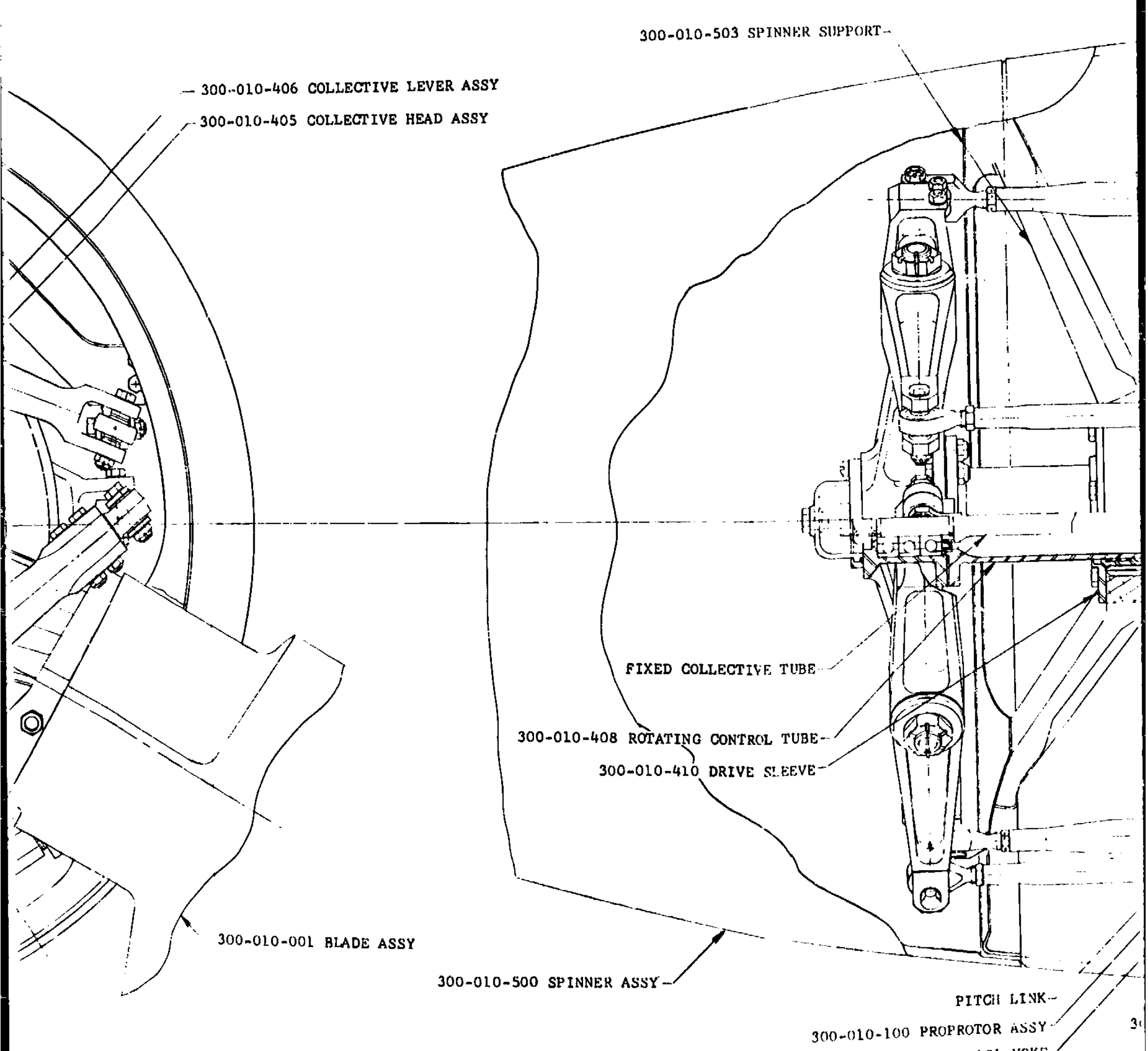
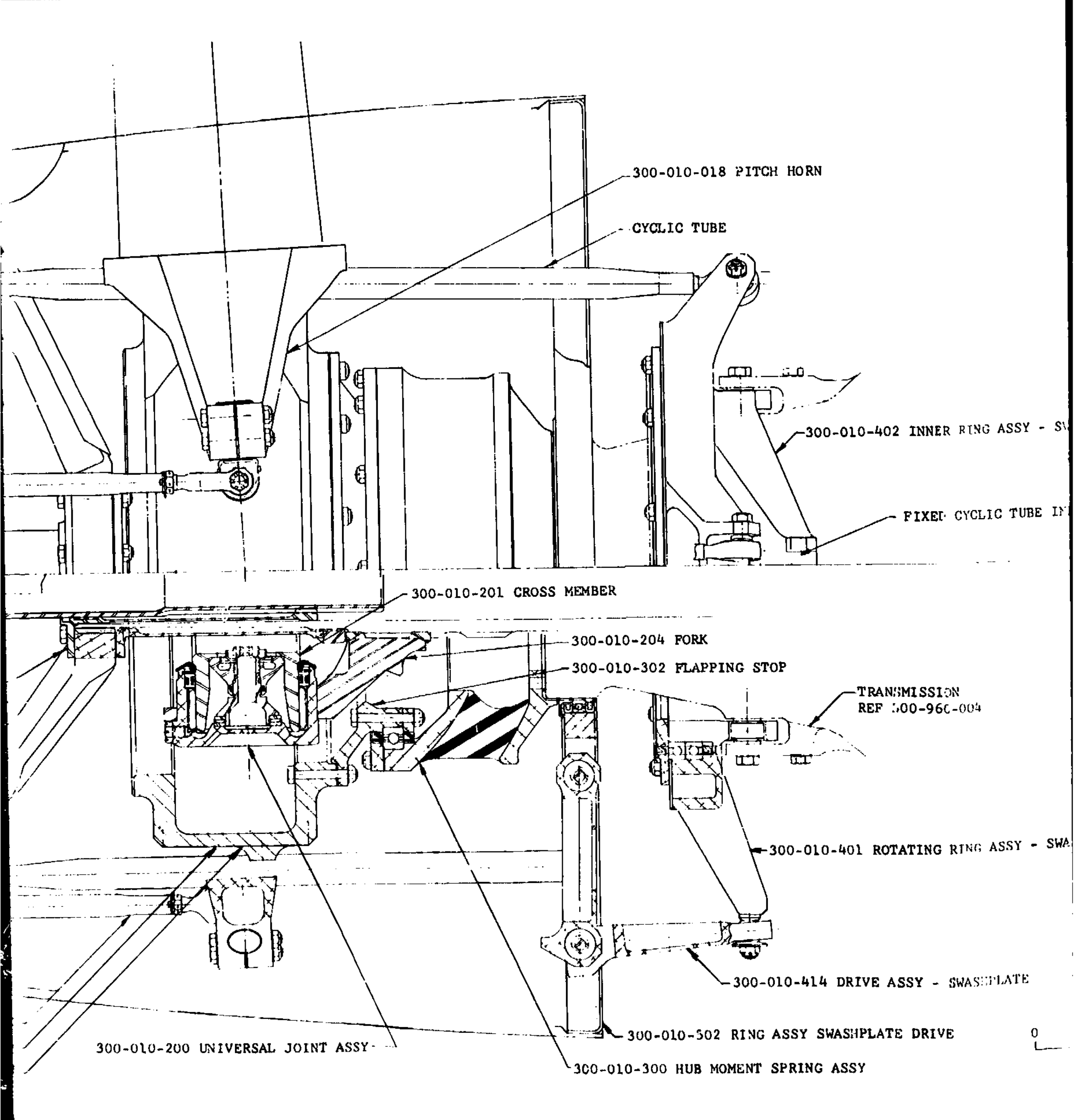
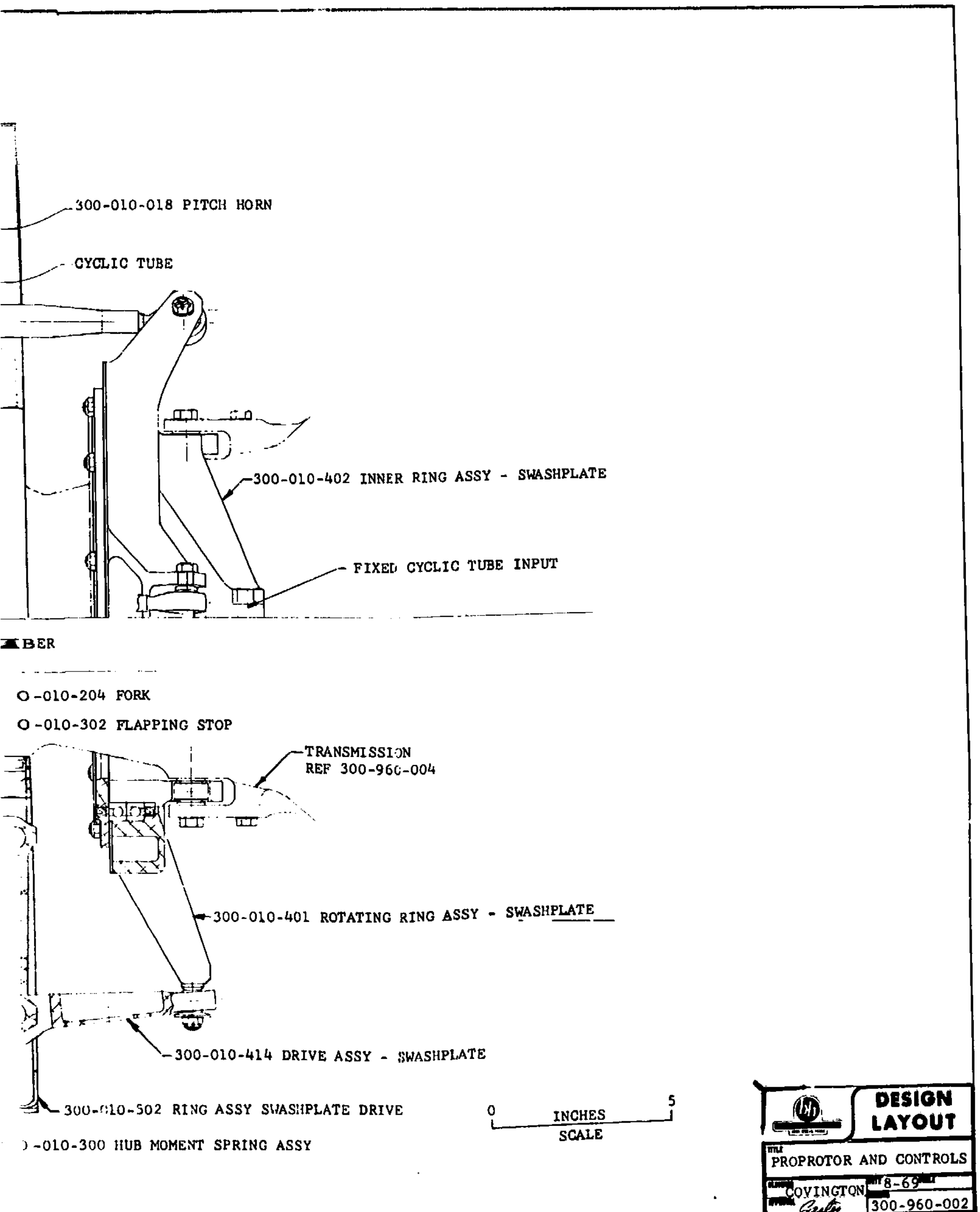

300-010-503 SPINNER SUPPORT
300-010-406 COLLECTIVE LEVER ASSY
300-010-405 COLLECTIVE HEAD ASSY
FIXED COLLECTIVE TUBE
300-010-408 ROTATING CONTROL TUBE
300-010-410 DRIVE SLEEVE
300-010-001 BLADE ASSY
300-010-500 SPINNER ASSY
PITCH LINK
300-010-100 PROPROTOR ASSY
300-010-101 YOKE



300-010-018 PITCH HORN
CYCLIC TUBE
300-010-402 INNER RING ASSY - S
FIXED CYCLIC TUBE
300-010-201 CROSS MEMBER
300-010-204 FORK
300-010-302 FLAPPING STOP
TRANSMISSION
REF 300-960-004
300-010-401 ROTATING RING ASSY - SWA
300-010-414 DRIVE ASSY - SWASHPLATE
300-010-302 RING ASSY SWASHPLATE DRIVE
300-010-200 UNIVERSAL JOINT ASSY
300-010-300 HUB MOMENT SPRING ASSY
0



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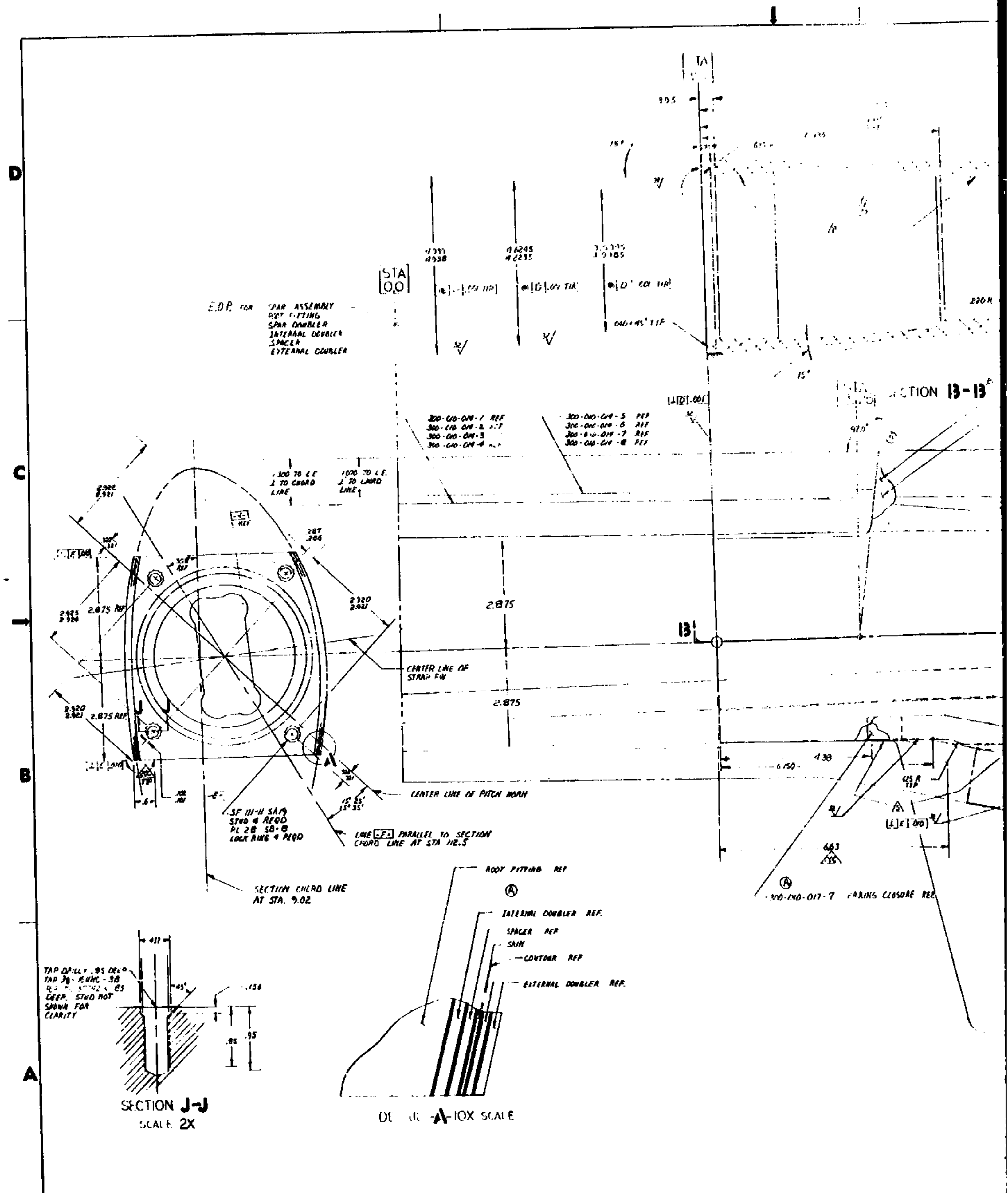
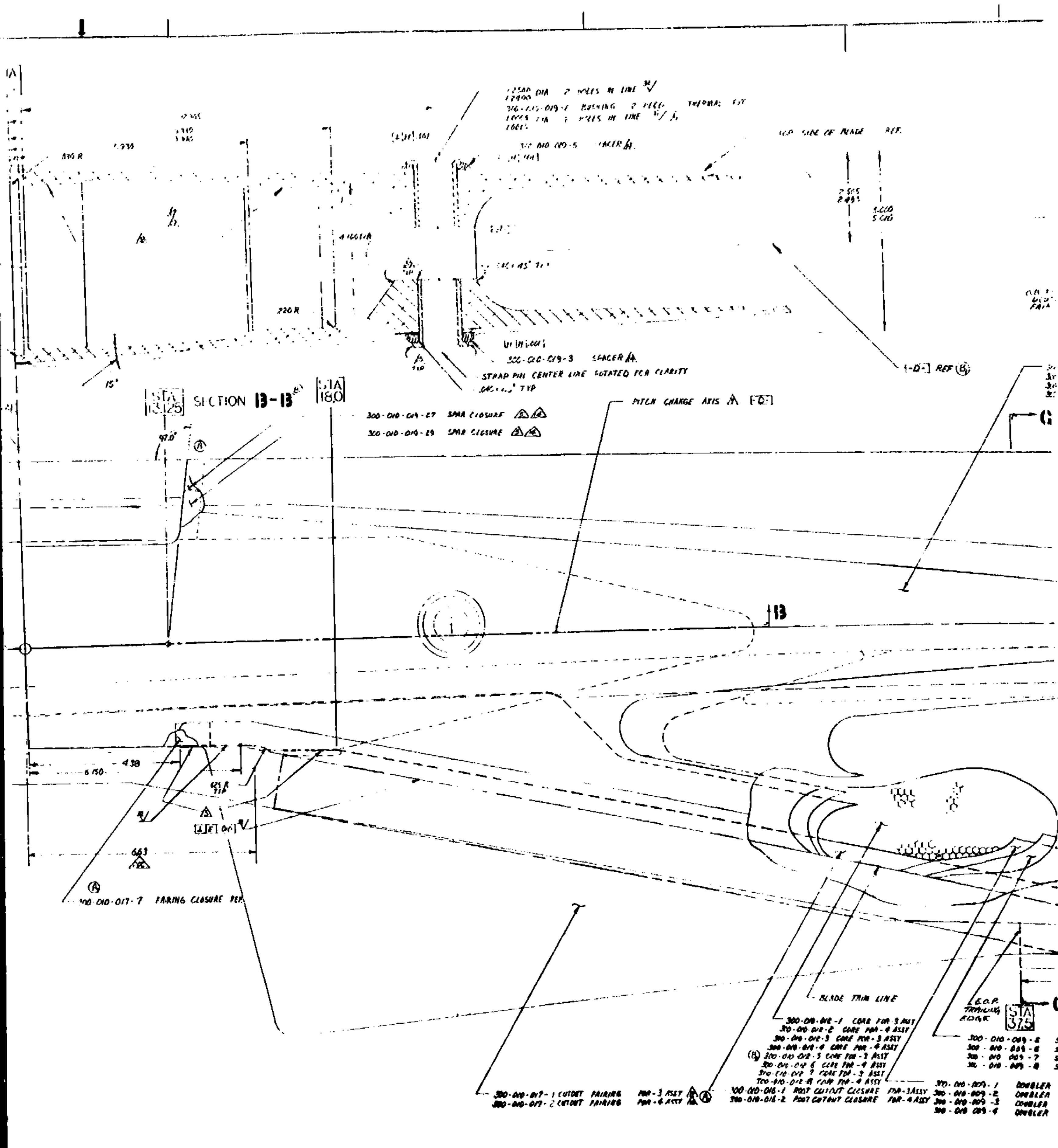
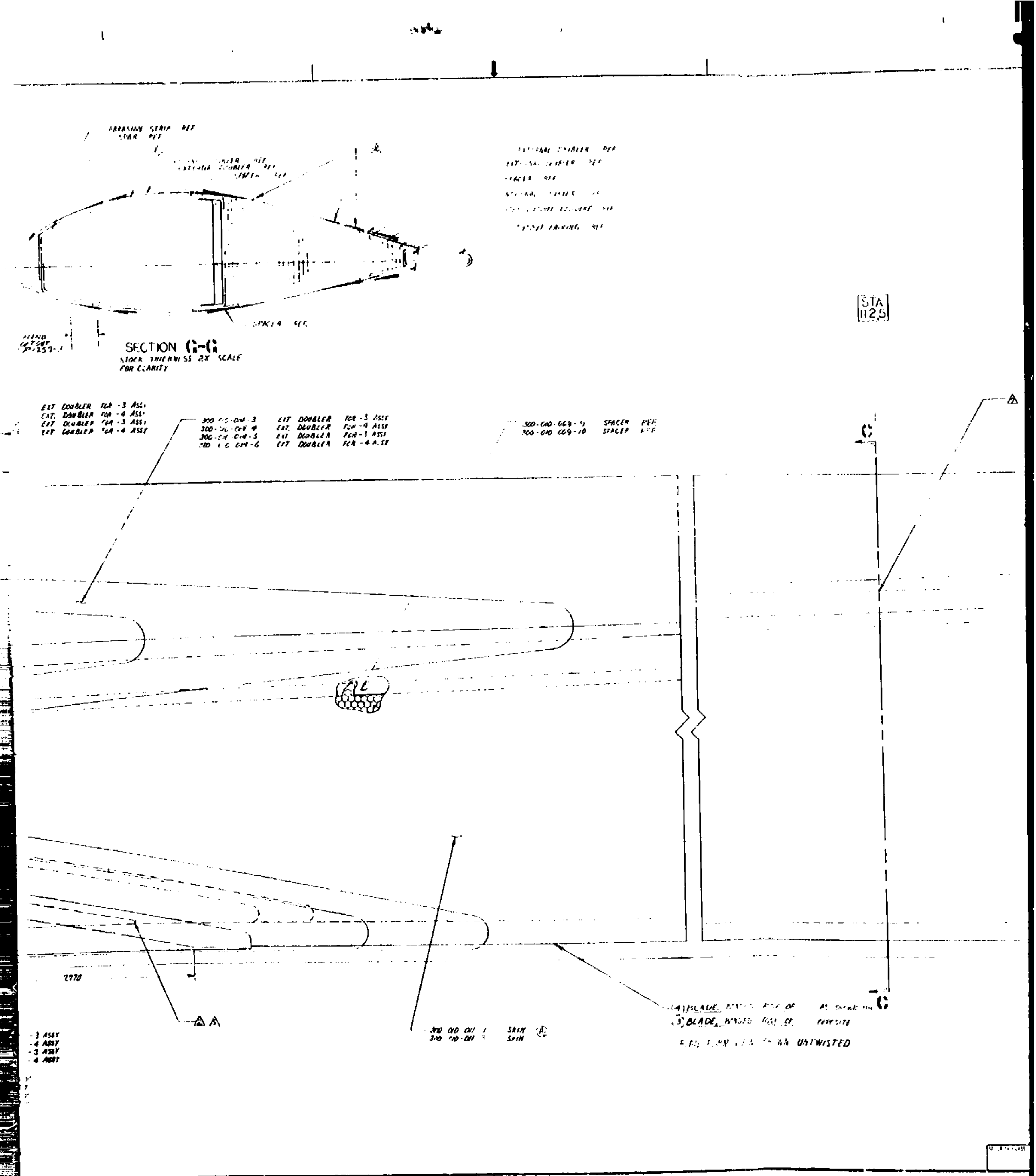

SECTION B-B
SECTION J-J
SCALE 2X
DETAIL -A- 10X SCALE
CENTER LINE OF STRAP FW
CENTER LINE OF PITCH HORN
SECTION CHORD LINE AT STA. 9.02
LINE -F- PARALLEL TO SECTION CHORD LINE AT STA 112.5
SF III-II SAM STUD 4 REQD PL 28 S8-8 LOCK RING 4 REQD
E.D.P. FOR SPAR ASSEMBLY ROOT FITTING SPAR DOUBLER INTERNAL DOUBLER SPACER EXTERNAL DOUBLER
ROOT FITTING REF.
INTERNAL DOUBLER REF.
SPACER REF
SKIN
CONTOUR REF
EXTERNAL DOUBLER REF.
300-010-017-7 FAIRING CLOSURE REF
TAP DRILL .95 DEEP TAP 3/8-16UNC-3B 85 DEEP. STUD NOT SHOWN FOR CLARITY
2.875
2.875 REF
STA 0.0
D
C
B
A



DWG NO. 300-010-001 SHEET 3

FOLDOUT FRAME



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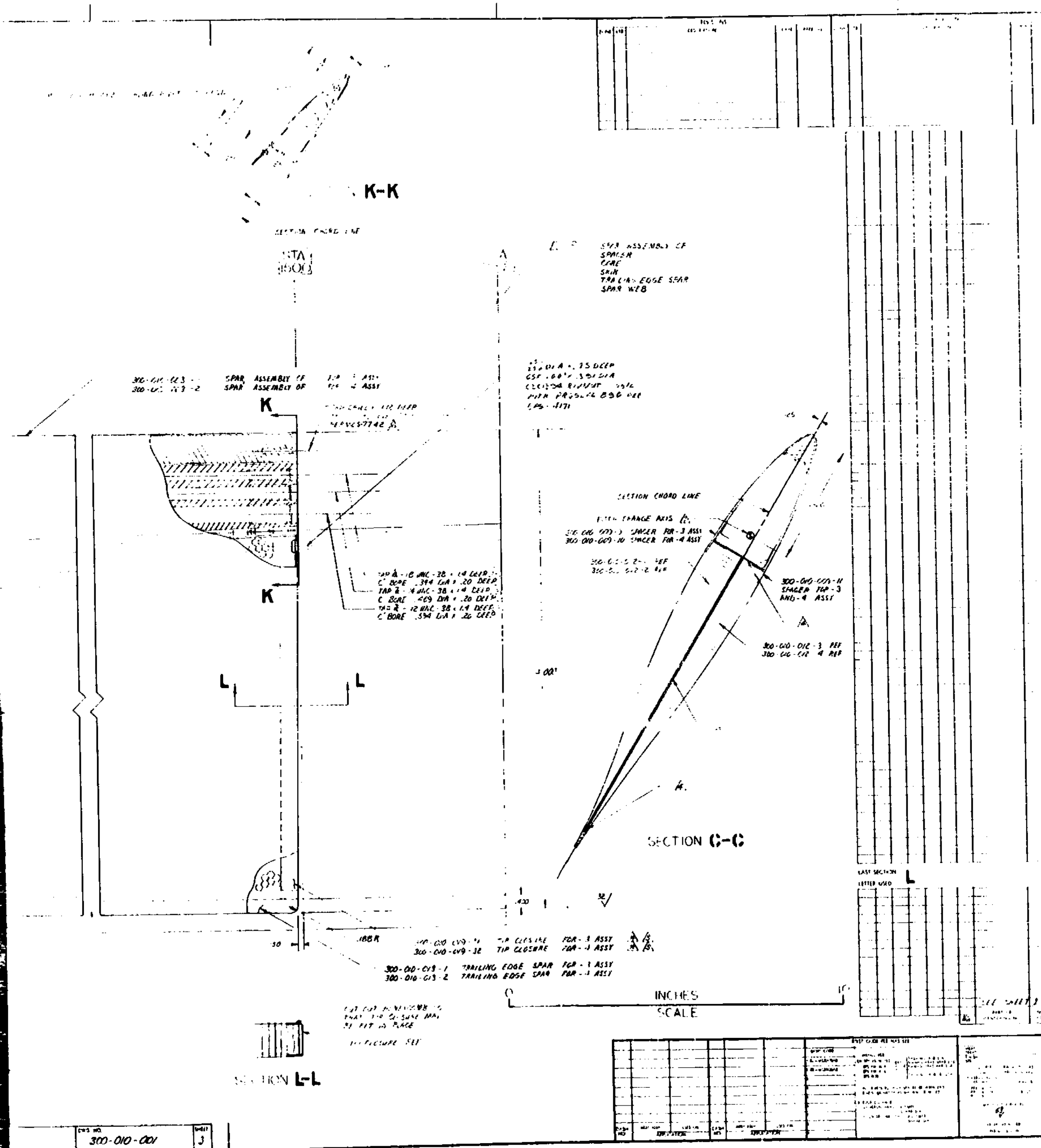
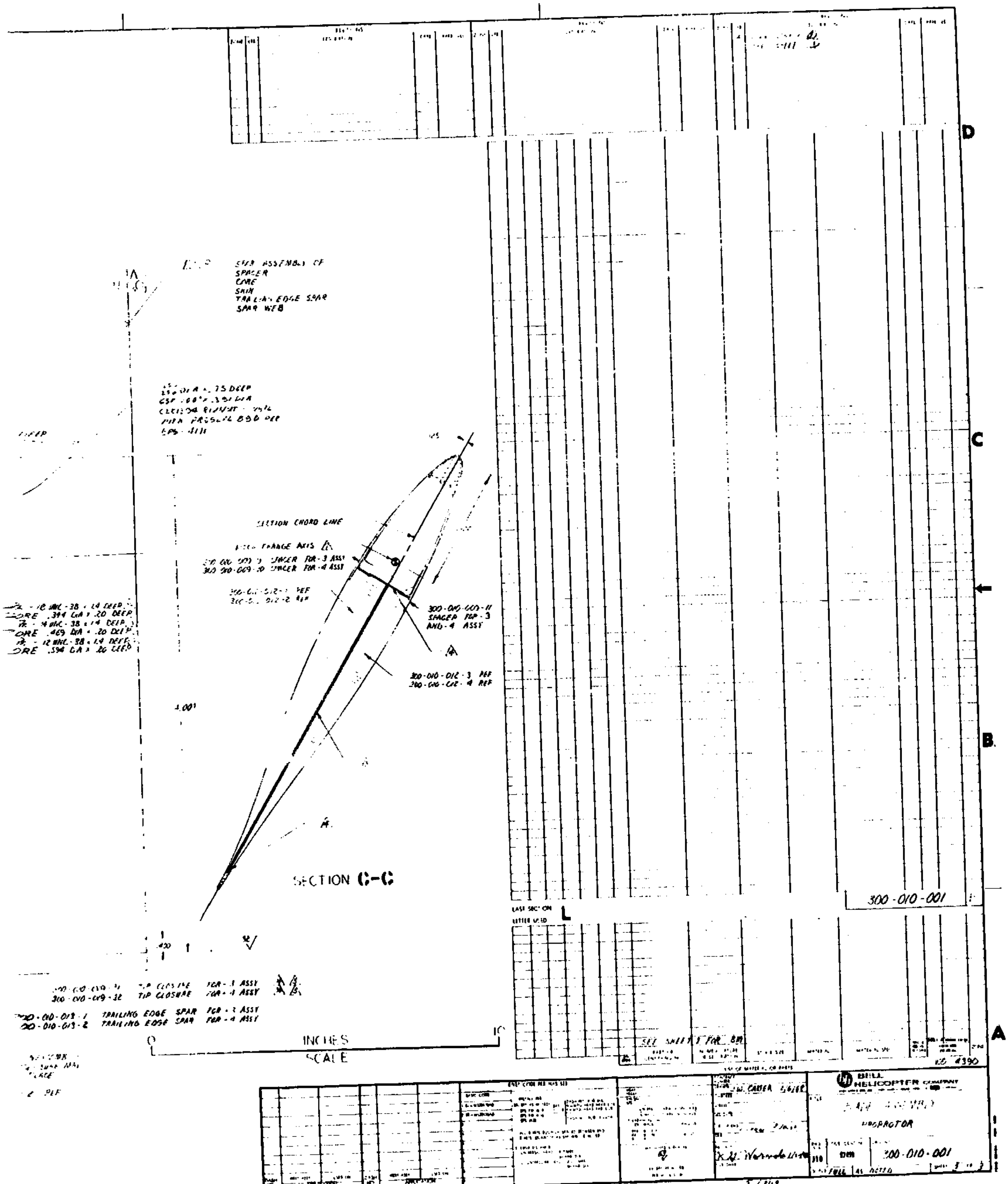
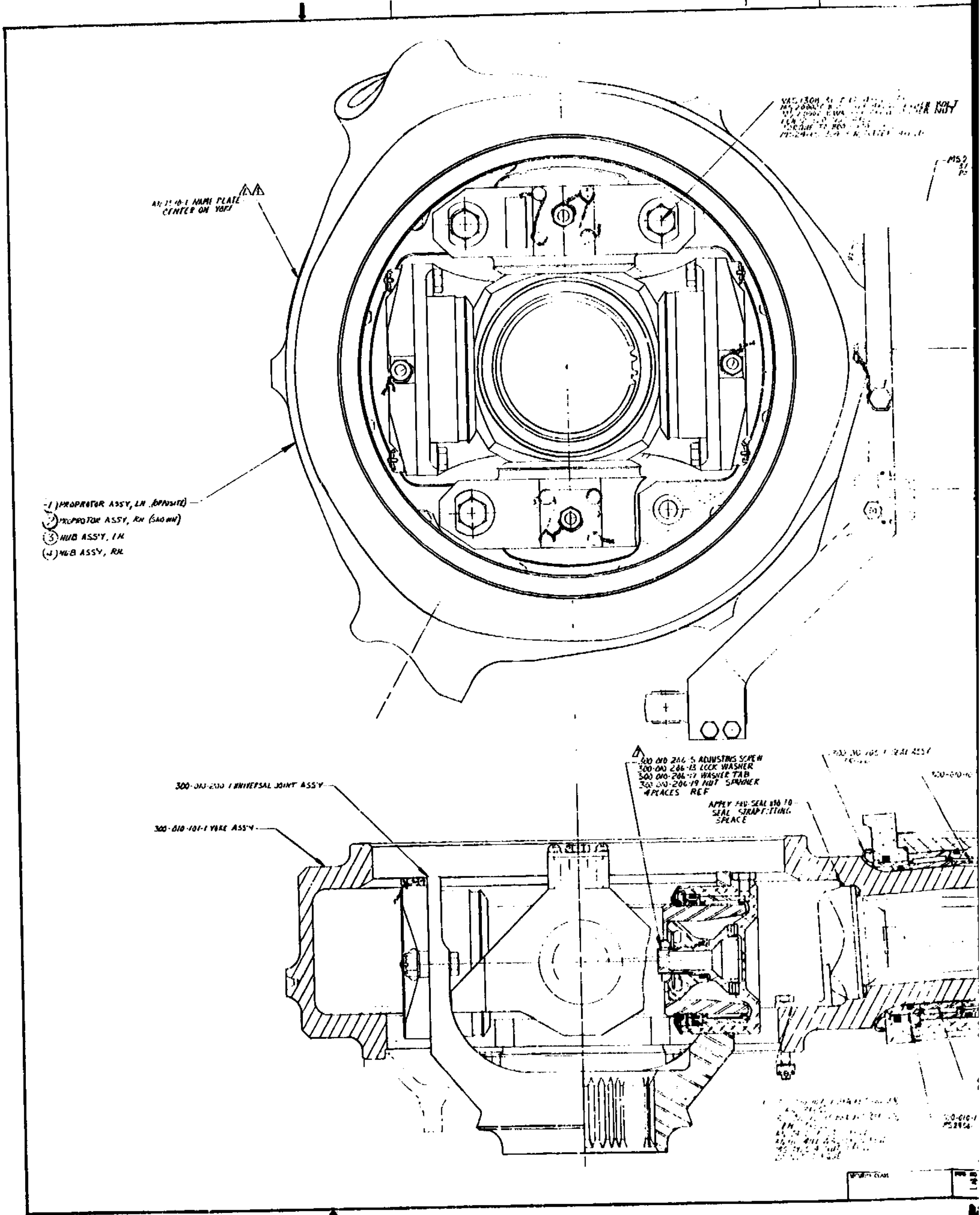


K-K
SPAR ASSEMBLY OF
SPACER
CORE
SKIN
TRAILING EDGE SPAR
SPAR WEB
SPAR ASSEMBLY OF
K
K
L
L
SECTION CHORD LINE
300-010-005-11
SPACER FOR -3
AND -4 ASSY
300-010-012-3 REF
300-010-012-4 REF
SECTION C-C
LAST SECTION LETTER USED L
300-010-019-1 TRAILING EDGE SPAR FOR -3 ASSY
300-010-019-2 TRAILING EDGE SPAR FOR -4 ASSY
TIP CLOSURE FOR -4 ASSY
INCHES
SCALE
SECTION L-L
DWG NO. 300-010-001
SHEET 3

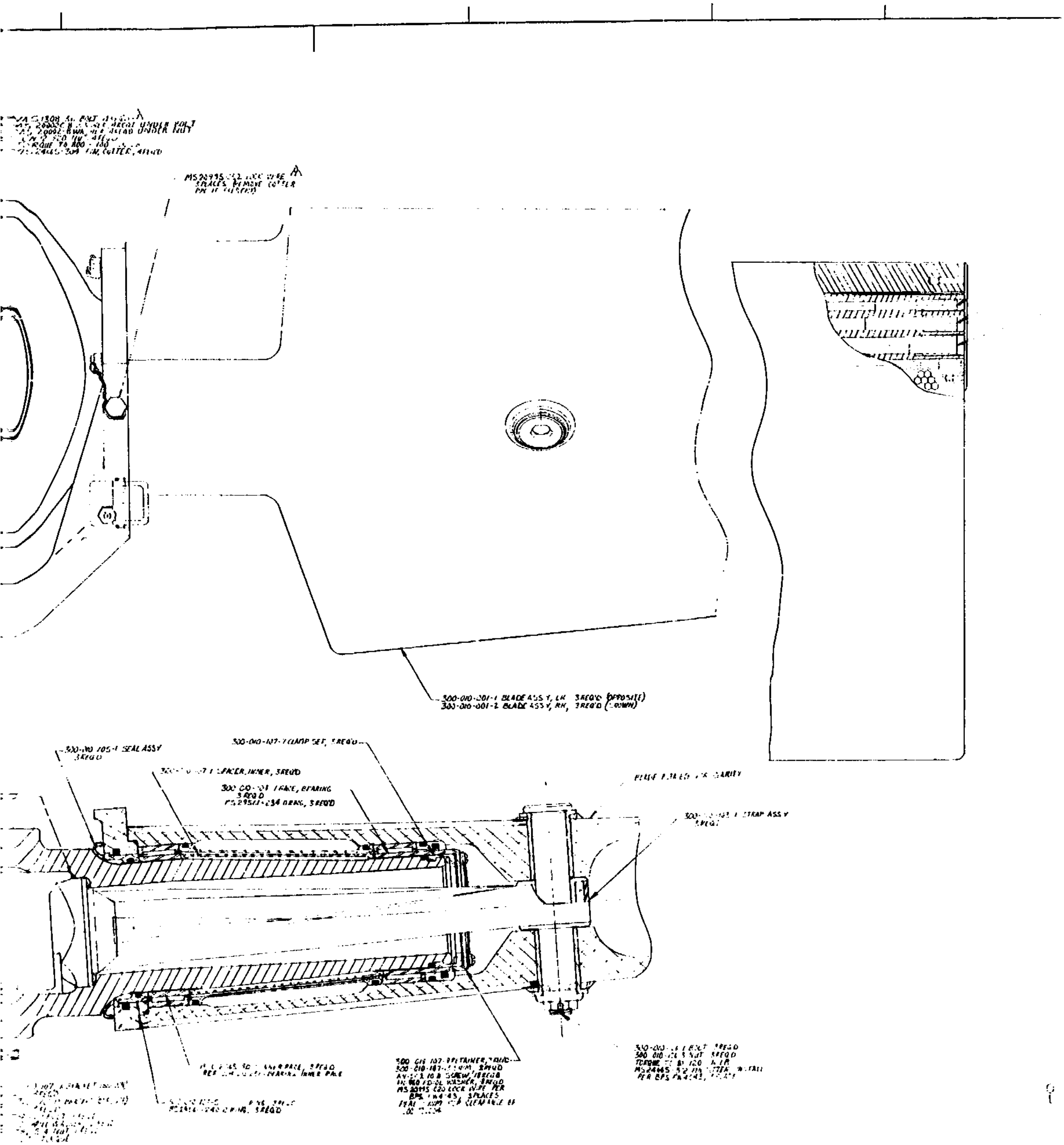


SPACER
CORE
SKIN
TRAILING EDGE SPAR
SPAR WEB
SECTION CHORD LINE
300-010-011-3 REF
300-010-012-4 REF
300-010-009-11
SPACER FOR-3
AND-4 ASSY
300-010-012-3 REF
300-010-012-4 REF
SECTION C-C
300-010-013-1 TRAILING EDGE SPAR FOR-3 ASSY
300-010-013-2 TRAILING EDGE SPAR FOR-4 ASSY
300-010-019-32 TIP CLOSURE FOR-4 ASSY
INCHES
SCALE
D
C
B
A
300-010-001
LAST SECTION
LETTER USED
L
SEE SHEET 1 FOR BM
BELL
HELICOPTER COMPANY
PROPROTOR
300-010-001
FULL
AS NOTED



NAME PLATE
CENTER ON YOKE
-1 PROPROTOR ASSY, LH (OPPOSITE)
-2 PROPROTOR ASSY, RH (SHOWN)
(3) HUB ASSY, LH
(4) HUB ASSY, RH
300-010-200 UNIVERSAL JOINT ASSY
300-010-101-1 YOKE ASSY
300-010-206-5 ADJUSTING SCREW
300-010-206-13 LOCK WASHER
300-010-206-17 WASHER TAB
300-010-206-19 NUT SPANNER
4 PLACES REF
APPLY PRO-SEAL 890 TO SEAL STRAP FITTING SPLACE

FOLDOUT FRA...



300-010-100

FOLDOUT

300-010-017-7 BALANCE WEIGHT
300-010-017-9 BALANCE WEIGHT
300-010-017-11 BALANCE WEIGHT
300-010-017-13 BALANCE WEIGHT
300-010-017-15 BALANCE WEIGHT
300-010-017-17 BALANCE WEIGHT

3	3			IR 65265-50-1	INNER RACE	
		AR	AR	EPON 934	ADHESIVE	
AR	AR			892	PRO SEAL	
4	4			LCN12 820	NUT	PURCHASE FROM SPS FASTNERS
		4	4	MS24665-304	PIN COTTER	
AR	AR			MS 20995 C	LOCK WIRE	
4	4			MS 20002 C 8	WASHER	
3	3			MS29513-240	O RING	
3	3			MS29513-234	O RING	
		3	3	MS 24665-372	PIN, COTTER	
6	6			MS17825-4	NUT	
4	4			MS20002-8	WASHER	
4	4			MS1308-36	BOLT	
		1	1	AN 759-1	NAME PLATE	
12	12			AN 960-416L	WASHER	
18	18			AN 960-PD10L	WASHER	
18	18			AN 502-10-8	SCREW	
6	6			AN 174-21	BOLT	
1	1			300-010-200-1	UNIVERSAL JOINT ASSY	
5				-107-16	BRACKET	
	3			-107-15	BRACKET LH	
3	3			-107-13	SHIM	
3	3			-107-5	RETAINER	
3	3			-107-7	CLAMP SET	
3	3			-107-5	RING	
3	3			-107-1	SPACER	
		3	3	-1-2	NUT	
		3	3	-106-1	BOLT ASSY	
3	3				[illegible] ASSY	
3	3				[illegible], CASING	

300-0

LAST SECTION

LETTER USED

2	2			-103-1	STRAP ASSY	
1	1			-102-1	SEAL ASSY	
		AR	AR	-017-16	BALANCE WEIGHT	
		AR	AR	-017-15	BALANCE WEIGHT	
		AR	AR	-017-13	BALANCE WEIGHT	
		AR	AR	-017-11	BALANCE WEIGHT	
		AR	AR	-017-9	BALANCE WEIGHT	
		AR	AR	-017-7	BALANCE WEIGHT	
		3		-001-2	BLADE ASSY, RH.	
			3	-001-1	BLADE ASSY, LH.	
		1		-100-4	HUB ASSY, RH	
			1	-100-3	HUB ASSY, LH	
				-100-2	PROPROTOR ASSY, RH	
				300-010-100-1	PROPROTOR ASSY, LH	
-4	-3	-2	-1	PART OR IDENTIFYING NO.	NOMENCLATURE OR DESCRIPTION	SPEC / MATERIAL

LIST OF MATERIAL OR PARTS

8. LOCK WIRE PER BPS FW 4043
7. CUT WEIGHT TO LENGTH AS REQUIRED FOR BALANCE; NO WEIGHT MAY BE CUT LESS THAN .50 LONG. BALANCE PROPROTOR ASSEMBLY USING MINIMUM WEIGHT IN EACH BLADE.
6. REMOVE NUT TAB WASHER & LOCK WASHER. TO ADJUST END PLAY & TO CENTER HUB ASSY WITH UNIVERSAL JOINT CENTER WITH ADJUSTING SCREWS. CENTER WITHIN .002. TORQUE SCREWS TO 10-20 IN-LB & REPLACE LOCKWASHER, TABWASHER & NUT. TORQUE NUT 50-100 IN-LB & LOCK.
6. FILL SPINDLE BRG & TRUNNION BRG CAVITIES WITH SAE 10W-30 MIL-L-2104 OIL BY WORKING ALL AIR OUT AS OIL IS ADDED. FILL UNTIL RESERVOIRS ARE 1/2 FULL.
5. THIS PART IS INNER RACE OF 204-010-251-1 BEARING
4. BOND WITH EPON 934 PER FW4355
3. LOCKWIRE & COTTER PIN PER BPS FW4043
2. TORQUE PER BPS FW4418 EXCEPT AS SHOWN
1. FOR HUB ASSY NO. & HUB SERIAL NO. MARK PER BPS FW 4150

NOTES:

			-2	286-940-920		
DASH NO.	NEXT ASSY	USED ON	DASH NO.	NEXT ASSY	USED ON	
APPLICATION			APPLICATION			

0 INCHES SCALE 10

SCH. LENASE 10 11 68
LOVINGTON 11 68
W. Oswald
BELL HELICOPTER
TITLE PROPROTOR AS
300
SCALE FULL

-4	-3	-2	-1	PART OR IDENTIFYING NO.	NOMENCLATURE OR DESCRIPTION	REMARKS
3	3			IR [illegible]-10-1	INNER RACE	5
		AR	AR	EPON 934	ADHESIVE	
AR	AR			29[illegible]	PRO SEAL	
4	4			LCN12 820	NUT	PURCHASE FROM SPS FASTNERS
		4	4	MS24665-304	PIN COTTER	
AR	AR			MS 20995-[illegible]	LOCK WIRE	
4	4			MS 20002C 8	WASHER	
3	3			MS29561-240	O RING	
3	3			MS29561-254	O RING	
		3	3	MS 24665-372	PIN COTTER	
6	6			MS17825-4	NUT	
4	4			MS20002-8	WASHER	
4	4			MS1308-36	BOLT	
		1	1	AN 750-1	NAME PLATE	
12	12			AN 910-416L	WASHER	
18	18			AN 960-PD10L	WASHER	
18	18			AN 502-10-8	SCREW	
6	6			AN 174-21	BOLT	
1	1			300-010-200-1	UNIVERSAL JOINT ASSY	
3				107-16	BRACKET, RH	
	3			107-15	BRACKET, LH	
3	3			107-13	SHIM	
3	3			107-9	RETAINER	
3	3			107-7	CLAMP SET	
3	3			107-5	RING	
3	3			107-1	SPACER	
		3	3	[illegible]-3	NUT	
		3	3	106-1	BOLT ASSY	
3	3			[illegible]	[illegible] ASSY	
3	3			[illegible]	[illegible] BEARING	

LAST SECTION

LETTER USED

-4	-3	-2	-1	PART OR IDENTIFYING NO.	NOMENCLATURE OR DESCRIPTION	STOCK SIZE	MATERIAL	MATERIAL SPEC
2	2			103-1	STRAP ASSY			
7	7			102-1	YOKE ASSY			
		AR	AR	011-16	BALANCE WEIGHT			
		AR	AR	011-15	BALANCE WEIGHT			
		AR	AR	011-13	BALANCE WEIGHT			
		AR	AR	011-11	BALANCE WEIGHT			
		AR	AR	011-9	BALANCE WEIGHT			
		AR	AR	[illegible]	BALANCE WEIGHT			
		3		001-2	BLADE ASSY, RH			
			3	001-1	BLADE ASSY, LH			
		1		100-4	HUB ASSY, RH			
			1	100-3	HUB ASSY, LH			
				100-2	PROPROTOR ASSY, RH			
				300-010-100-1	PROPROTOR ASSY, LH			

LIST OF MATERIAL OR PARTS

300-010-100

W/O 4390

NOTES:

1. FOR HUB ASSY NO. & [illegible] SERIAL NO. MARK PER BPS FW 4150
2. TORQUE PER BPS FW 4018 EXCEPT AS SHOWN
3. LOCKWIRE & COTTER PIN PER BPS FW 4043
4. BOND WITH EPON 934 PER FW 4335
5. THIS PART IS INNER RACE OF 204-040-251-1 BEARING
6. FILL SPINDLE BRG & TRUNNION BRG CAVITIES WITH SAE 10W-30 MIL-L-2104 OIL BY WORKING ALL AIR OUT AS OIL IS ADDED. FILL UNTIL RESERVOIRS ARE 1/2 FULL.
7. REMOVE NUT, TAB WASHER & LOCK WASHER TO ADJUST END PLAY & TO CENTER HUB ASSY WITH UNIVERSAL JOINT. CENTER WITH ADJUSTING SCREWS. CENTER WITHIN .002. TORQUE SCREWS TO 10-20 IN-LB & REPLACE LOCKWASHER, TABWASHER & NUT. TORQUE NUT 50-100 IN-LBS & LOCK.
8. CUT WEIGHT TO LENGTH AS REQUIRED FOR BALANCE; NO WEIGHT MAY BE CUT LESS THAN .50 LONG. BALANCE PROPROTOR ASSEMBLY USING MINIMUM WEIGHT IN EACH BLADE.
9. LOCK WIRE PER BPS FW 4043

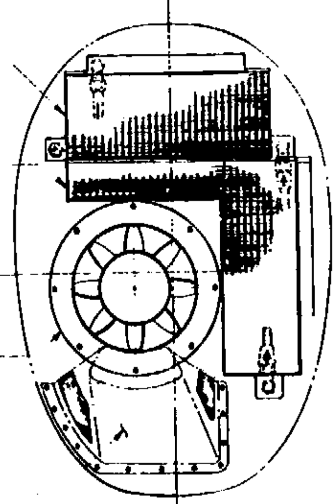
BELL HELICOPTER COMPANY

PROPROTOR ASSY

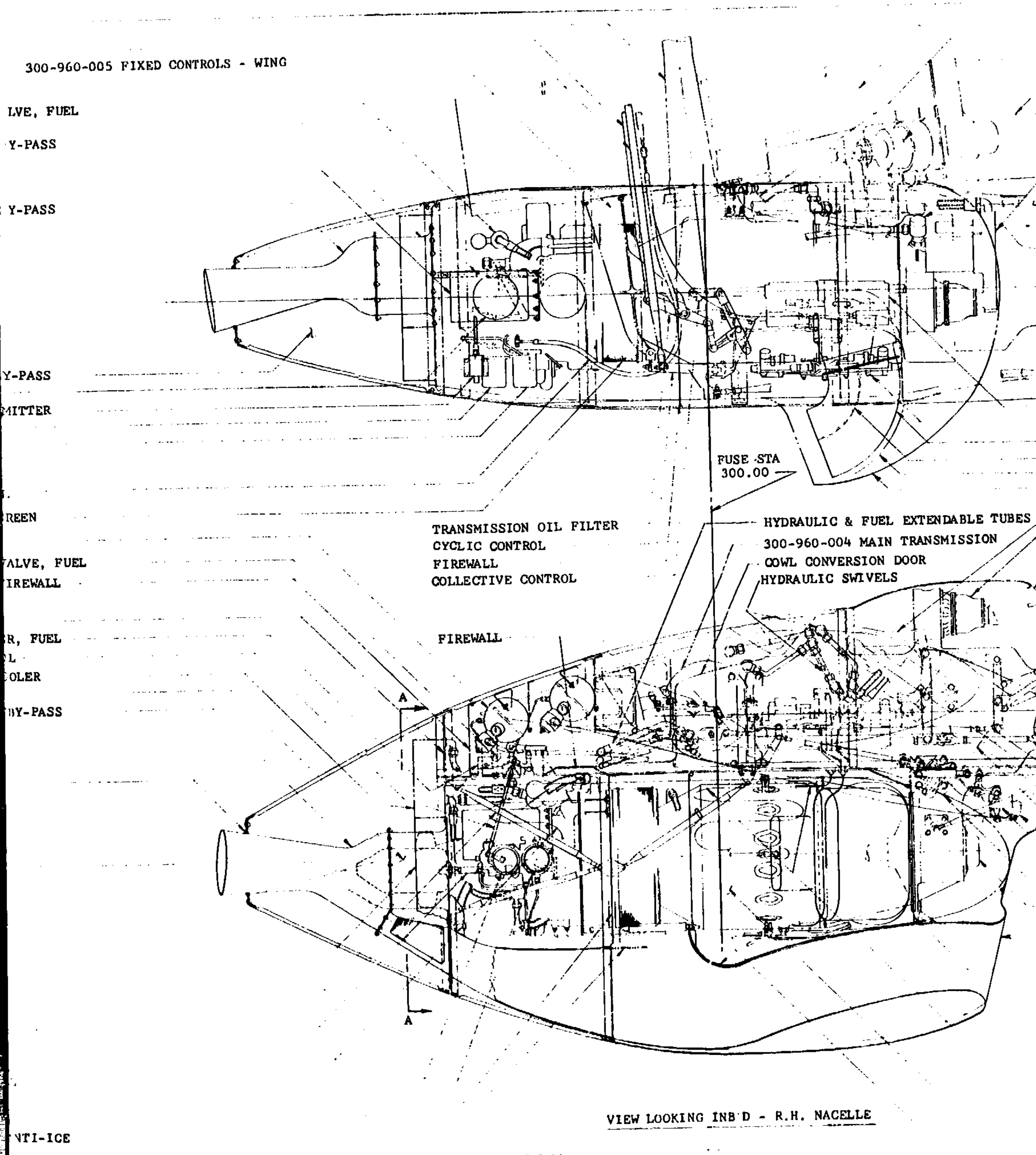
SCH. RELEASE 6-11-68

204-947-002

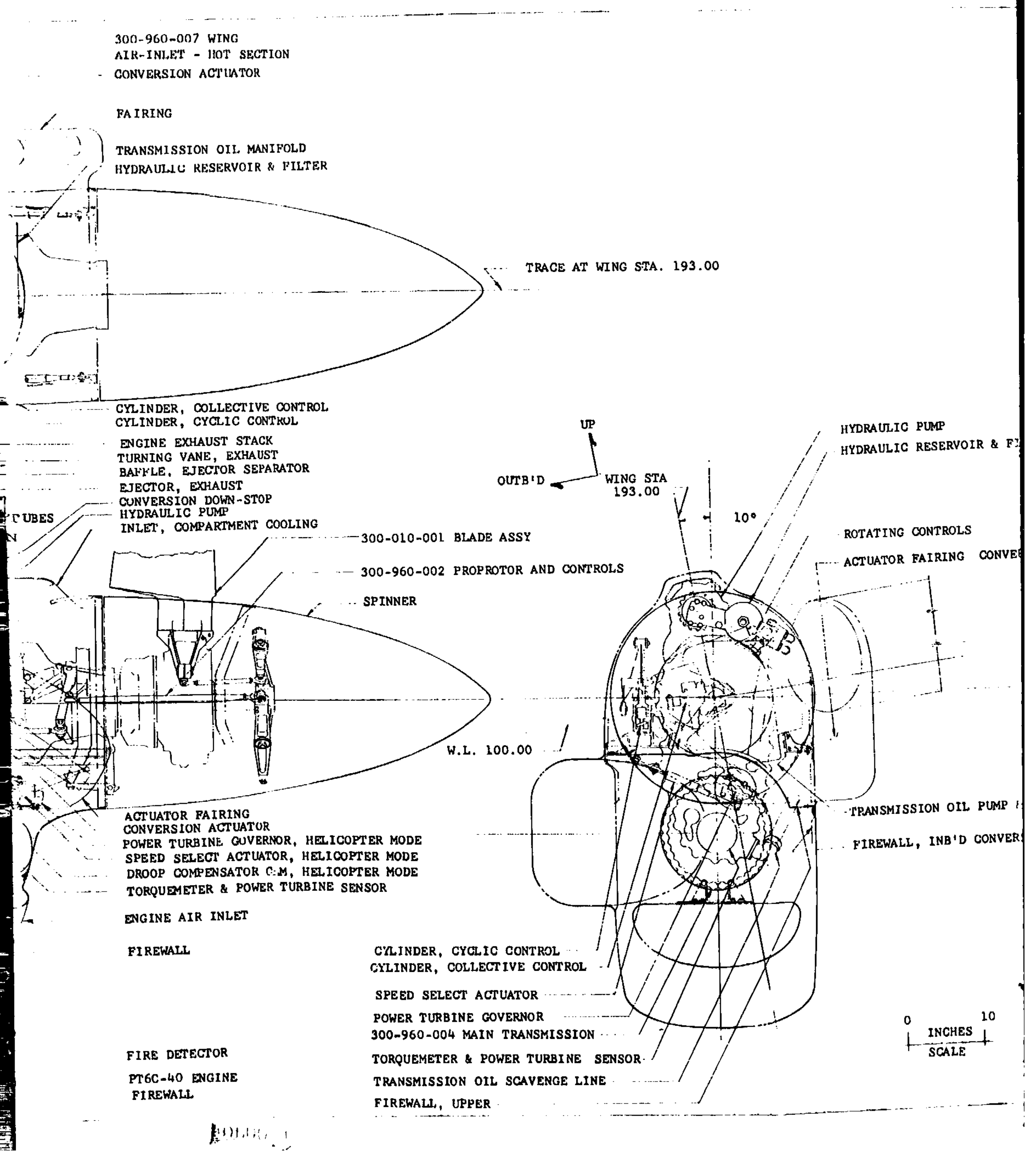
300-010-100


SPINDLE
COLLECTIVE CONTROL
CYCLIC CONTROL
FIXED CONTROL MOUNT
300-960-005 FIXED CONTROLS
FIREWALL SHUT-OFF VALVE, FUEL
BLOWER DRIVE SHAFT
BLOWER, OIL COOLER BY-PASS
STARTER-GENERATOR
EJECTOR, INDUCTION BY-PASS
PLENUM, INDUCTION BY-PASS
FIRE EXTINGUISHER
FUEL PRESSURE TRANSMITTER
FUEL FILTER, ENGINE
OIL FILTER, ENGINE
GAS PRODUCER CONTROL
ENGINE AIR INLET SCREEN
FIRE EXTINGUISHER
FIREWALL SHUT-OFF VALVE, FUEL
CLOSE-OUT BAFFLE, FIREWALL
BAFFLE, OIL COOLER
PRESSURE TRANSMITTER, FUEL
GAS PRODUCER CONTROL
TRANSMISSION OIL COOLER
BLOWER, OIL COOLER
EJECTOR, INDUCTION BY-PASS
TRANSMISSION OIL COOLER
PLENUM SEAL
ENGINE OIL COOLER
BLOWER, OIL COOLER
STARTER GENERATOR
ENGINE OIL COOLER
COWL SUPPORT
BY-PASS DUCT, COOLINC AIR
BLOWER DRIVESHAFT
FIRE DETECTOR
FILTER, ENGINE FUEL
FILTER, ENGINE OIL
INDUCTION BAFFLE
SCREEN, INDUCTION ANTI-ICE

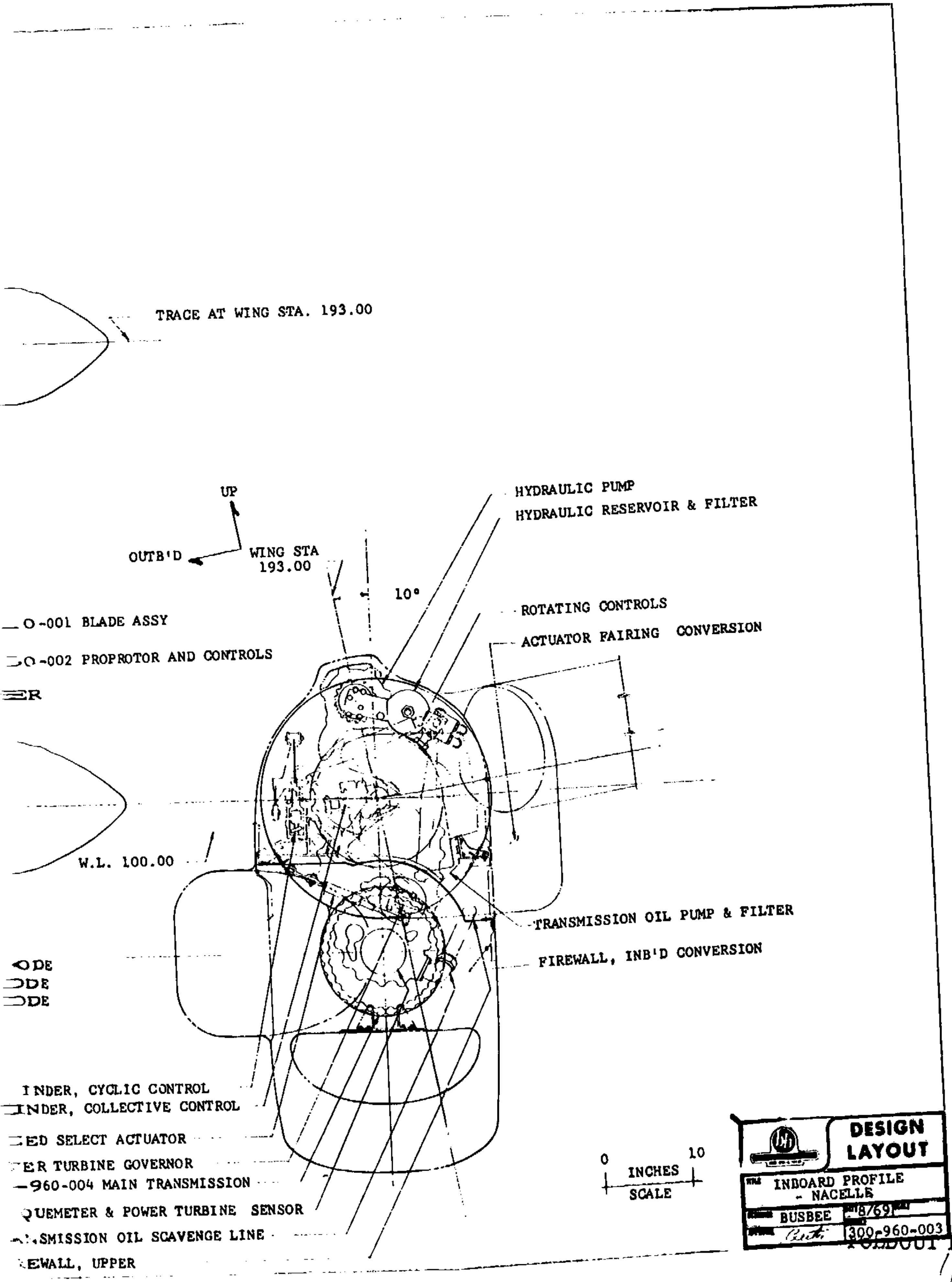
SECTION A-A


300-960-005 FIXED CONTROLS - WING
LVE, FUEL
Y-PASS
Y-PASS
Y-PASS
MITTER
REEN
ALVE, FUEL
IREWALL
R, FUEL
L
OLER
BY-PASS
NTI-ICE
FUSE STA
300.00
TRANSMISSION OIL FILTER
CYCLIC CONTROL
FIREWALL
COLLECTIVE CONTROL
FIREWALL
HYDRAULIC & FUEL EXTENDABLE TUBES
300-960-004 MAIN TRANSMISSION
COWL CONVERSION DOOR
HYDRAULIC SWIVELS
A
A
VIEW LOOKING INB'D - R.H. NACELLE

FOLDOUT



300-960-007 WING
AIR-INLET - HOT SECTION
CONVERSION ACTUATOR
FAIRING
TRANSMISSION OIL MANIFOLD
HYDRAULIC RESERVOIR & FILTER
TRACE AT WING STA. 193.00
CYLINDER, COLLECTIVE CONTROL
CYLINDER, CYCLIC CONTROL
ENGINE EXHAUST STACK
TURNING VANE, EXHAUST
BAFFLE, EJECTOR SEPARATOR
EJECTOR, EXHAUST
CONVERSION DOWN-STOP
HYDRAULIC PUMP
INLET, COMPARTMENT COOLING
TUBES
300-010-001 BLADE ASSY
300-960-002 PROPROTOR AND CONTROLS
SPINNER
W.L. 100.00
ACTUATOR FAIRING
CONVERSION ACTUATOR
POWER TURBINE GOVERNOR, HELICOPTER MODE
SPEED SELECT ACTUATOR, HELICOPTER MODE
DROOP COMPENSATOR C.M, HELICOPTER MODE
TORQUEMETER & POWER TURBINE SENSOR
ENGINE AIR INLET
FIREWALL
FIRE DETECTOR
PT6C-40 ENGINE
FIREWALL
UP
OUTB'D
WING STA
193.00
10°
HYDRAULIC PUMP
HYDRAULIC RESERVOIR & FI
ROTATING CONTROLS
ACTUATOR FAIRING
CONVE
TRANSMISSION OIL PUMP
FIREWALL, INB'D CONVER
CYLINDER, CYCLIC CONTROL
CYLINDER, COLLECTIVE CONTROL
SPEED SELECT ACTUATOR
POWER TURBINE GOVERNOR
300-960-004 MAIN TRANSMISSION
TORQUEMETER & POWER TURBINE SENSOR
TRANSMISSION OIL SCAVENGE LINE
FIREWALL, UPPER
0
10
INCHES
SCALE



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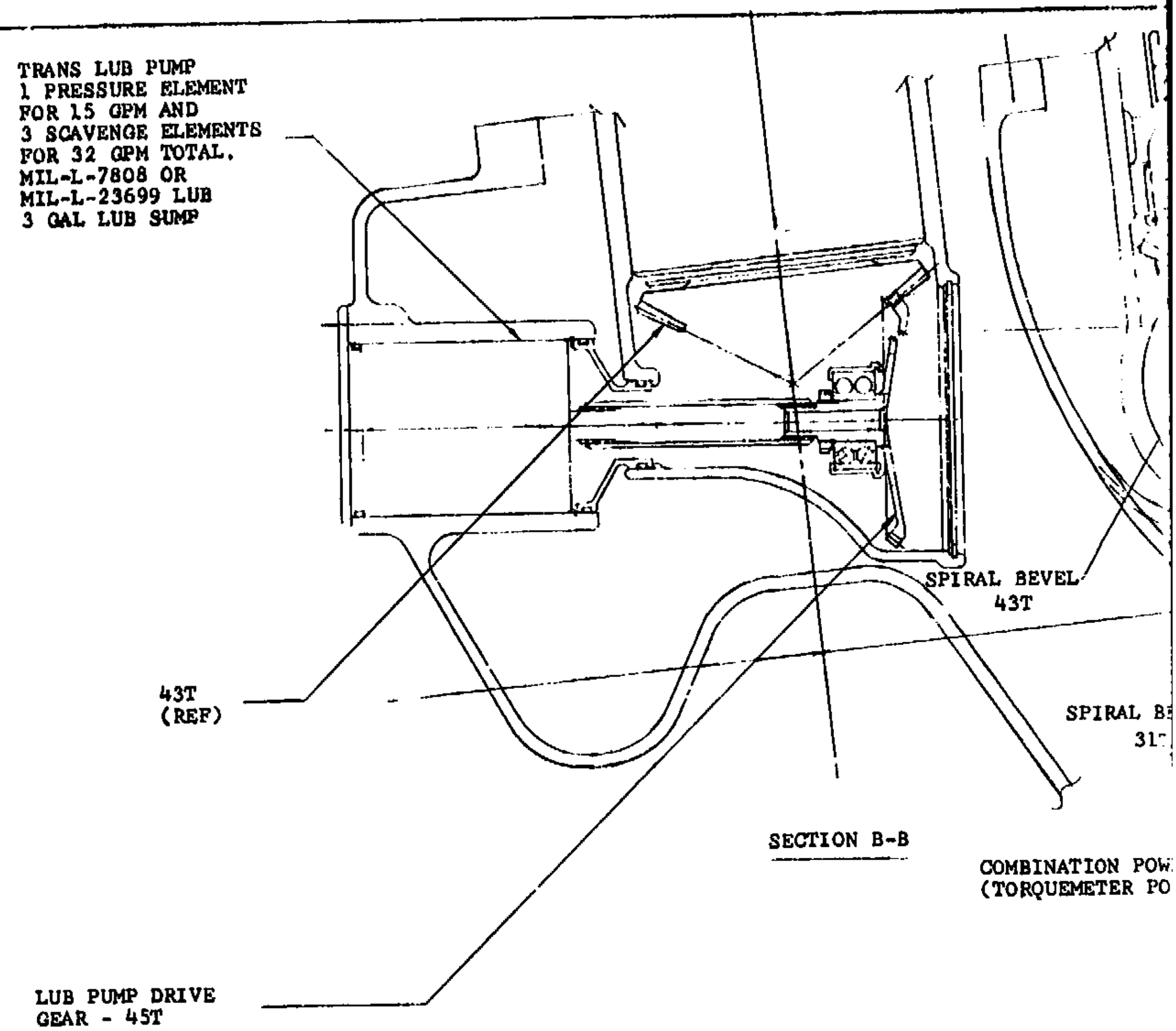
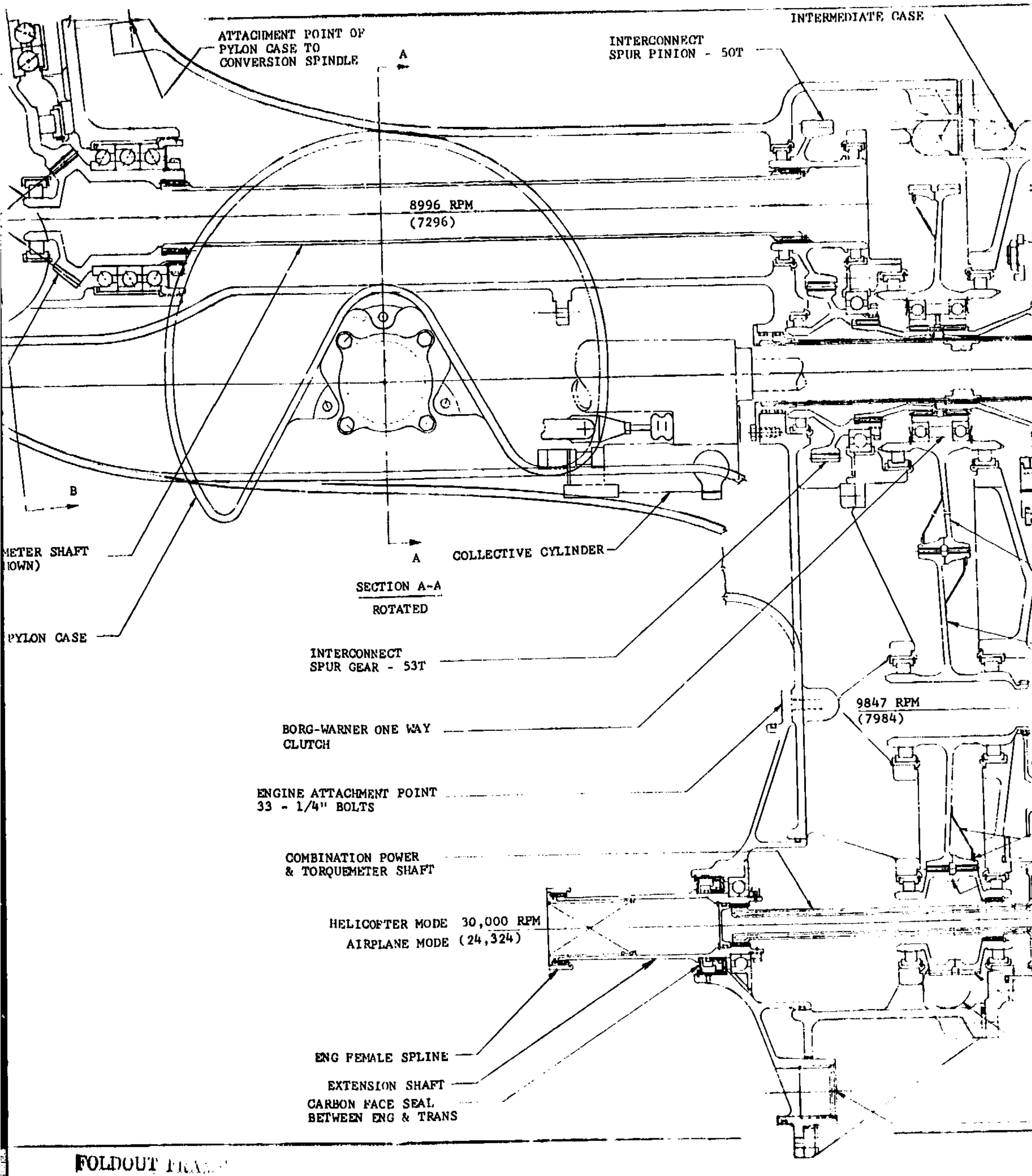
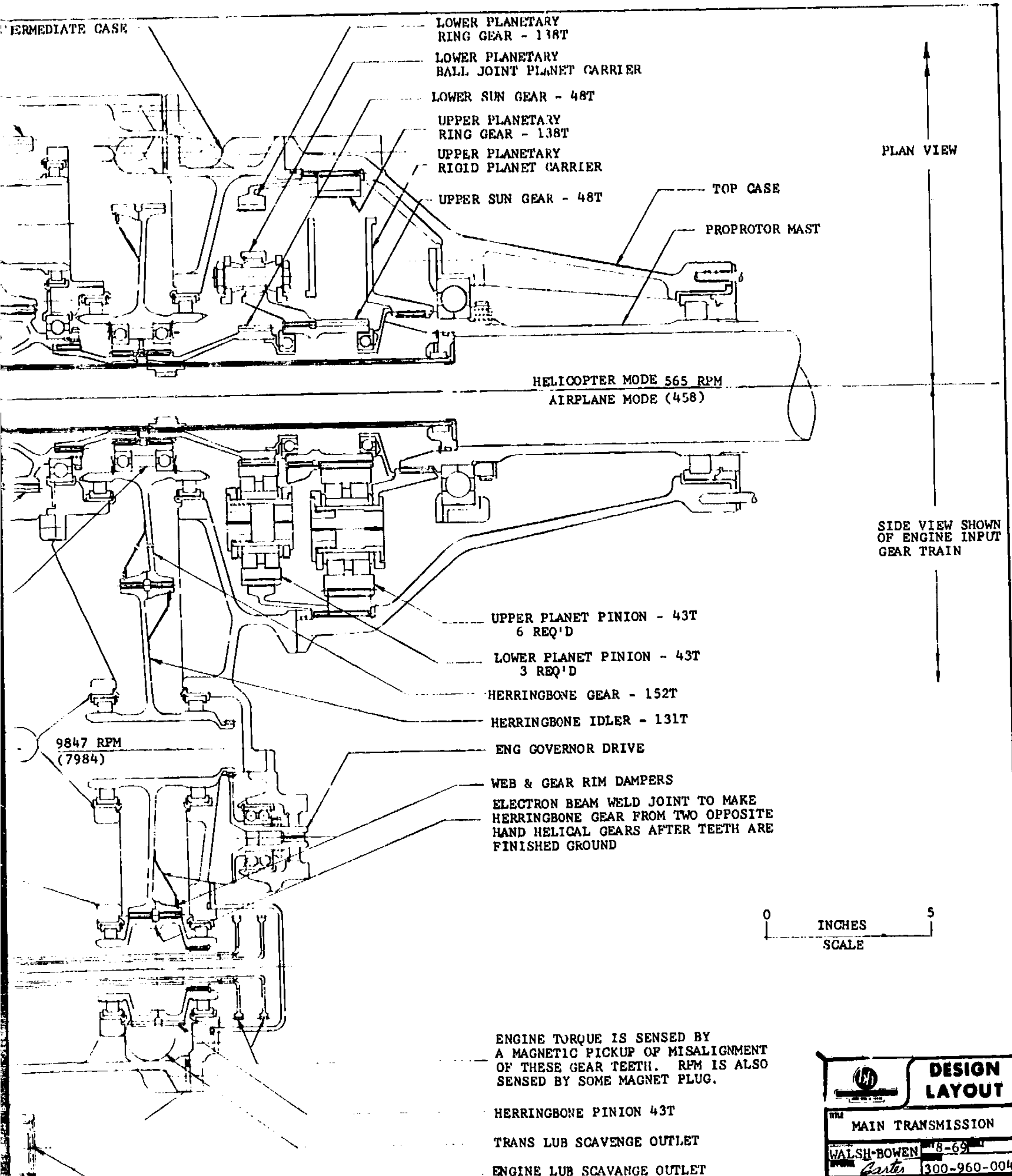


FOLDOUT FRAME



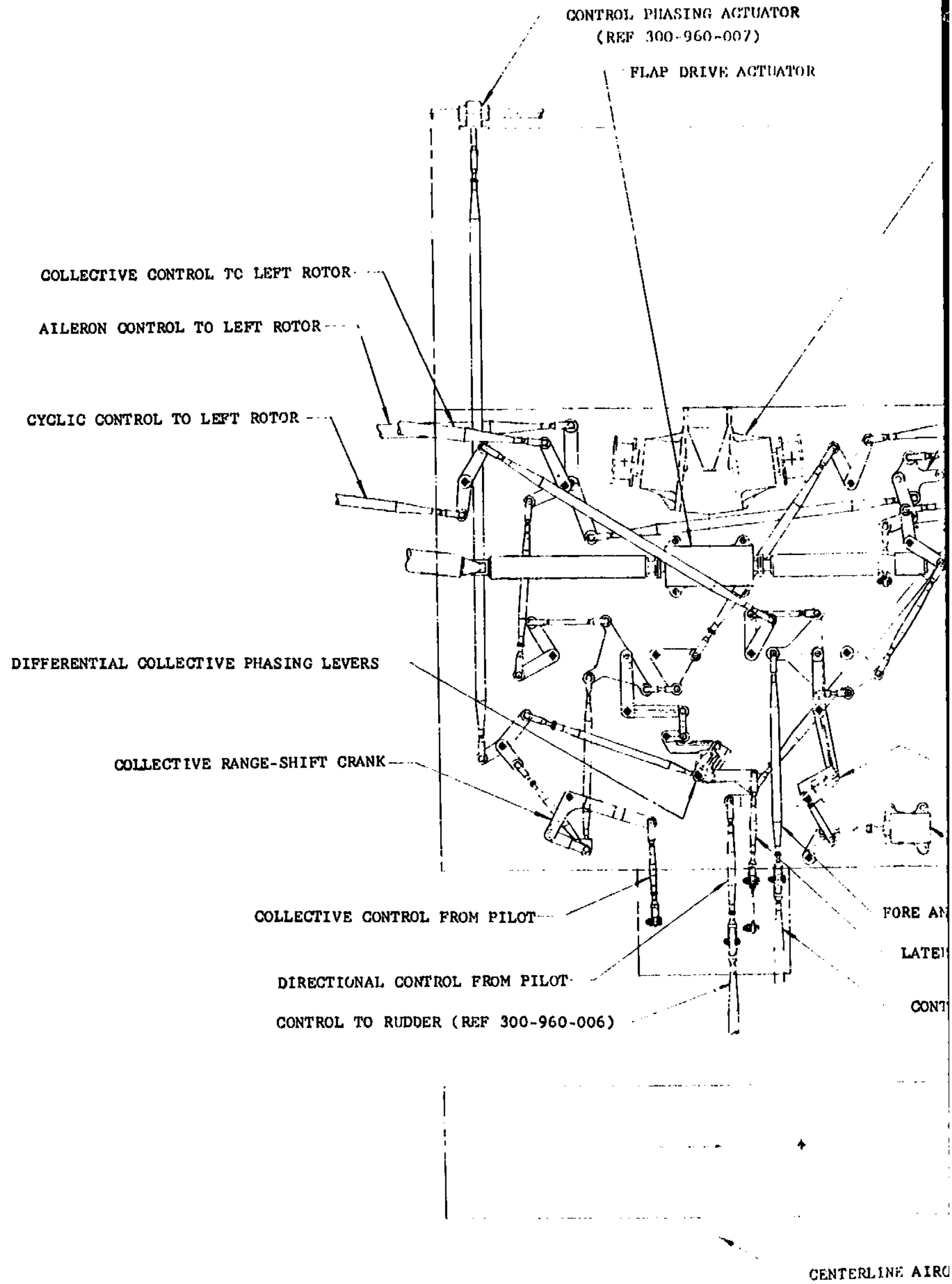
ATTACHMENT POINT OF PYLON CASE TO CONVERSION SPINDLE
A
INTERCONNECT SPUR PINION - 50T
INTERMEDIATE CASE
8996 RPM
(7296)
B
METER SHAFT
OWN)
A
COLLECTIVE CYLINDER
SECTION A-A
ROTATED
PYLON CASE
INTERCONNECT SPUR GEAR - 53T
9847 RPM
(7984)
BORG-WARNER ONE WAY CLUTCH
ENGINE ATTACHMENT POINT
33 - 1/4" BOLTS
COMBINATION POWER & TORQUEMETER SHAFT
HELICOPTER MODE 30,000 RPM
AIRPLANE MODE (24,324)
ENG FEMALE SPLINE
EXTENSION SHAFT
CARBON FACE SEAL BETWEEN ENG & TRANS

FOLDOUT FRA...



ERMEDIATE CASE
LOWER PLANETARY RING GEAR - 138T
LOWER PLANETARY BALL JOINT PLANET CARRIER
LOWER SUN GEAR - 48T
UPPER PLANETARY RING GEAR - 138T
UPPER PLANETARY RIGID PLANET CARRIER
UPPER SUN GEAR - 48T
TOP CASE
PROPROTOR MAST
PLAN VIEW
HELICOPTER MODE 565 RPM
AIRPLANE MODE (458)
SIDE VIEW SHOWN OF ENGINE INPUT GEAR TRAIN
UPPER PLANET PINION - 43T
6 REQ'D
LOWER PLANET PINION - 43T
3 REQ'D
HERRINGBONE GEAR - 152T
HERRINGBONE IDLER - 131T
9847 RPM
(7984)
ENG GOVERNOR DRIVE
WEB & GEAR RIM DAMPERS
ELECTRON BEAM WELD JOINT TO MAKE HERRINGBONE GEAR FROM TWO OPPOSITE HAND HELICAL GEARS AFTER TEETH ARE FINISHED GROUND
0
INCHES
5
SCALE
ENGINE TORQUE IS SENSED BY A MAGNETIC PICKUP OF MISALIGNMENT OF THESE GEAR TEETH. RPM IS ALSO SENSED BY SOME MAGNET PLUG.
HERRINGBONE PINION 43T
TRANS LUB SCAVENGE OUTLET
ENGINE LUB SCAVANGE OUTLET
DESIGN LAYOUT
MAIN TRANSMISSION
WALSH-BOWEN
8-69
Carter
300-960-004

FOLDOUT FRA



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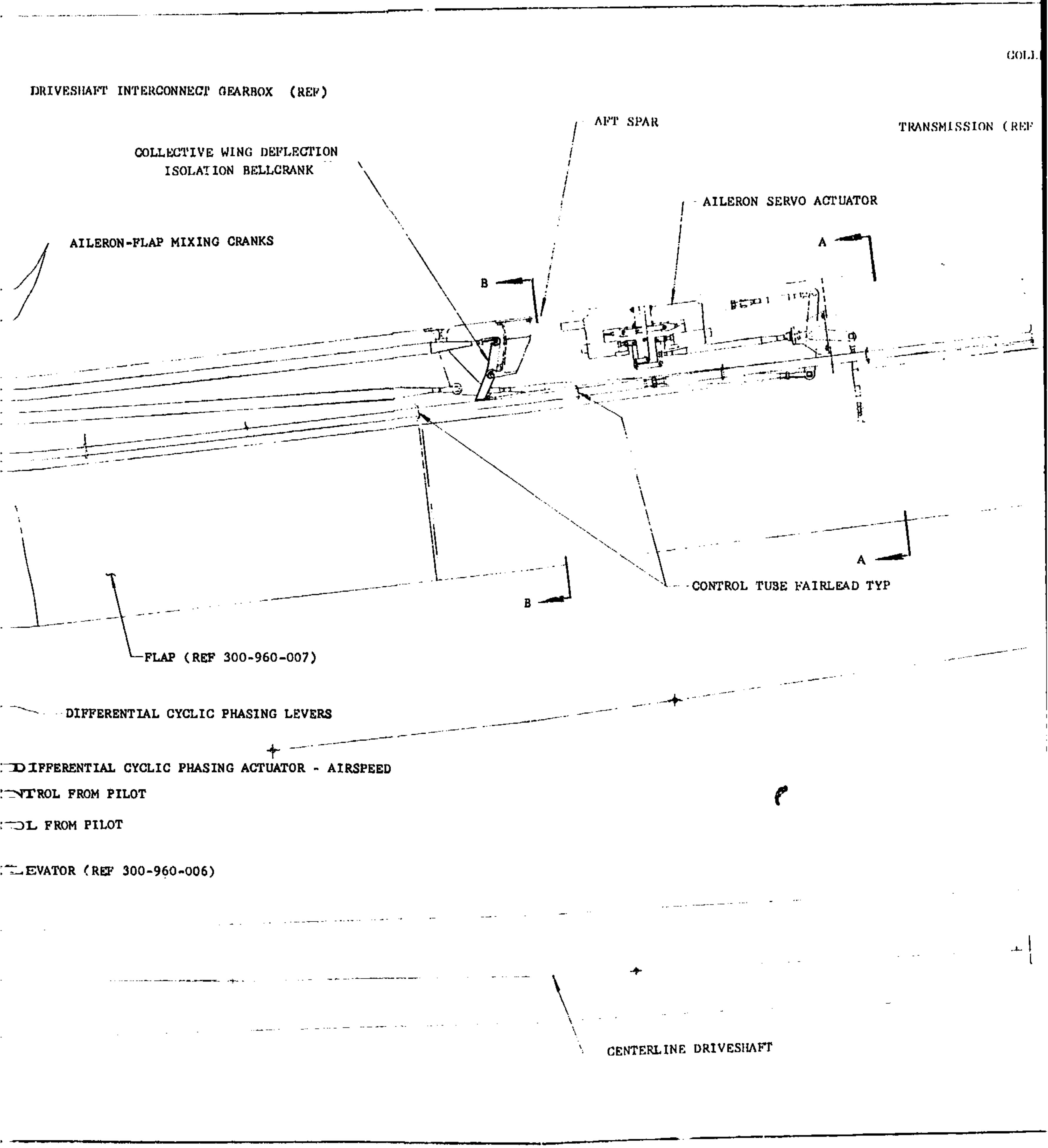


COLL
DRIVESHAFT INTERCONNECT GEARBOX (REF)
AFT SPAR
TRANSMISSION (REF
COLLECTIVE WING DEFLECTION
ISOLATION BELLCRANK
AILERON SERVO ACTUATOR
AILERON-FLAP MIXING CRANKS
A
B
CONTROL TUBE FAIRLEAD TYP
A
B
FLAP (REF 300-960-007)
DIFFERENTIAL CYCLIC PHASING LEVERS
DIFFERENTIAL CYCLIC PHASING ACTUATOR - AIRSPEED
NTROL FROM PILOT
L FROM PILOT
EVATOR (REF 300-960-006)
CENTERLINE DRIVESHAFT

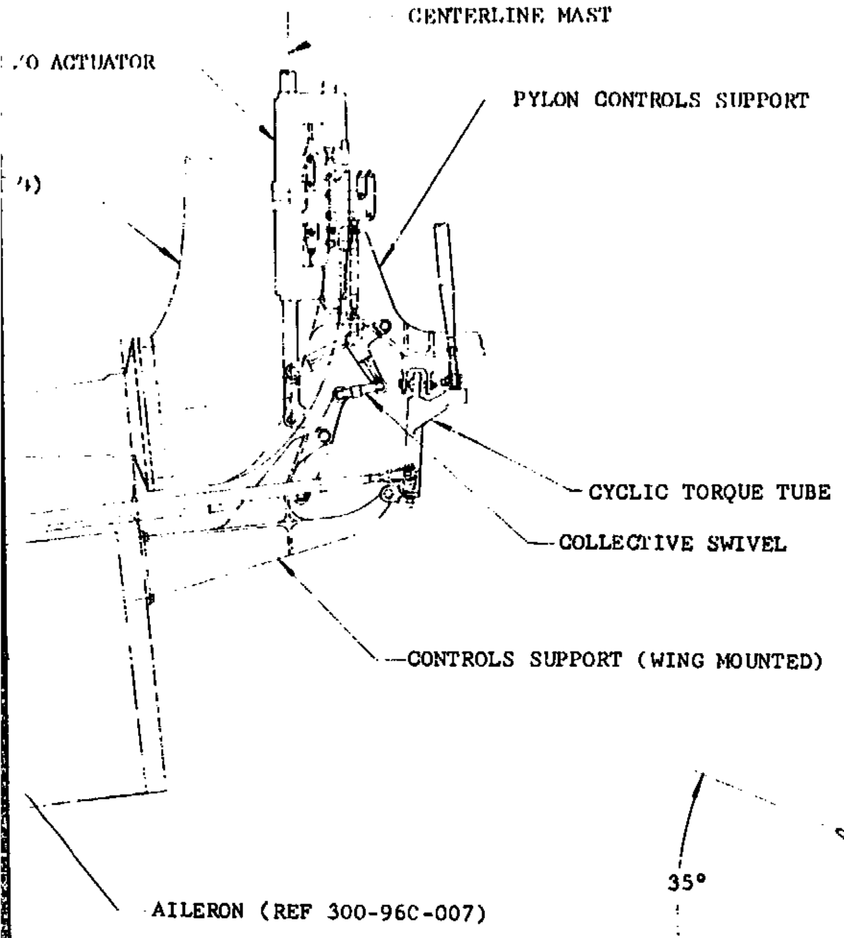
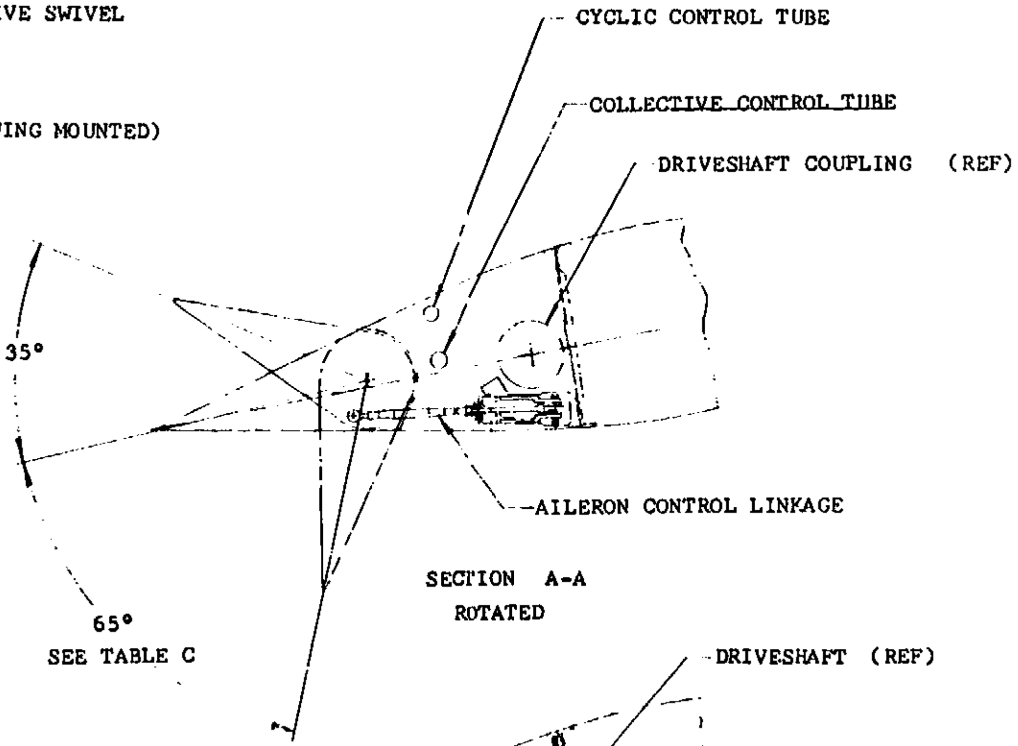
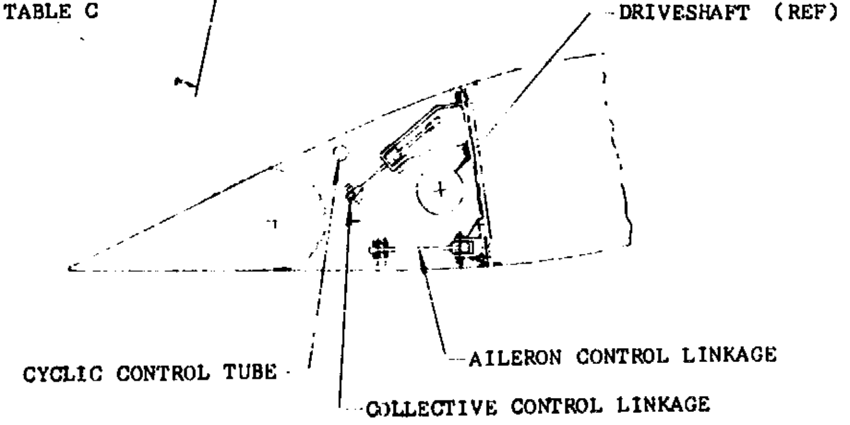


TABLE C

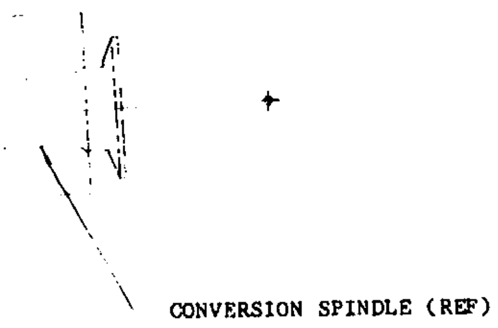
FLAP AILERON POSITION	AILERON CONTROL
+10°	+25.0°/-15.0°
0°	+21.6°/-15.2°
-30°	+18 0°/-12.5°
-60°	-14.2°/- 5.0°



SECTION A-A
ROTATED



SECTION B-B
ROTATED



3. ✦ INDICATES BELLCRANK PIVOT POINTS ATTACHED TO FIXED STRUCTURE

2. COLLECTIVE CONTROLS SHOWN IN MAX PITCH POSITION - ALL OTHER CONTROLS SHOWN IN NEUTRAL POSITION

1. ALL CONTROLS SHOWN IN AIRPLANE MODE

NOTES

ST

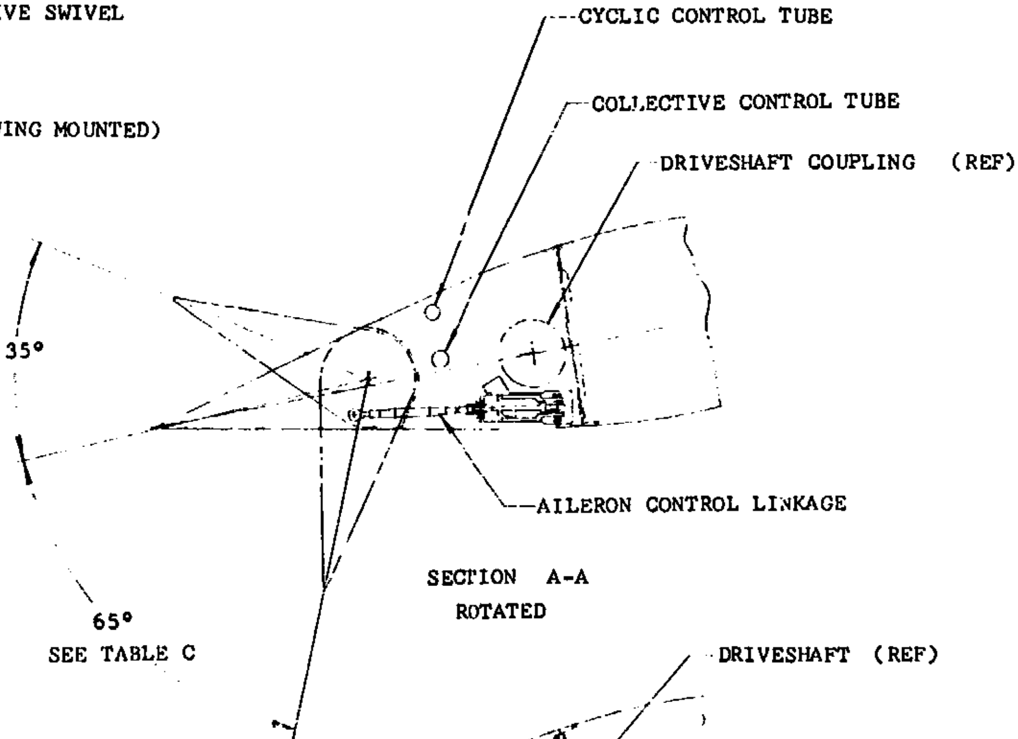
CONTROLS SUPPORT

TABLE C	
FLAP AILERON POSITION	AILERON CONTROL
+10°	+25.0°/-15.0°
0°	+21.6°/-15.2°
-30°	+18.0°/-12.5°
-60°	-14.2°/- 5.0°

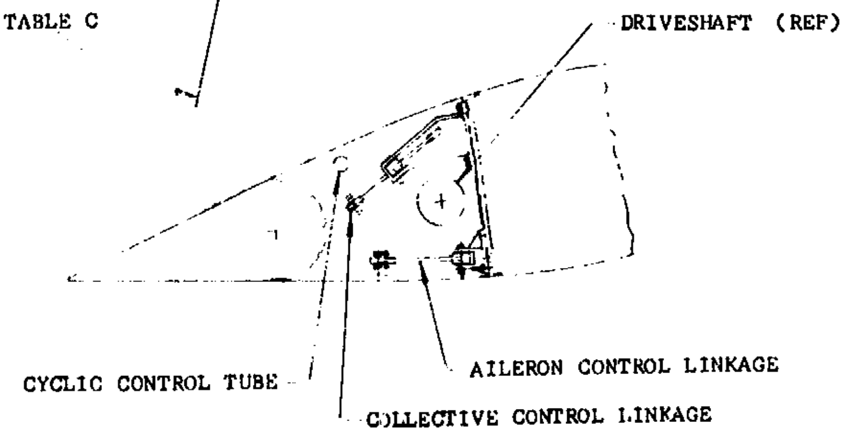
CYCLIC TORQUE TUBE

COLLECTIVE SWIVEL

PORT (WING MOUNTED)



SECTION A-A
ROTATED



0 10
INCHES
SCALE

SECTION B-B
ROTATED

NOTES

1. ALL CONTROLS SHOWN IN AIRPLANE MODE
2. COLLECTIVE CONTROLS SHOWN IN MAX PITCH POSITION - ALL OTHER CONTROLS SHOWN IN NEUTRAL POSITION
3. ✦ INDICATES BELLCRANK PIVOT POINTS ATTACHED TO FIXED STRUCTURE

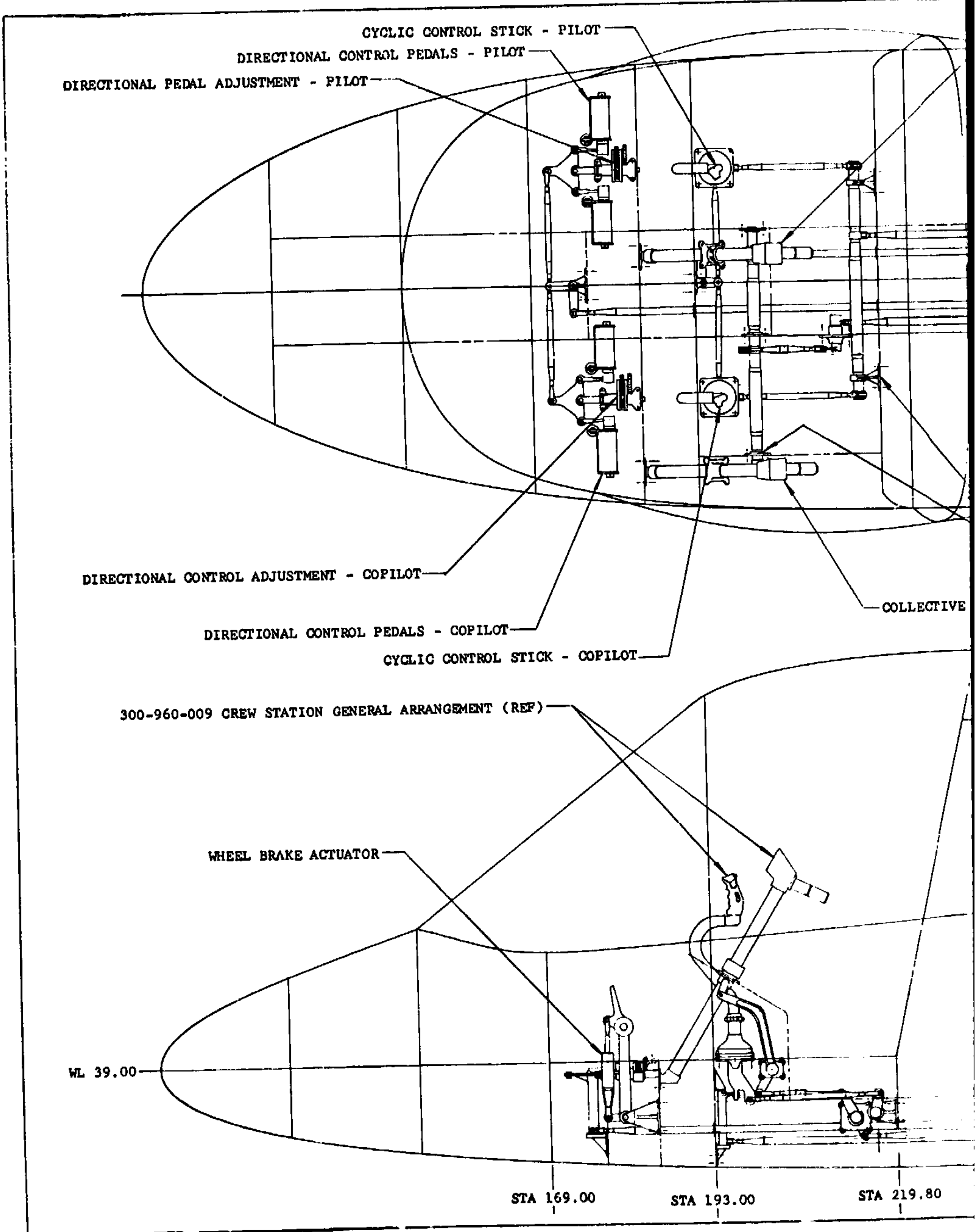
DESIGN LAYOUT

FIXED CONTROLS - WING

ESSARY

8-69

300-069-005



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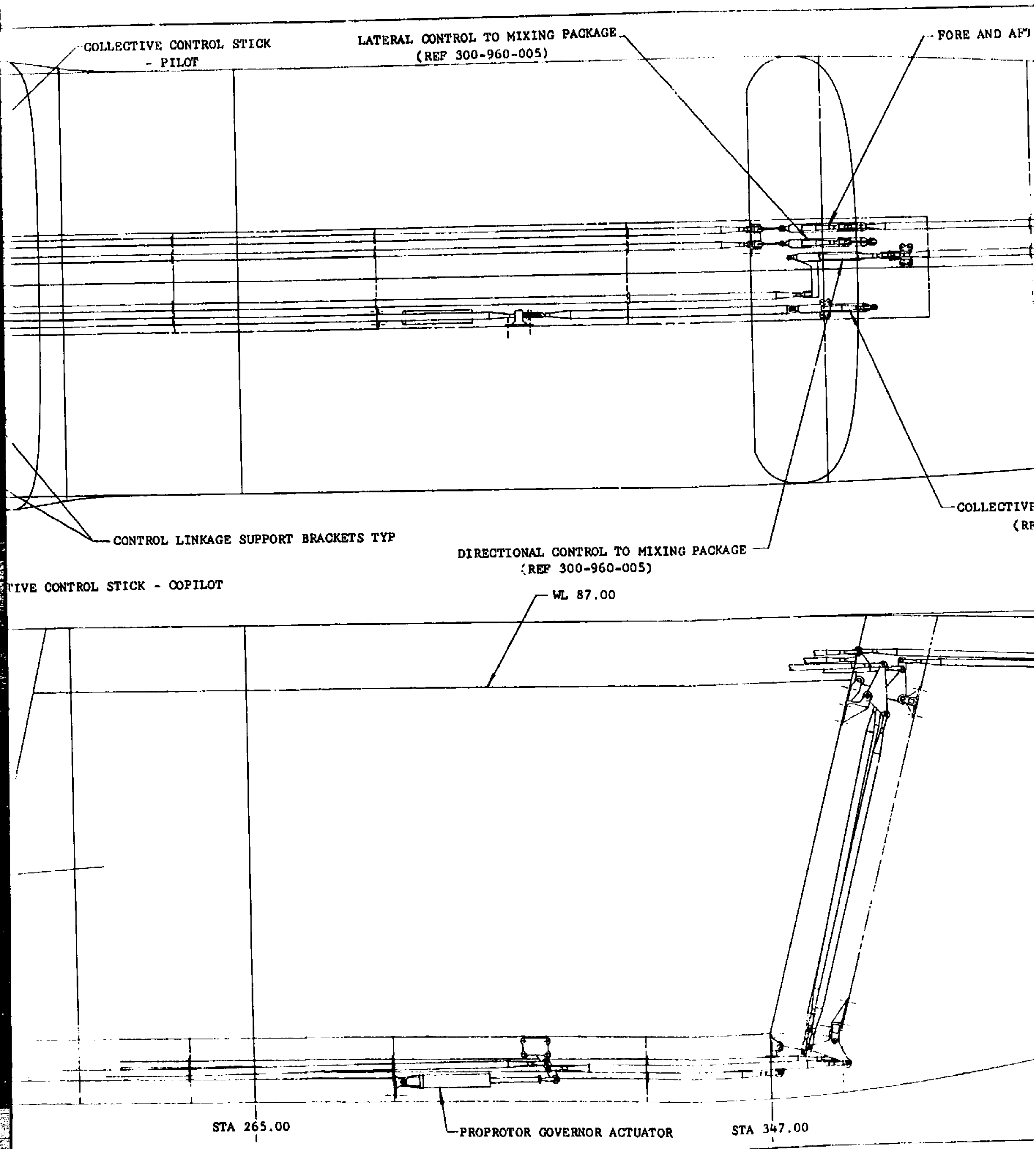
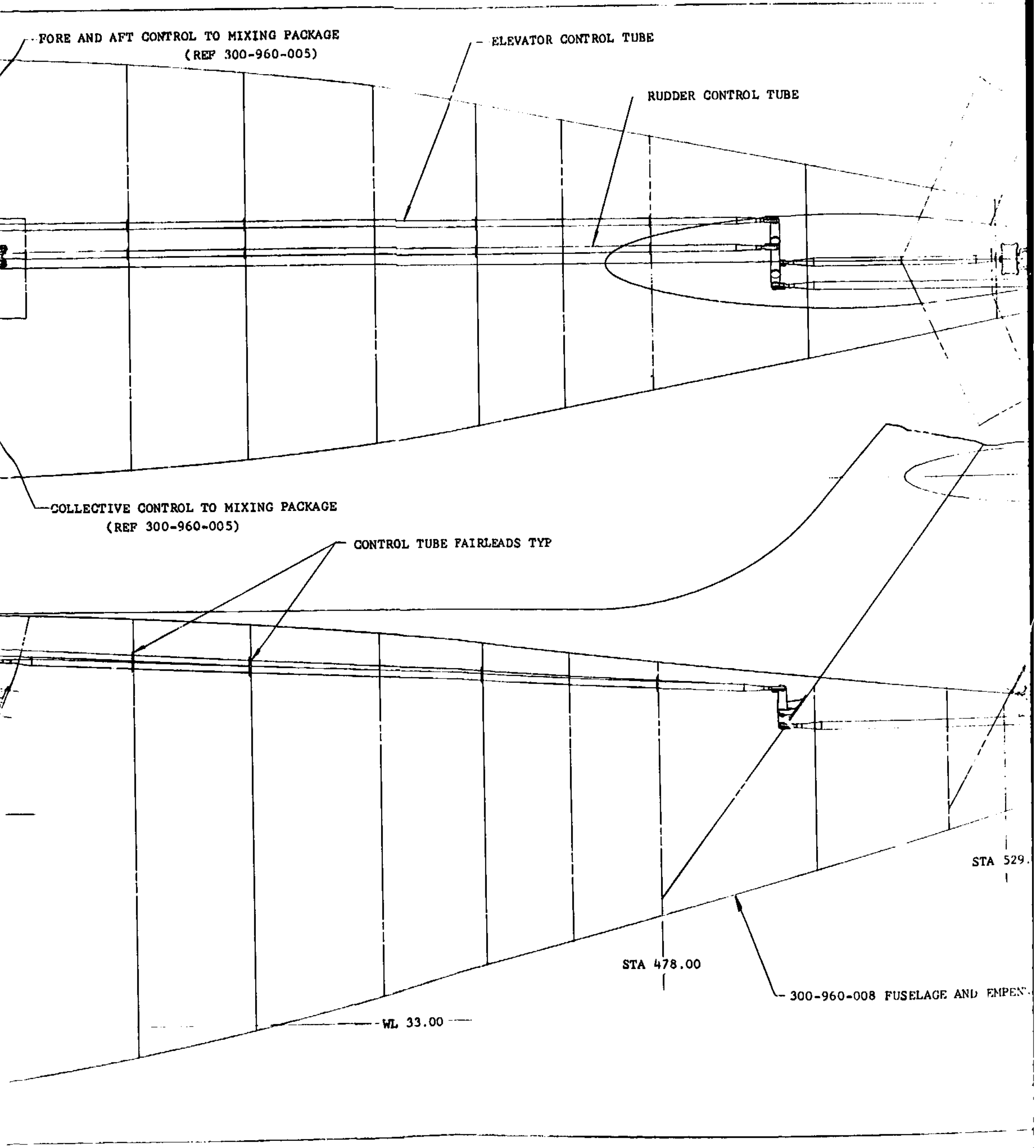

COLLECTIVE CONTROL STICK - PILOT
LATERAL CONTROL TO MIXING PACKAGE (REF 300-960-005)
FORE AND AF
CONTROL LINKAGE SUPPORT BRACKETS TYP
DIRECTIONAL CONTROL TO MIXING PACKAGE (REF 300-960-005)
COLLECTIVE
(RE
TIVE CONTROL STICK - COPILOT
WL 87.00
STA 265.00
PROPROTOR GOVERNOR ACTUATOR
STA 347.00

FOLDOUT FRA



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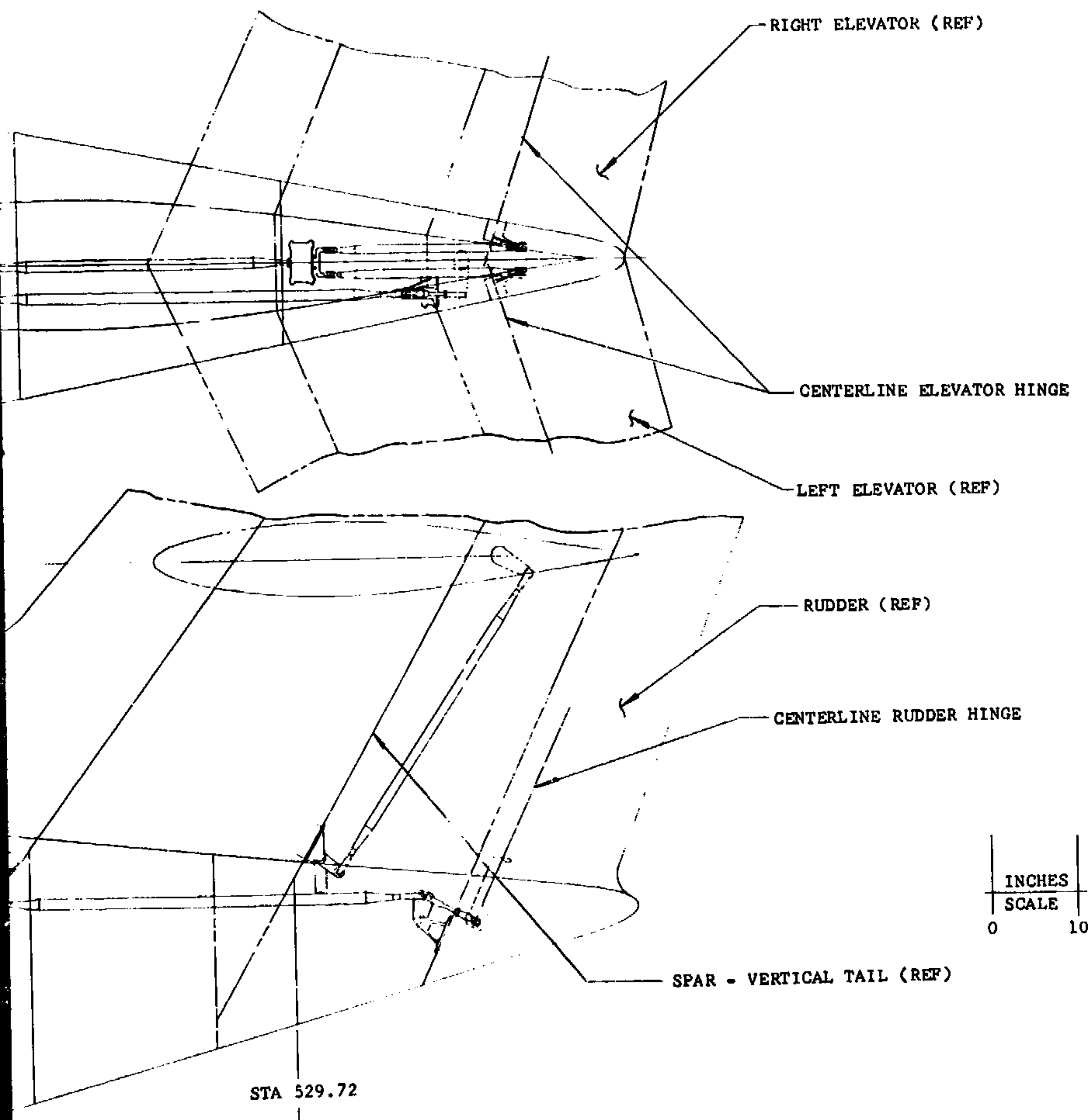


0-960-008 FUSELAGE AND EMPENNAGE (REF)

2. COLLECTIVE CONTROLS SHOWN IN MAX PITCH POSITION - ALL OTHER CONTROLS SHOWN IN NEUTRAL POSITION

1. ALL CONTROLS SHOWN IN AIRPLANE MODE

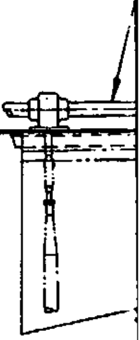
NOTES

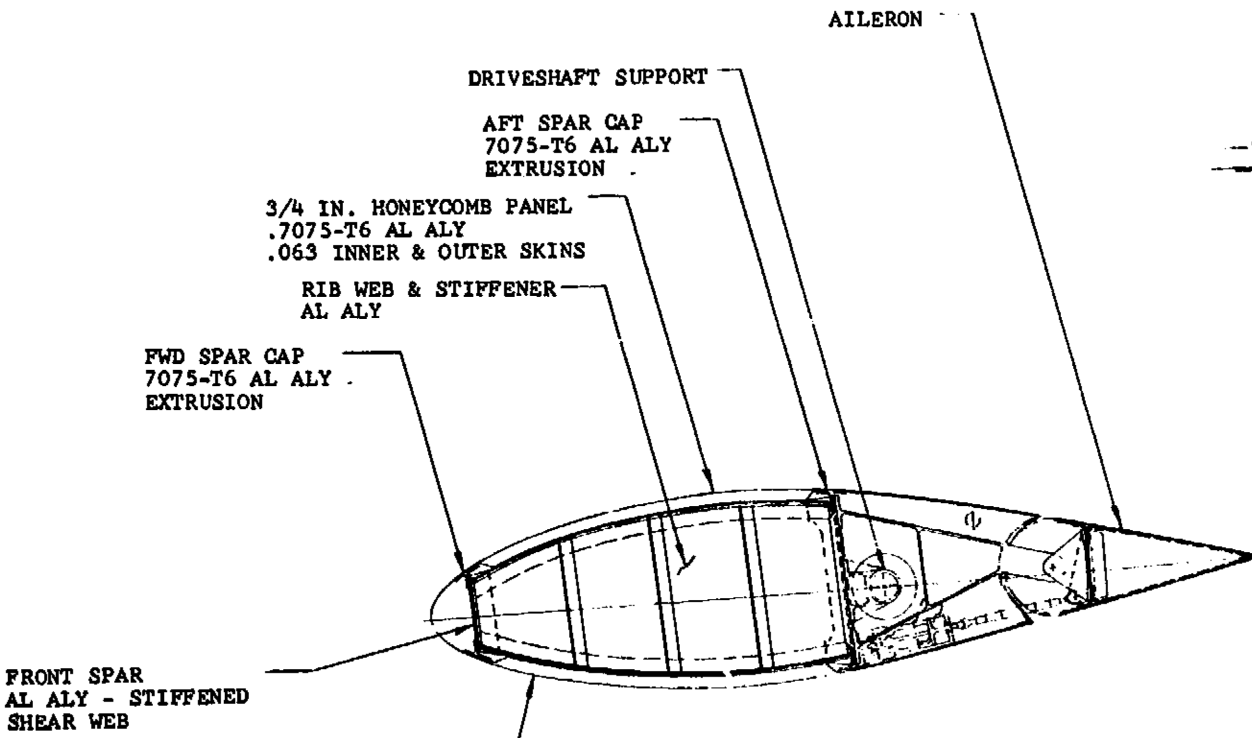
DESIGN LAYOUT

FIXED CONTROLS
FUSELAGE

ESSARY | 8-69

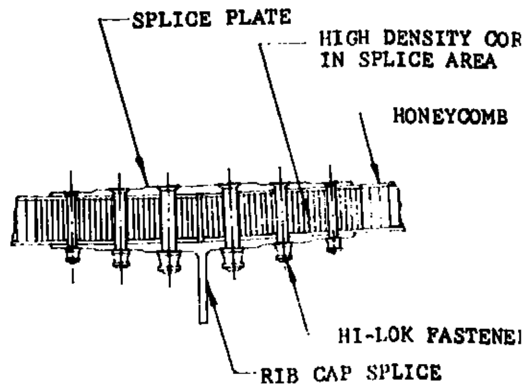
300-960-006





SECTION C-C

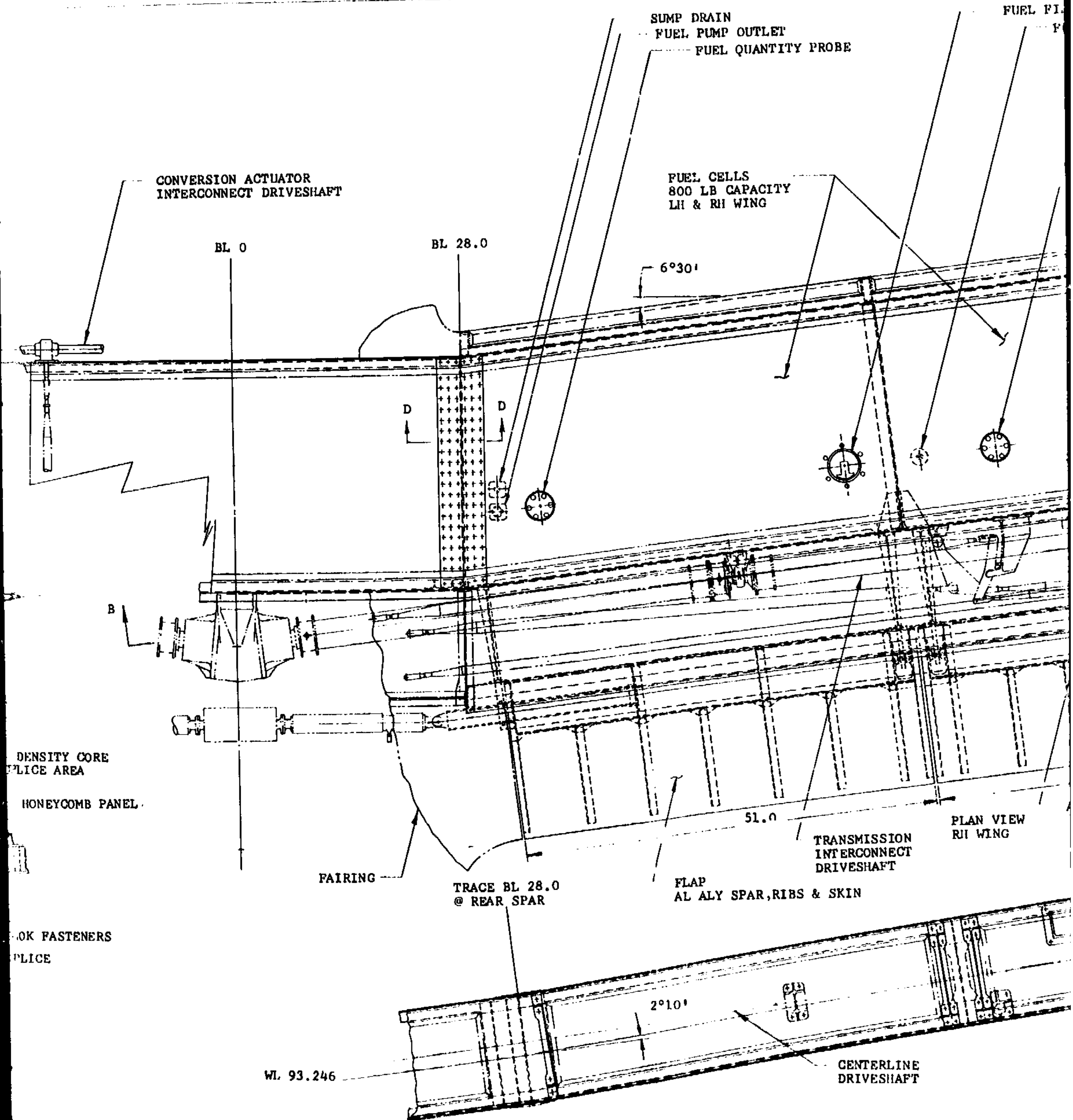
ROTATED 90° CCW



SECTION D-D

WING PANEL SPLICE
TYPICAL UPPER & LOWER
SURFACE

0 5
INCHES
SCALE D-D



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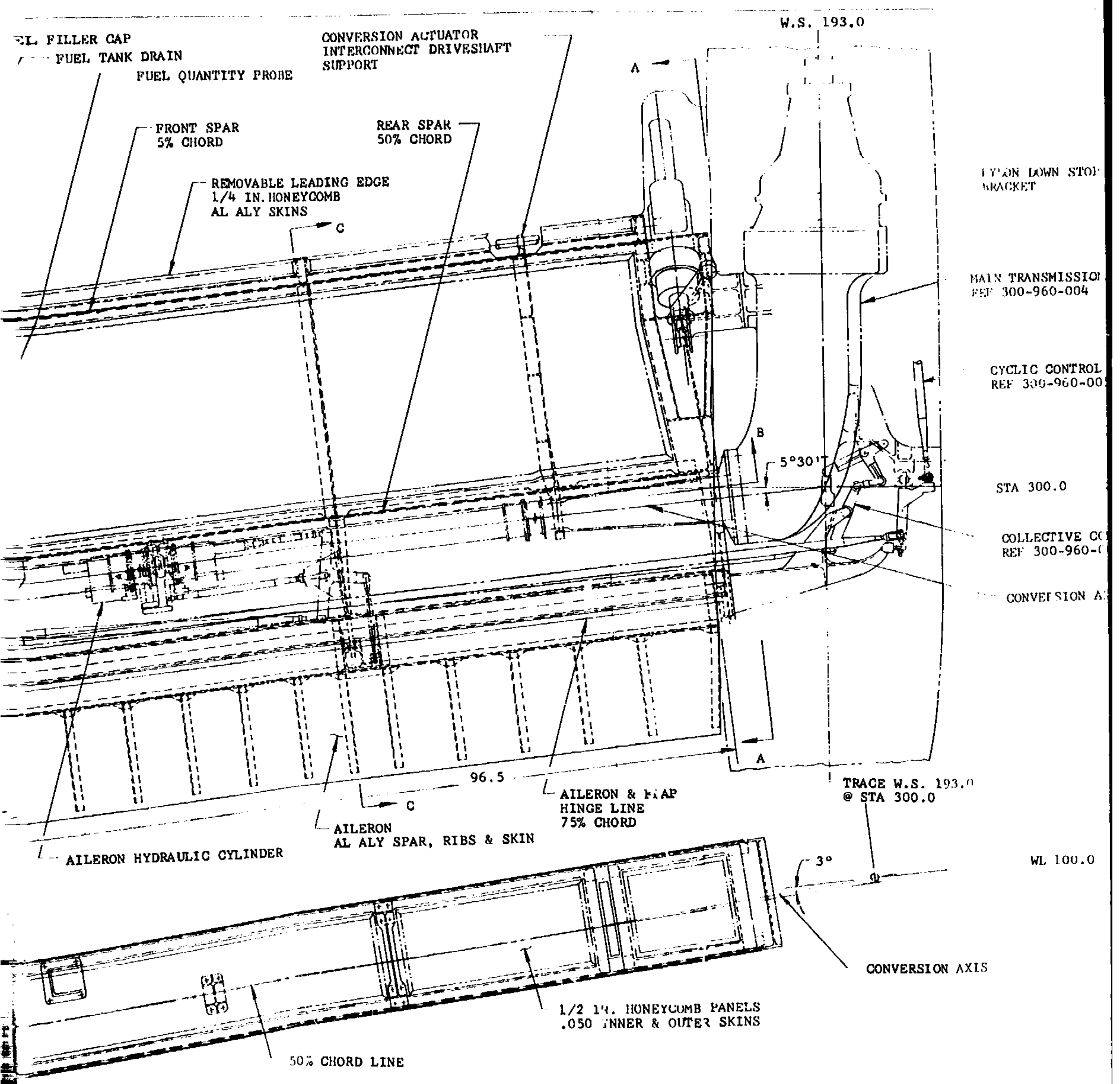
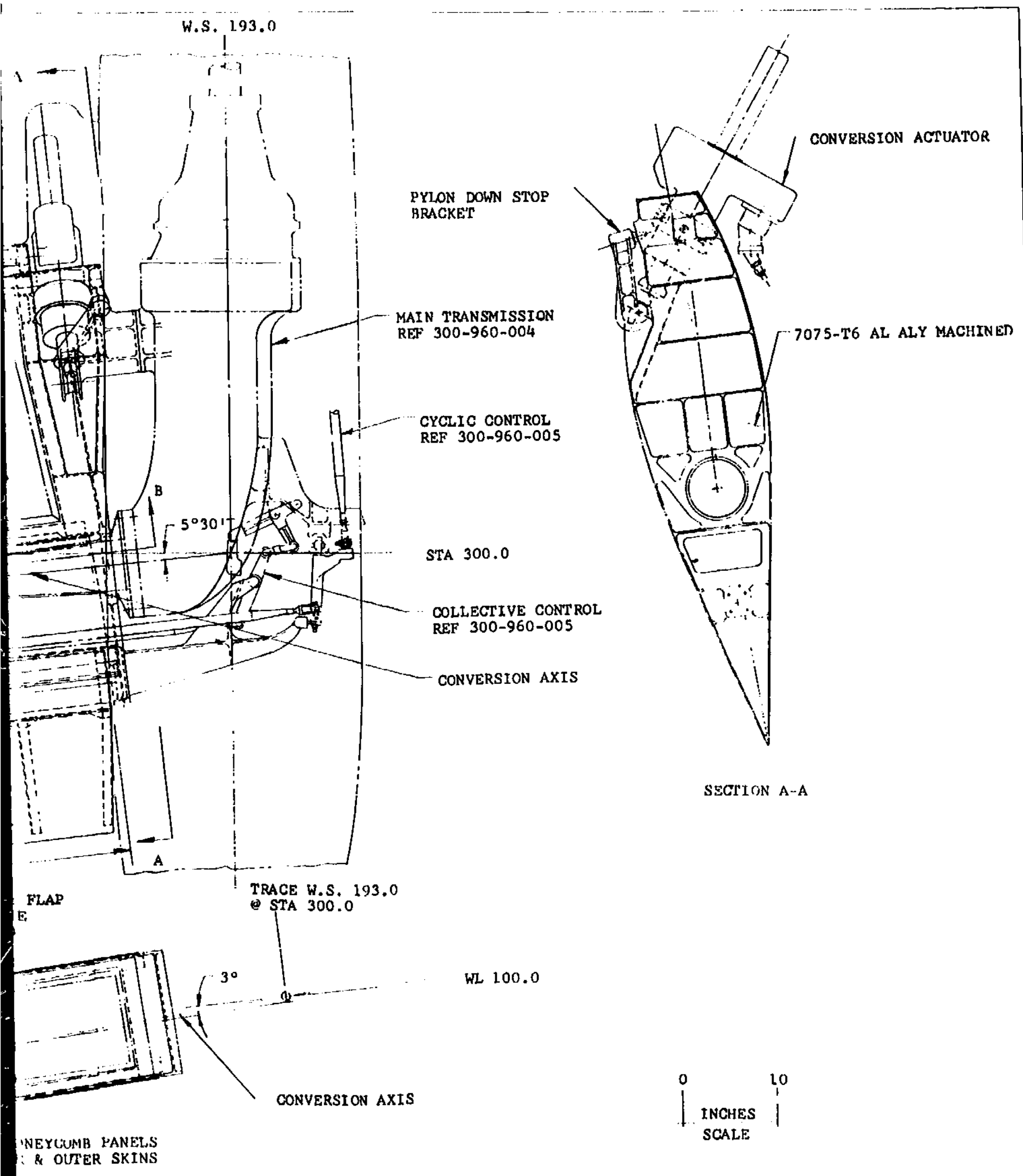

FUEL TANK DRAIN
FUEL QUANTITY PROBE
CONVERSION ACTUATOR
INTERCONNECT DRIVESHAFT
SUPPORT
W.S. 193.0
A
FRONT SPAR
5% CHORD
REAR SPAR
50% CHORD
REMOVABLE LEADING EDGE
1/4 IN. HONEYCOMB
AL ALY SKINS
C
MAIN TRANSMISSION
CYCLIC CONTROL
B
5°30'
STA 300.0
CONVERSION
A
96.5
C
AILERON & FLAP
HINGE LINE
75% CHORD
TRACE W.S. 193.0
@ STA 300.0
AILERON
AL ALY SPAR, RIBS & SKIN
AILERON HYDRAULIC CYLINDER
3°
WL 100.0
CONVERSION AXIS
1/2 IN. HONEYCOMB PANELS
.050 INNER & OUTER SKINS
50% CHORD LINE

SECTION B-B

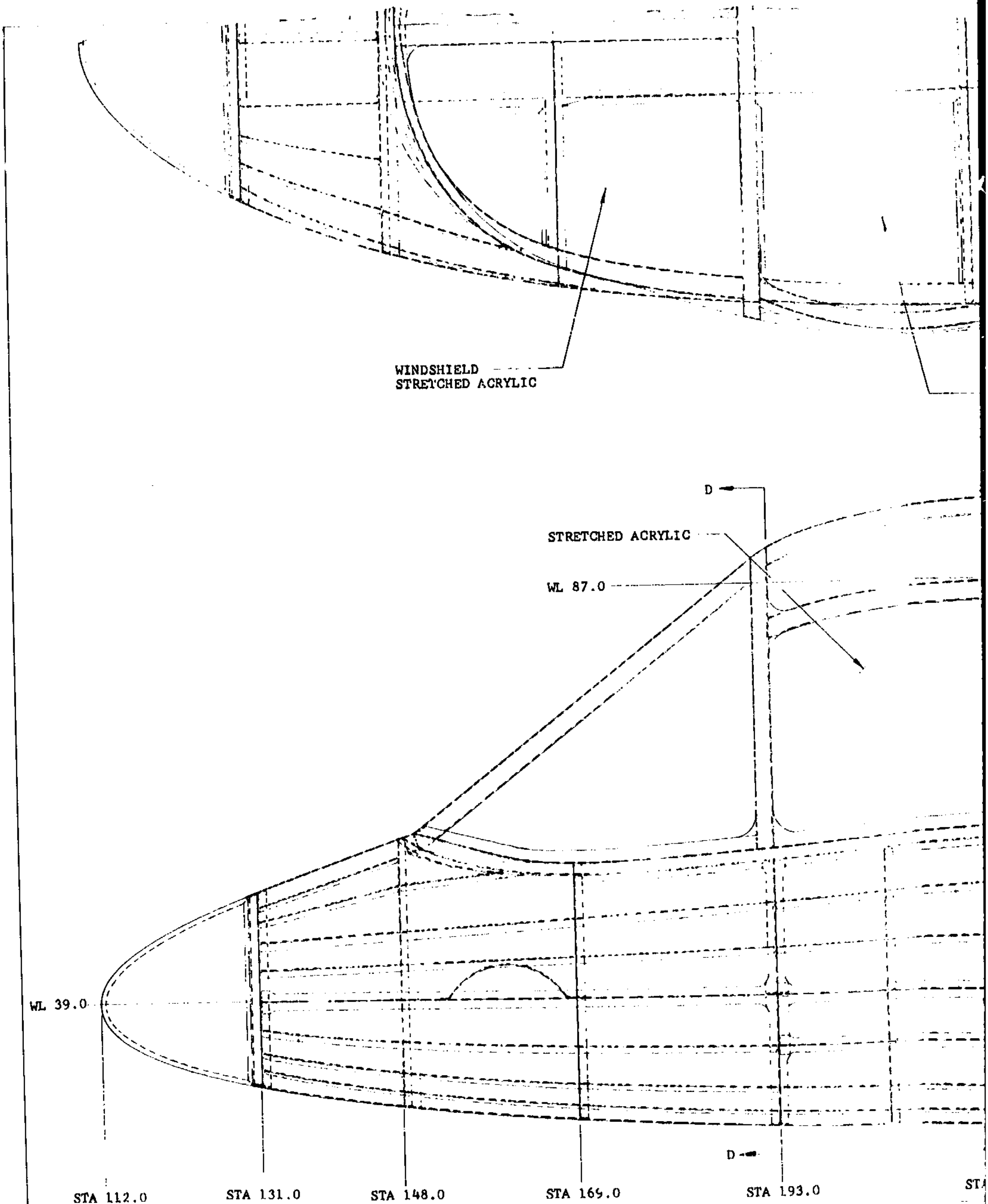


DESIGN LAYOUT

WING ASSEMBLY

G. SMITH | 8-69

300-960-007



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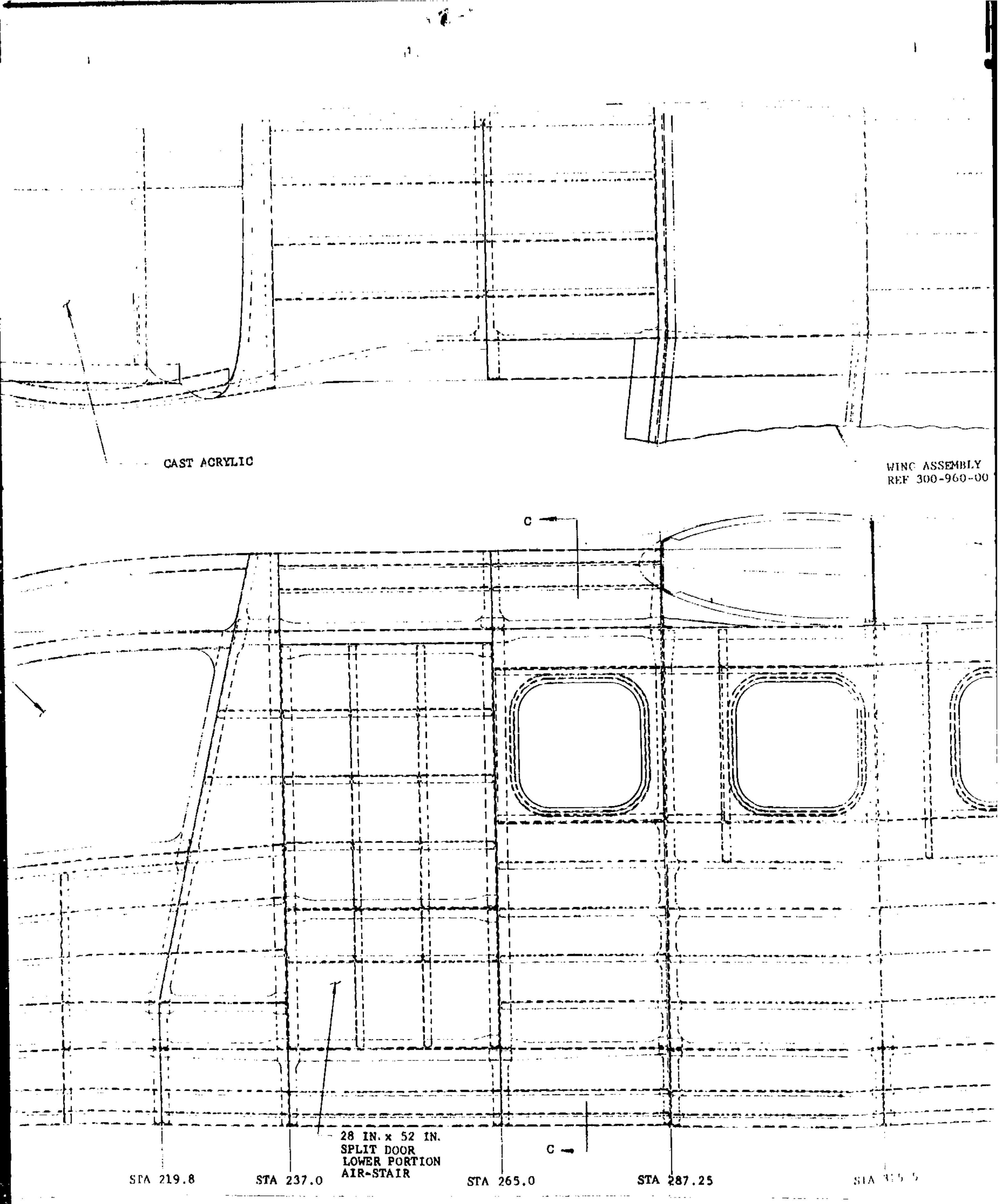

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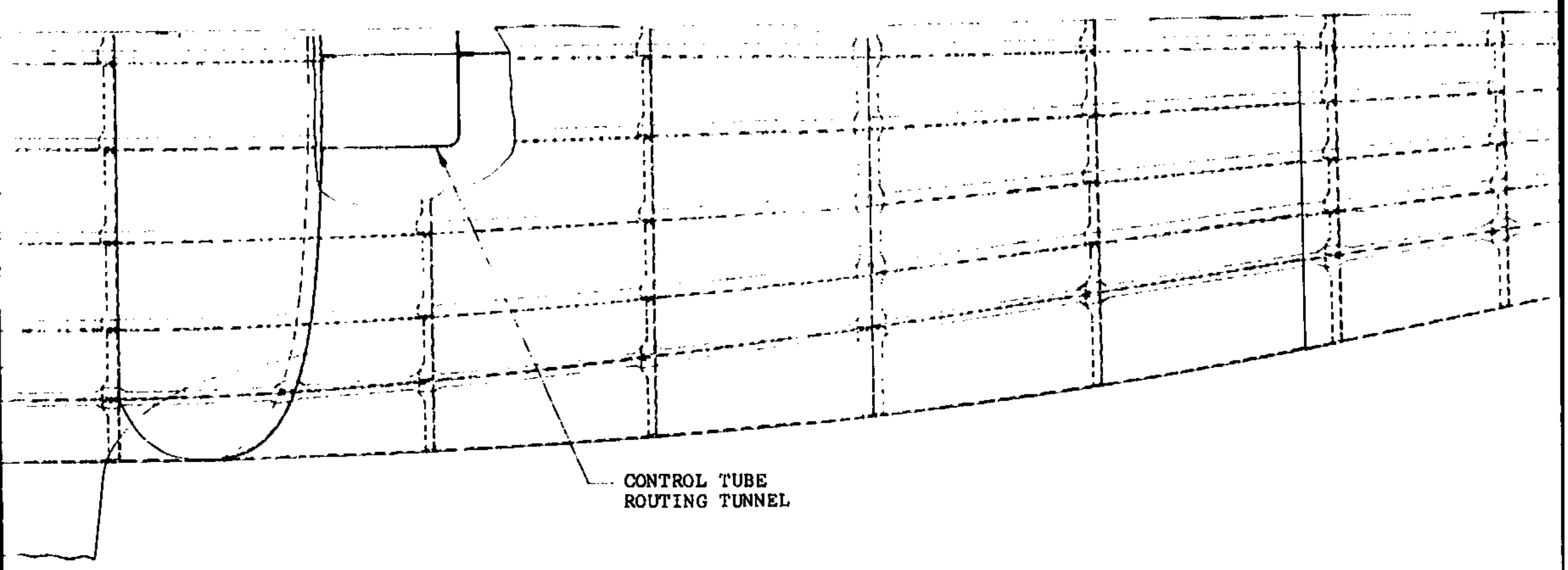

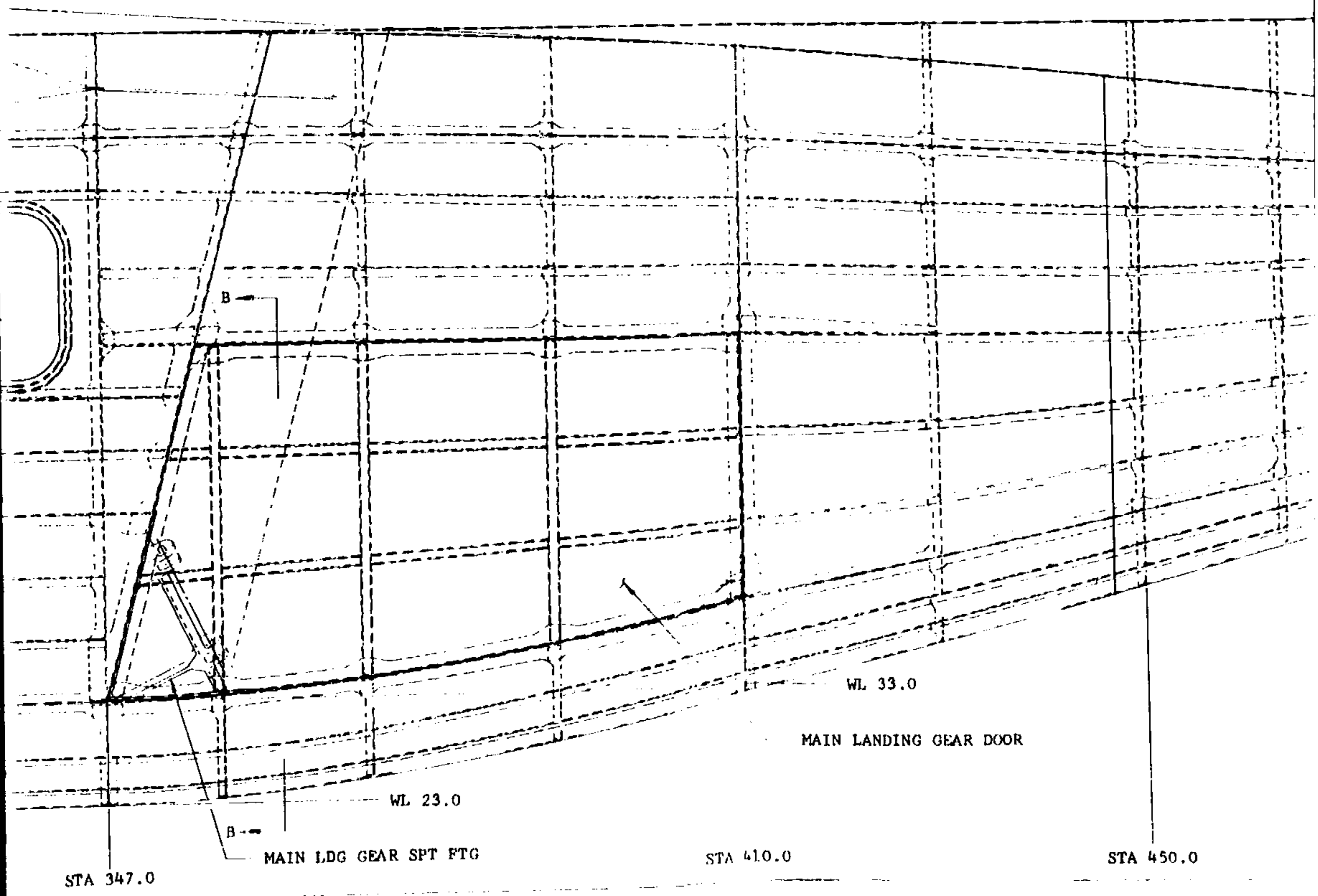

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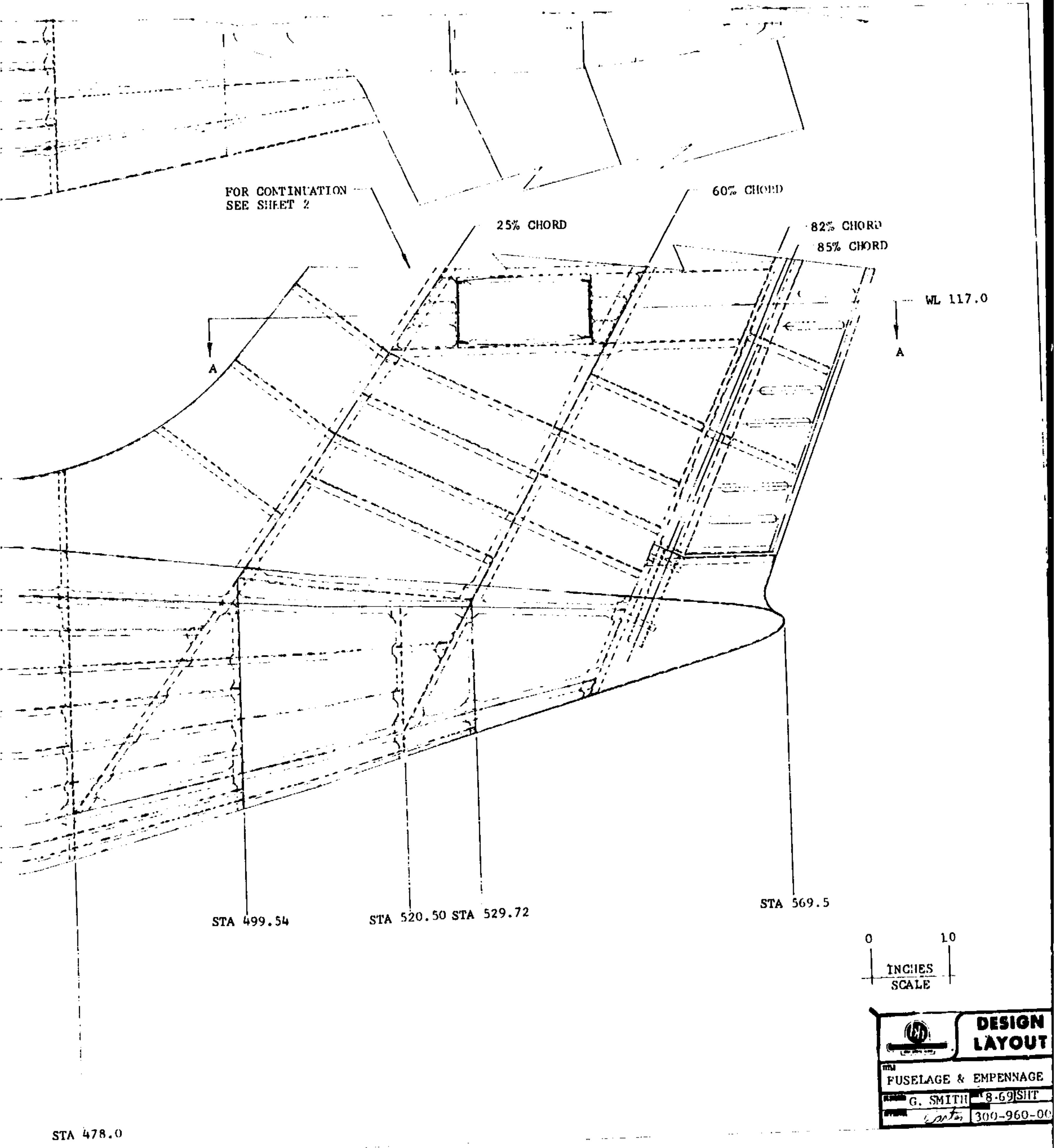
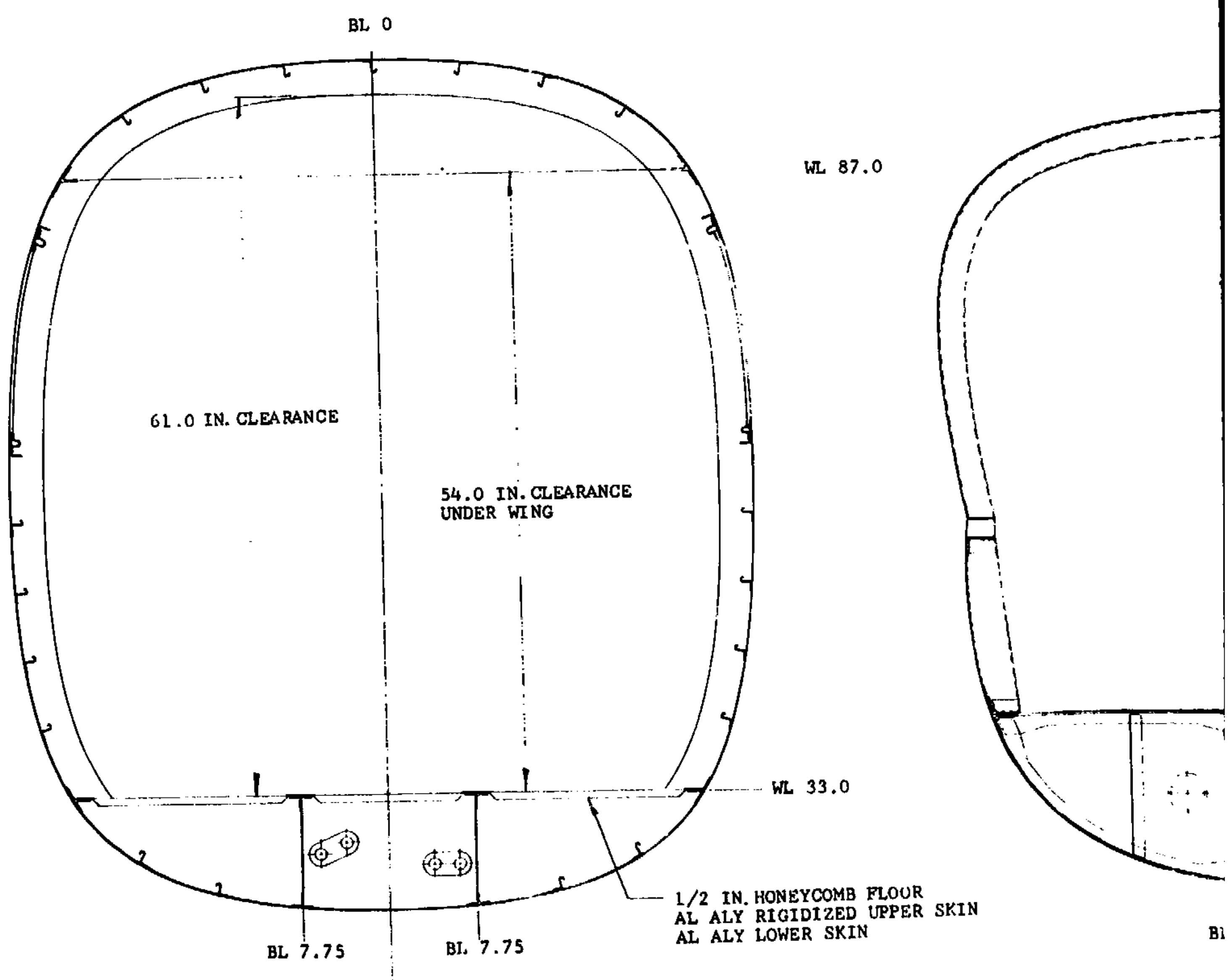


DESIGN LAYOUT

FUSELAGE & EMPENNAGE

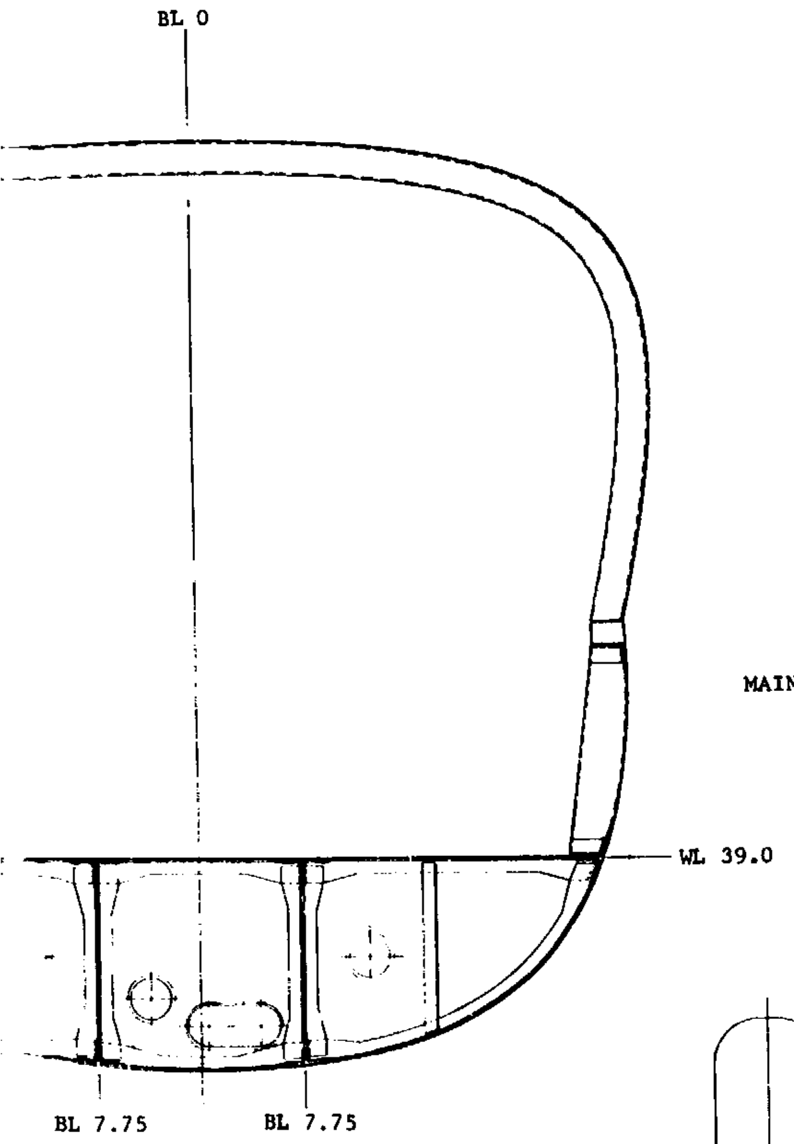
G. SMITH | 8-69 | SHT

300-960-00

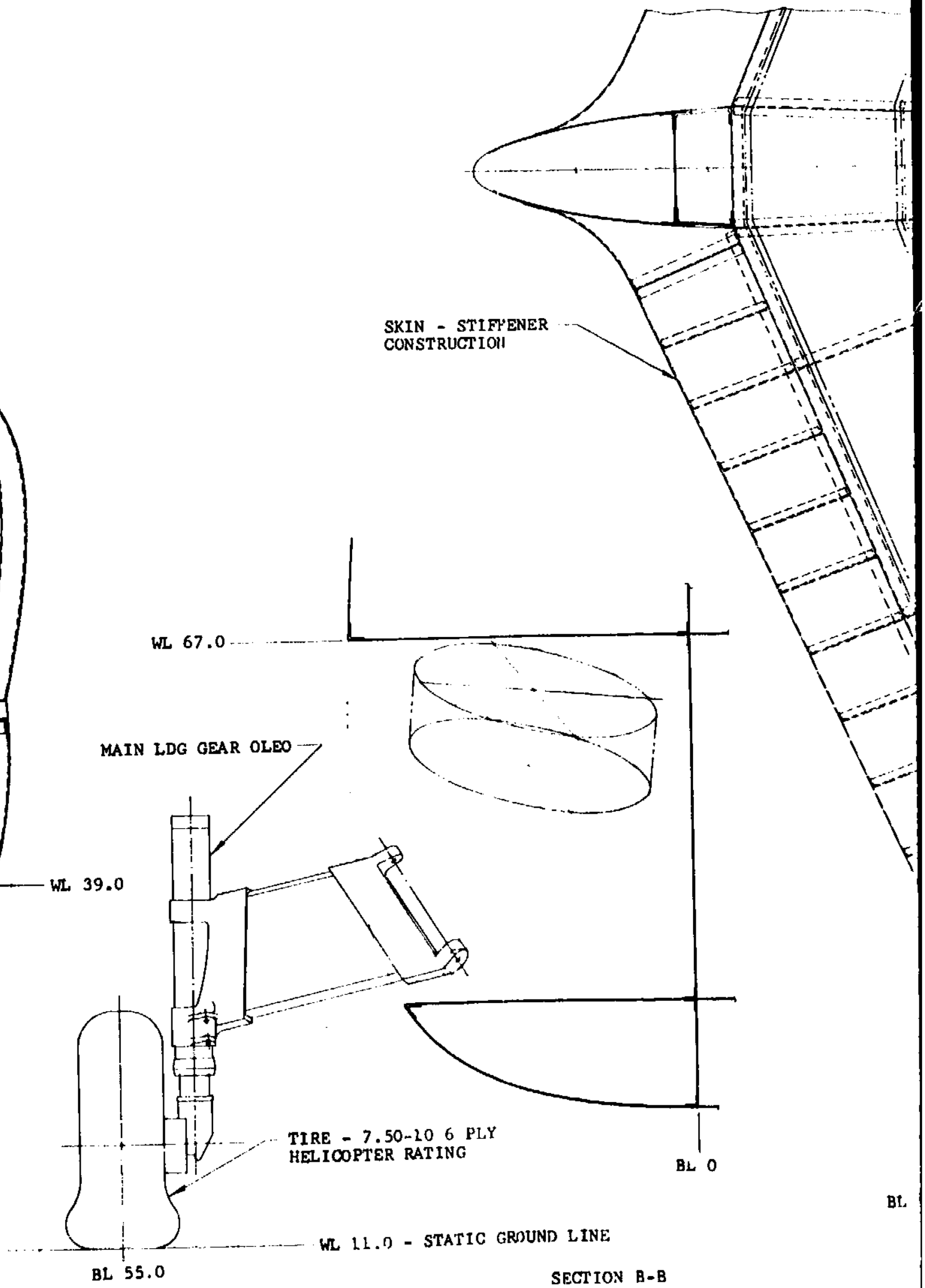


SECTION C-C

FOLDOUT FRAME

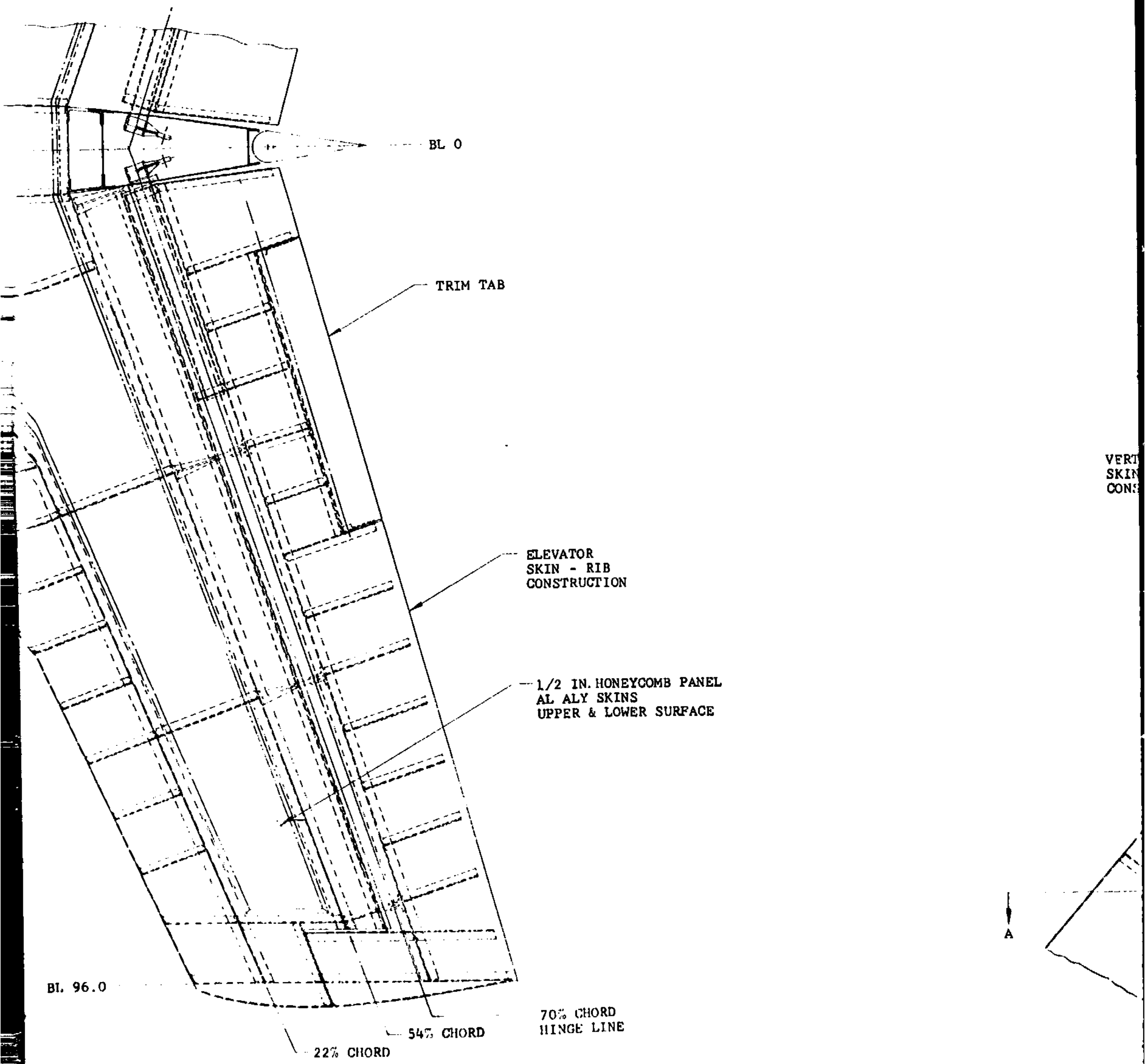


SECTION D-D

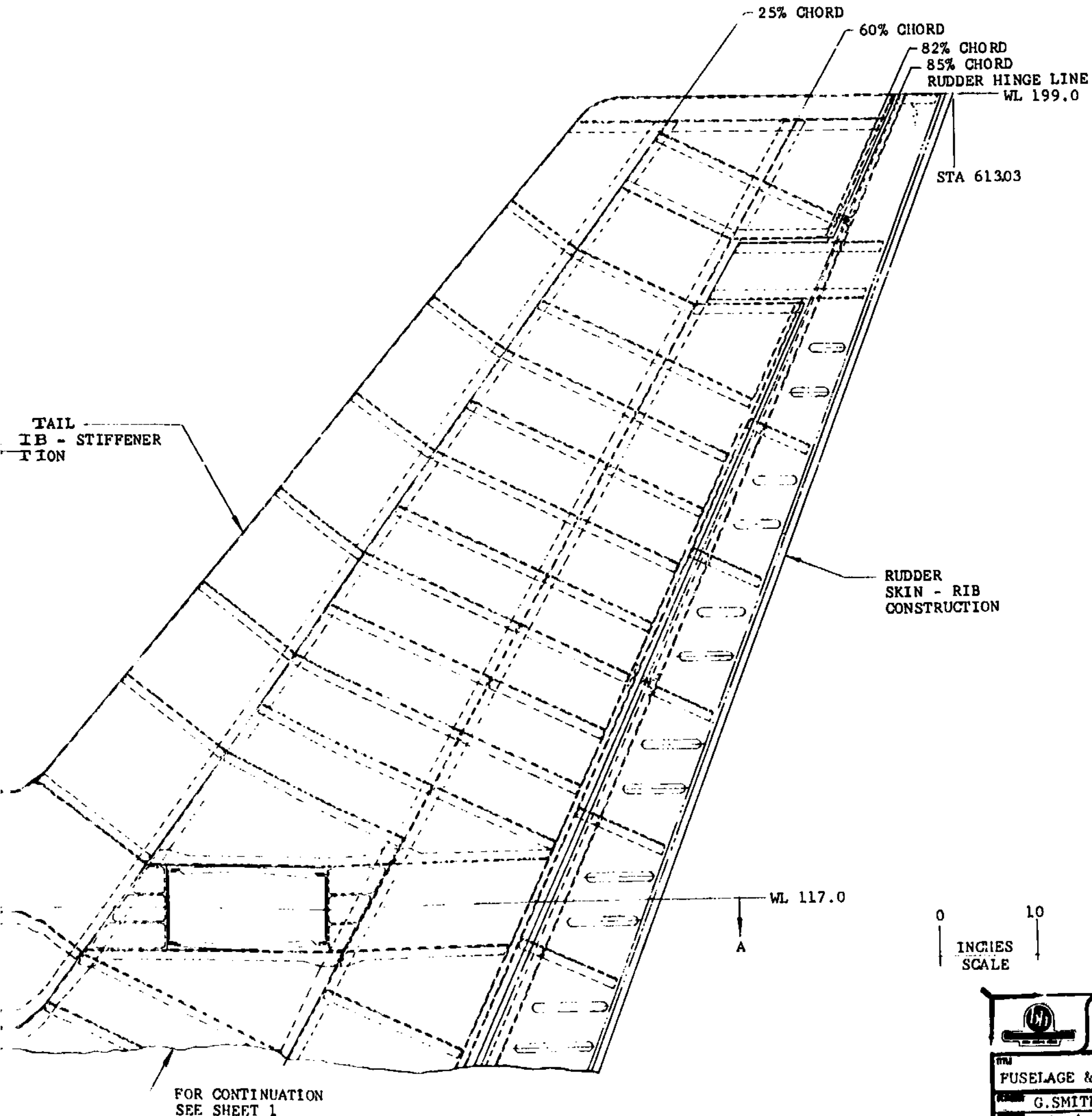


SECTION B-B

FOLDOUT FRAME


BL 0
TRIM TAB
ELEVATOR
SKIN - RIB
CONSTRUCTION
1/2 IN. HONEYCOMB PANEL
AL ALY SKINS
UPPER & LOWER SURFACE
A
BL 96.0
70% CHORD
HINGE LINE
54% CHORD
22% CHORD

SECTION A-A

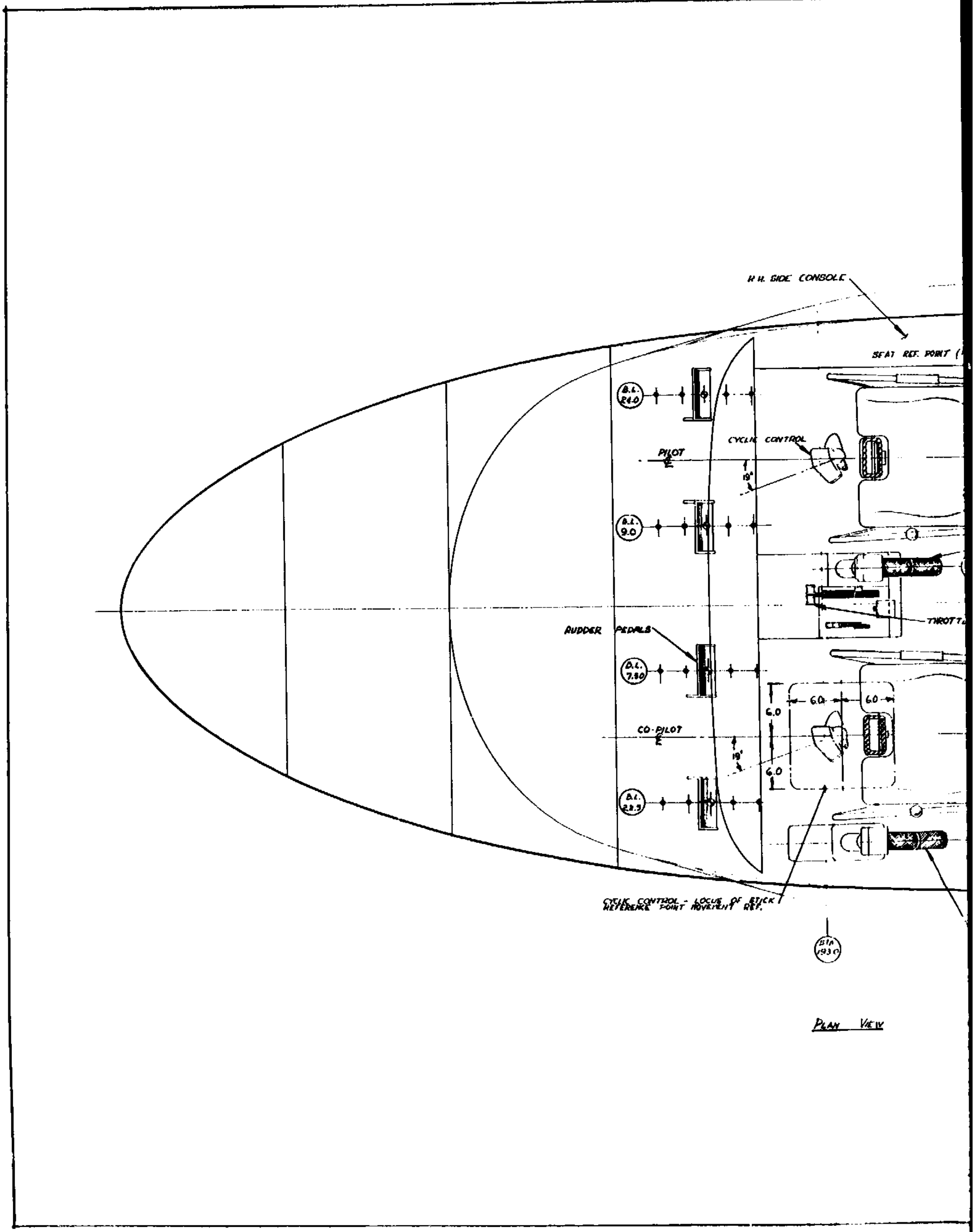


DESIGN LAYOUT

FUSELAGE & EMPENNAGE

G.SMITH | 8-69 | SHT 2

300-960-008



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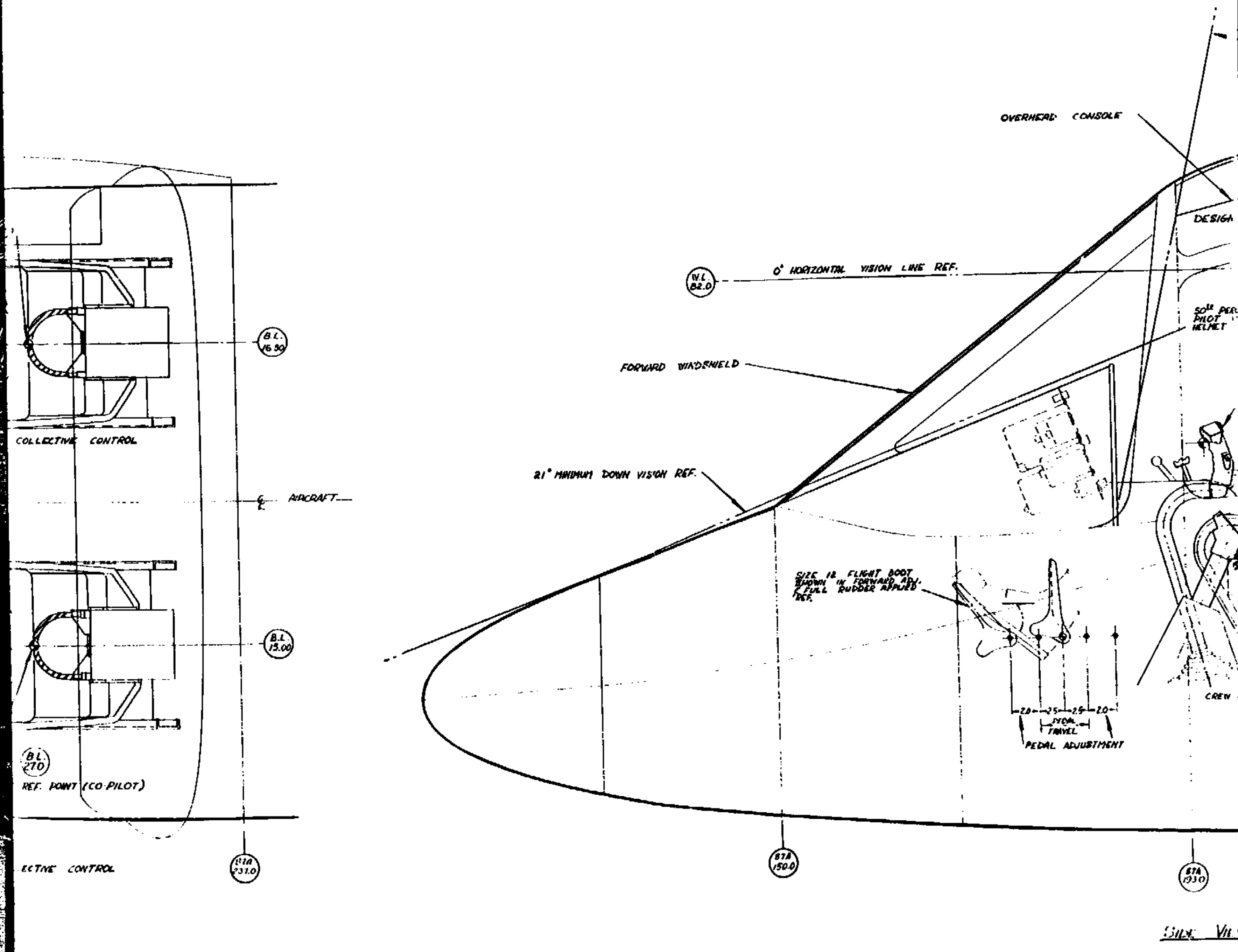
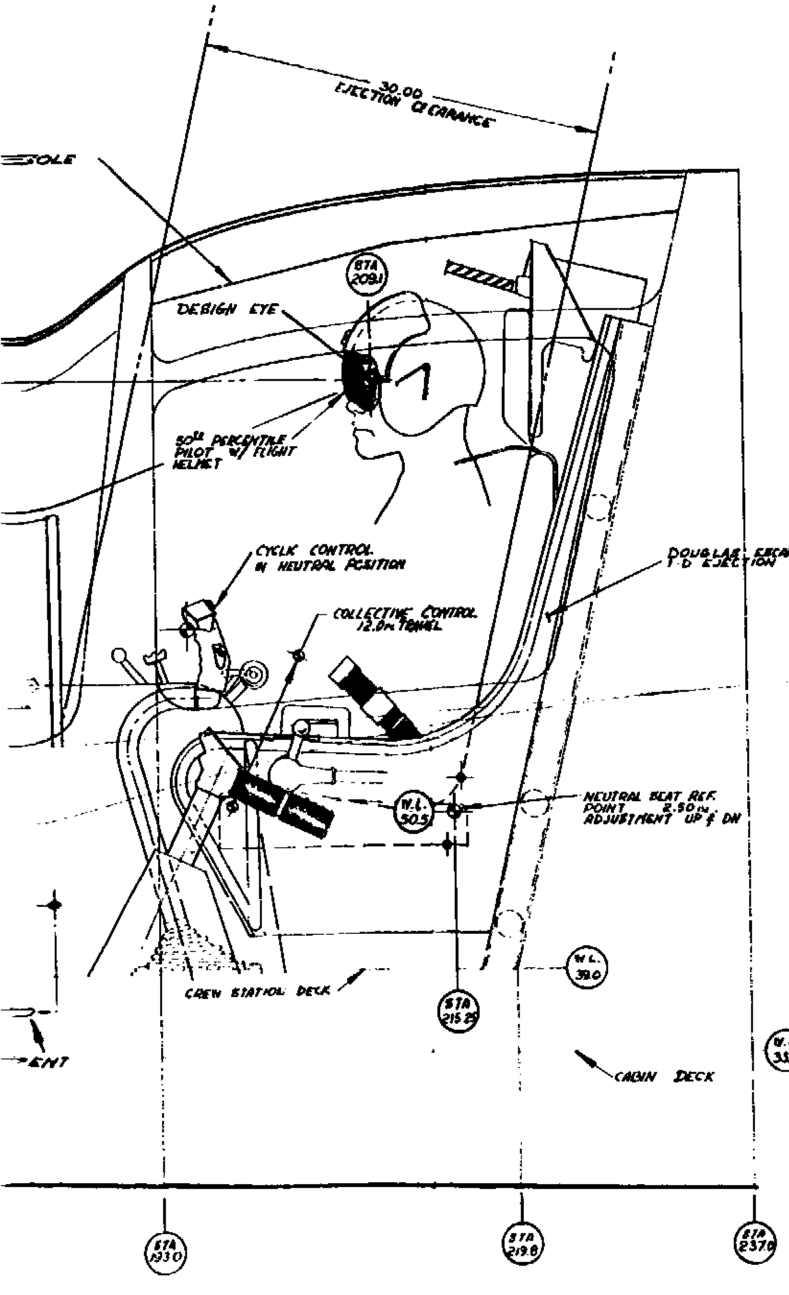
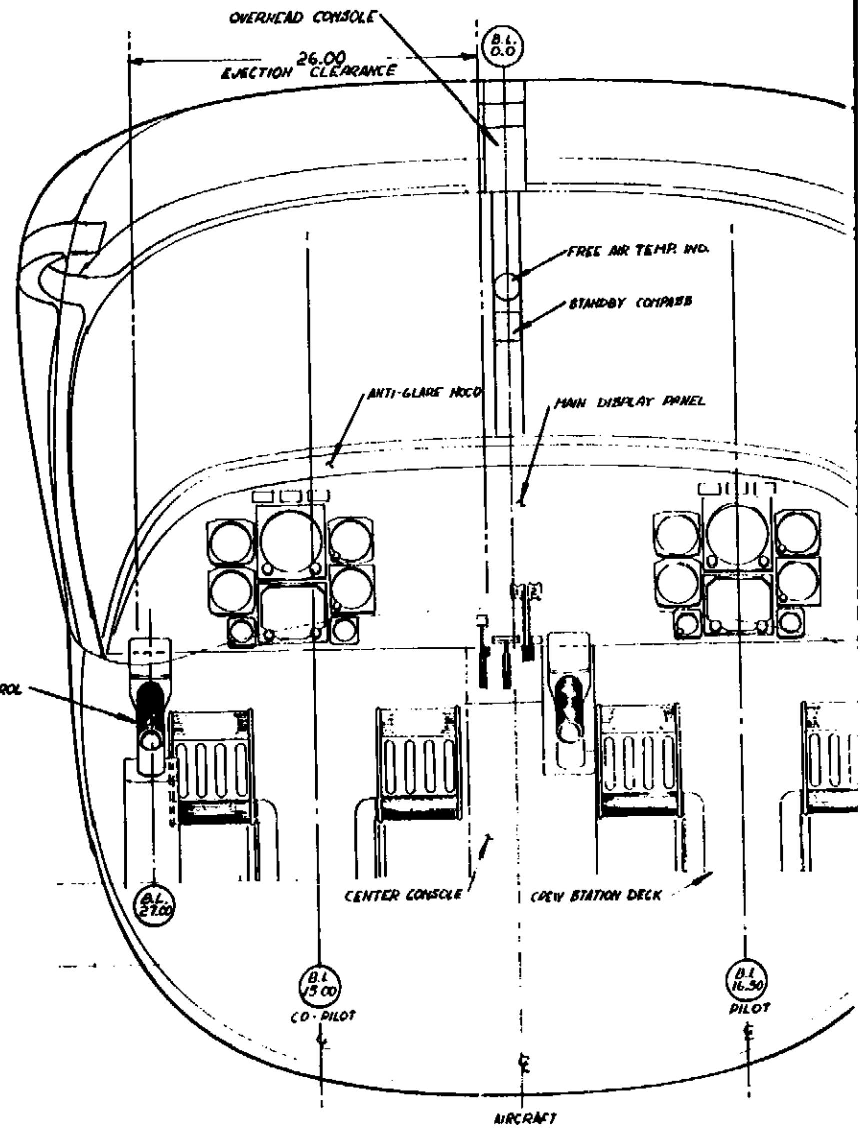
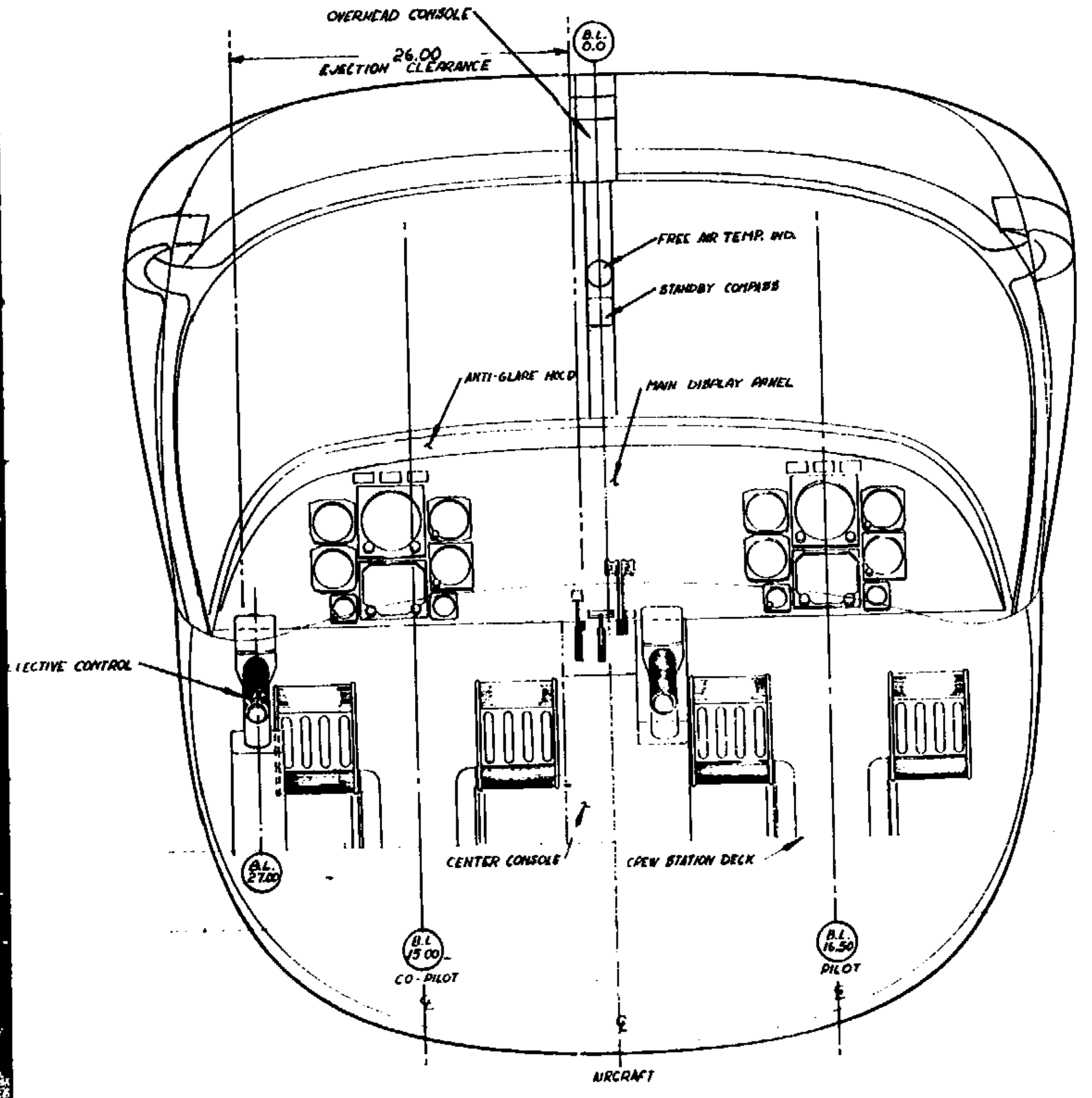

OVERHEAD CONSOLE
DESIGN
W.L. 82.0
0° HORIZONTAL VISION LINE REF.
FORWARD WINDSHIELD
21° MINIMUM DOWN VISION REF.
B.L. 16.50
COLLECTIVE CONTROL
AIRCRAFT
B.L. 15.00
B.L. 27.0
REF. POINT (CO-PILOT)
ECTIVE CONTROL
STA 237.0
SIZE 12 FLIGHT BOOT SHOWN IN FORWARD ADJ. & FULL RUDDER APPLIED REF.
2.0
2.5
2.5
2.0
PEDAL TRAVEL
PEDAL ADJUSTMENT
CREW
STA 150.0
STA 193.0
SIDE VI



SIDE VIEW



VIEW LOOKING FORWARD



VIEW LOOKING FORWARD

0 INCHES 10
SCALE

DESIGN LAYOUT

TITLE MODEL 300 CREW STATION GENERAL ARRANGEMENT

Thomas H White

SCALE 1/4

300-960-009

FOLDOUT FR...

11