## USAAVLABS TECHNICAL REPORT 67-1

# PRELIMINARY DESIGN OF A ROTOR SYSTEM FOR A HOT CYCLE HEAVY-LIFT HELICOPTER

By

J. R. Simpson

March 1967

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-225(T) Task II
HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

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This report has been prepared by Hughes Tool Company, Aircraft Division, under the provisions of Contract DA 44-177-AMC-225(T), Task II, to present the preliminary design of a hot cycle rotor system.

The report is published for the dissemination of information and the reporting of program results.

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# PRELIMINARY DESIGN OF A ROTOR SYSTEM FOR A HOT CYCLE HEAVY-LIFT HELICOPTER

## HTC-AD 66-17

by

J. R. Simpson

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Culver City, California

for

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FORT EUSTIS, VIRGINIA

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# ABSTRACT

Under the terms of Contract DA 44-177-AMC-225(T) Task II, Hughes Tool Company - Aircraft Division has completed the preliminary design study of a rotor system for a Hot Cycle Heavy-Lift Helicopter.

During the study program, extending from March 1965 to August 1966, accomplishments were as follows. An analytical procedure was developed that permits calculation of fully coupled blade response and dynamic stability characteristics. Parametric and configuration studies to reflect basic characteristics of the rotor system on the design characteristics and mission requirements were conducted. Design layouts, structural design studies, and detailed weight analyses were made. The design and analysis were limited to the integrated lift-propulsion system with emphasis on the rotor system. This effort resulted in the selection, preliminary design, and determination of performance of the optimum rotor for the heavy-lift mission requirements. Also, a fully coupled rotor dynamic analysis of the optimum rotor was made and a full-scale mockup of the rotor hub area was constructed.

The Hot Cycle heavy-lift helicopter with the selected rotor as designed exceeds the performance requirements for a 20-ton heavy-lift mission by as much as 6 tons, a 12-ton transport mission by approximately 2 tons, and a 1,500-nautical-mile ferry range by as much as 600 nautical miles. Fuel utilization (namely, ton-miles of payload per pound of fuel) proved to be outstanding.

# FOREWORD

This report was prepared in accordance with Task II of Contract DA 44-177-AMC-225(T) for the U. S. Army Aviation Materiel Laboratories. The contract became effective on 17 March 1965. Work was completed on 31 August 1966. The report summarizes the preliminary design program, including the parametric studies and an integrated preliminary design.

The work was accomplished by Hughes Tool Company - Aircraft Division in Culver City, California, under the direction of Mr. H. O. Nay, Director of Aeronautical Engineering, and Mr. C. R. Smith, Manager, Hot Cycle Department, and under the direct supervision of Mr. J. R. Simpson, Project Engineer, Hot Cycle Heavy-Lift Helicopter.

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## LIST OF SYMBOLS

| Symbol | Identity | Units |
|---|---|---|
| A | Area (rotor disc area) | sq ft |
| $A_b$ | Blade area | sq ft |
| $A_D$ | Duct area | sq in. |
| AE | Available energy | BTU/lb |
| B | Coefficient determined from detailed layout weights | nondimensional |
| b | Span or number of blades, as applicable | ft |
| C | Chord | in. |
| c | Chord length | in. |
| $\bar{c}$ | Nondimensional coefficient $\left(\frac{c}{45}\right)$ | nondimensional |
| $C_L$ | Lift coefficient | nondimensional |
| cf | Centrifugal force | lb |
| $C_p$ | Specific heat at constant pressure | nondimensional |
| $C_r$ | Chord at root | in. |
| $C_t$ | Chord at tip | in. |
| $C_v$ | Velocity coefficient | in. |
| $C_1$ and $C_2$ | Constants used in range computations | nondimensional |
| cg | Center of gravity | nondimensional |
| D | Drag | lb |
| $D_h$ | Hydraulic diameter | in. |
| E | Modulus of elasticity | lb/in.$^2$ |
| f | Design stress factor or friction coefficient, as applicable | nondimensional |
| fps | Feet per second | |

| Symbol | Identity | Units |
|---|---|---|
| G | Shear modulus of elasticity | lb/in.$^2$ |
| g | Gravity | in./sec$^2$ |
| H | Altitude in nautical miles | nmi |
| I | Area moment of inertia | in.$^4$ |
| J | Torsional stiffness parameter | in.$^4$ |
| K | Kinetic energy index | ft-lb/sec |
| KN, kn | Knots | |
| $\ell$ | 20-percent radius station | |
| LE | Leading edge | |
| M | Mach number | nondimensional |
| m | Exponent determined from statistical data | nondimensional |
| N | Number of engines | |
| n | Ultimate load factor, exponent determined from statistical data or station location, as applicable | nondimensional |
| NMI, nmi | Nautical miles | |
| OGE | Out of ground effect | |
| P | Pressure | lb/sq ft |
| $P_T$ | Total pressure | lb/sq ft |
| psi | Pounds per square inch | |
| psig | Pounds per square inch gage | |
| q | Dynamic pressure | lb/sq ft |
| $q_0$ | Dynamic pressure in free stream | lb/sq ft |
| R | Rotor radius or gas constant for air, as applicable | ft, 53.35 ft-lb/ $^o$R/lb |
| $R_{sp}$ | Specific range | nmi/lb of fuel |
| rhp | Rotor horsepower | hp |

| Symbol | Identity | Units |
|---|---|---|
| rpm | Revolutions per minute | |
| r | Radius of an element | ft |
| SFC | Specific fuel consumption | lb/hr rhp |
| SL | Sea level | |
| STOL | Short takeoff and landing | |
| T | Rotor thrust or temperature, as applicable | lb or $^{o}$R |
| TE | Trailing edge | |
| t | Thickness | in. |
| $\bar{t}$ | Thickness ratio | $\frac{\%c}{100}$ |
| $t_r$ | Thickness at root | in. |
| $t_t$ | Thickness at tip | in. |
| $V_j$ | Jet velocity | ft/sec |
| $V_{ne}$ | Design maximum level flight speed | knots |
| $V_T$, $V_t$ | Tip velocity | ft/sec |
| W | Weight | lb |
| $W_8$ | Flow at exhaust | lb/sec |
| $W_{ac}$ | Weight of air conditioning and anti-icing group | lb |
| $W_{BU}$ | Weight of ideal blade | lb |
| $W_b$ | Weight of fuselage | lb |
| $W_c$ | Weight of cargo handling equipment | lb |
| $W_{cff}$ | Weight of cruise fan system - fixed | lb |
| $W_{cfr}$ | Weight of cruise fan system - removable | lb |
| $W_e$ | Empty weight | lb |
| $W_{el}$ | Weight of electrical group | lb |
| $W_{en}$ | Weight of electronics group | lb |

| Symbol | Identity | Units |
|---|---|---|
| $W_f$ | Fuel flow | lb/hr |
| $W_{fc}$ | Weight of flight controls | lb |
| $W_{fe}$ | Weight of furnishings and equipment | lb |
| $W_{f_t}$ | Additional increment of fuel for climb | lb |
| $W_g$ | Gross weight | lb |
| $W_h$ | Weight of hydraulic and pneumatic equipment | lb |
| $W_{hy}$ | Weight of hover - yaw control group | lb |
| $W_i$ | Weight of instruments and navigational equipment | lb |
| $W_{lg}$ | Weight of alighting gear | lb |
| $W_{pp}$ | Weight of propulsion group | lb |
| $W_r$ | Weight of main rotor group | lb |
| $W_s$ | Weight at start of climb | lb |
| $W_{scf}$ | Weight of surface controls | lb |
| $W_{tg}$ | Weight of tail group | lb |
| $W_w$ | Weight of wing group | lb |
| $\gamma$ | Ratio of specific heat at constant pressure to the specific heat at constant volume | nondimensional |
| $\eta$ | Propulsive efficiency | nondimensional |
| $\sigma$ | Solidity | nondimensional |
| $\psi$ | Blade azimuth angle | deg |

The following symbols are applicable to Appendix III, Rotor Blade Equations:

| Symbol | Identity | Units |
|---|---|---|
| $\bar{d}$ | Inboard design station to end of reduced chord section | $\frac{\%R}{100}$ |
| $\bar{d}_1$ | Inboard design station to outboard design station | $\frac{\%R}{100}$ |
| $\bar{d}_2$ | Outboard design station to tip | $\frac{\%R}{100}$ |
| $\bar{d}_3$ | Inboard design station to lead-lag hinge | $\frac{\%R}{100}$ |
| $\bar{e}_f$ | Flap hinge offset | $\frac{\%R}{100}$ |
| $\bar{e}_L$ | Lag hinge offset | $\frac{\%R}{100}$ |
| $F_{BU}$ | Allowable ultimate bending stress | lb/in.$^2$ |
| $F_e$ | Endurance limit at 0 mean stress | lb/in.$^2$ |
| $F_{TCO}$ | Steady stress at knee of Goodman diagram | lb/in.$^2$ |
| $F_{CYCO}$ | Oscillatory stress at knee of Goodman diagram | lb/in.$^2$ |
| $F_{TU}$ | Allowable ultimate tensile stress | lb/in.$^2$ |
| $f_{CF}$ | Stress due to centrifugal force | lb/in.$^2$ |
| $\bar{h}$ | Helicopter center of gravity to rotor centerline | $\frac{\%R}{100}$ |
| $I_e$ | Blade flapping mass moment of inertia about flapping hinge | in. -lb-sec$^2$ |
| $I_p$ | Section polar mass moment of inertia about center of gravity | $\frac{\text{in. -lb-sec}^2}{\text{in.}}$ |

| Symbol | Identity | Units |
|---|---|---|
| $I_x$ | Flapwise structural moment of inertia | $in.^4$ |
| $I_y$ | Chordwise structural moment of inertia | $in.^4$ |
| $J$ | Section torsional stiffness parameter | $in.^4$ |
| $K_c$ | $\frac{\text{Chordwise moment}}{(W/b)\,(R)\,(C_{11})}$ | nondimensional |
| $K_F$ | $\frac{\text{Flapwise moment}}{(W/b)\,(R)\,(F_{13})}$ | nondimensional |
| $K_2$ | $\frac{\bar{x}_c - \bar{x}_f}{\bar{x}_r - \bar{x}_c}$ | nondimensional |
| $K_3$ | $w_{NB} \quad \frac{\bar{x}_{NB} - \bar{x}_c}{\bar{x}_R - \bar{x}_c}$ | lb/in. |
| $K_4$ | $1 + K_2$ | nondimensional |
| $\bar{l}$ | Flap hinge to inboard design station | $\frac{\%R}{100}$ |
| $\bar{l}_1$ | Flap hinge to end of reduced chord station | $\frac{\%R}{100}$ |
| $\bar{l}_2$ | Flap hinge to outboard design station | $\frac{\%R}{100}$ |
| $p$ | Static pressure | $lb/in.^2$ |
| $po$ | Total pressure × Design factor | $lb/in.^2$ |
| $T$ | Duct gas temperature | deg. F |
| $\bar{t}_F$, $\bar{t}_R$ | Spar depth | $\frac{\%C}{100}$ |
| $\bar{t}_t$, $\bar{t}_2$ | Airfoil thickness | $\frac{\%C}{100}$ |

| Symbol | Identity | Units |
|---|---|---|
| $\bar{t}_{SK}$ | Blade skin thickness | .01 in. |
| V | Ground wind | kn |
| W | Helicopter gross weight | lb |
| $W_{BU}$ | Total blade weight | lb |
| $\bar{W}_{BU}$ | $\frac{W_{BU}}{12R}$ | lb/in. |
| $W_{CA}$ | Cascade weight | lb |
| $\bar{W}_{CA}$ | $\frac{W_{CA}}{12R}$ | lb/in. |
| $W_t$ | Tip weight | lb |
| $\bar{W}_t$ | $\frac{W_t}{12R} X'_t$ | lb/in. |
| $w_{NB}$ | Weight of nonbending material | lb/in. |
| $w_R$, $w_F$ | Weight of spars (rear, front) | lb/in. |
| $w_T$ | Weight of total section | lb/in. |
| X | Distance from rotor centerline | $\frac{\%R}{100}$ |
| $X_A$ | Lift centroid to rotor centerline distance | $\frac{\%R}{100}$ |
| $\bar{X}_C$ | Blade section center of gravity location | $\frac{\%C}{100}$ |
| $\bar{X}_{Ct}$ | Blade section center of gravity location at the tip | $\frac{\%C}{100}$ |
| $\bar{X}_{CA}$ | Center of gravity location of cascade | $\frac{\%C}{100}$ |
| $\bar{X}_F$, $\bar{X}_R$ | Spar center of gravity locations | $\frac{\%C}{100}$ |

| Symbol | Identity | Units |
|---|---|---|
| $\bar{X}_{NB}$ | Nonbending material center of gravity location | $\frac{\%C}{100}$ |
| $\bar{X}_t$ | Center of gravity location of tip weight | $\frac{\%C}{100}$ |
| $X'_t$ | Center of gravity location of tip weight (flapwise) | $\frac{\%R}{100}$ |
| $\delta$ | Structure density | lb/in.$^3$ |
| $\lambda$ | $\frac{\text{Control moment - spring}}{\text{Control moment - lateral force}}$ | nondimensional |
| $\bar{\rho}_F$, $\bar{\rho}_R$ | Spar radii of gyration (area) | $\frac{\%C}{100}$ |
| $\bar{\rho}_{NB_o}$, $\bar{\rho}_{NB_i}$ | Nonbending material polar radius of gyration (mass) | $\frac{\%C}{100}$ |

Figure 1. Hot Cycle Heavy-Lift Helicopter Concept.

# SUMMARY

A parametric study and preliminary design program has defined the configuration and characteristics of a rotor for a 12- to 20-ton heavy-lift helicopter utilizing the Hot Cycle propulsion system. The objectives of the program were as follows:

1. Develop an analytic procedure that will permit calculation of fully coupled blade loads and dynamic stability characteristics.
2. Conduct parametric and configuration studies to determine the optimum Hot Cycle rotor system for a 12- to 20-ton-payload heavy-lift helicopter and investigate, on a limited basis, the features required to increase its cruise speed by a substantial amount.
3. Complete the preliminary design of the selected optimum rotor, including design layouts, structural design and weight analysis, stability and control studies, and static and dynamic loads analysis.
4. Construct a full-scale mockup of the rotor hub.

To accomplish the above objectives, computer programs were developed for the fully coupled rotor dynamic analysis and the parametric study. For the analysis, a digital computer program that has the capability of solving the full range of helicopter rotor dynamic problems was developed and checked against flight test data. A nonlinear representation of blade loads, including lift and moment hysteresis, is incorporated in the program to provide a more realistic analysis of fully coupled blade loads in forward flight. The development of this program has been summarized and previously submitted (Reference 1). For the parametric study, a computer program to determine the optimum rotor was developed to consider the effect of variables such as blade radius, chord, thickness, tip speed, blade spar location, duct shape, and aircraft configuration. Development and results of this program have been previously reported (Reference 2).

The results of the parametric study were reviewed, and a rotor was selected that was considered most nearly optimum for all the aircraft configurations studied. The selected rotor is a three-bladed, fully articulated rotor with 90-foot diameter and 60-inch chord. The study also included, in addition to the articulated rotors, configurations with

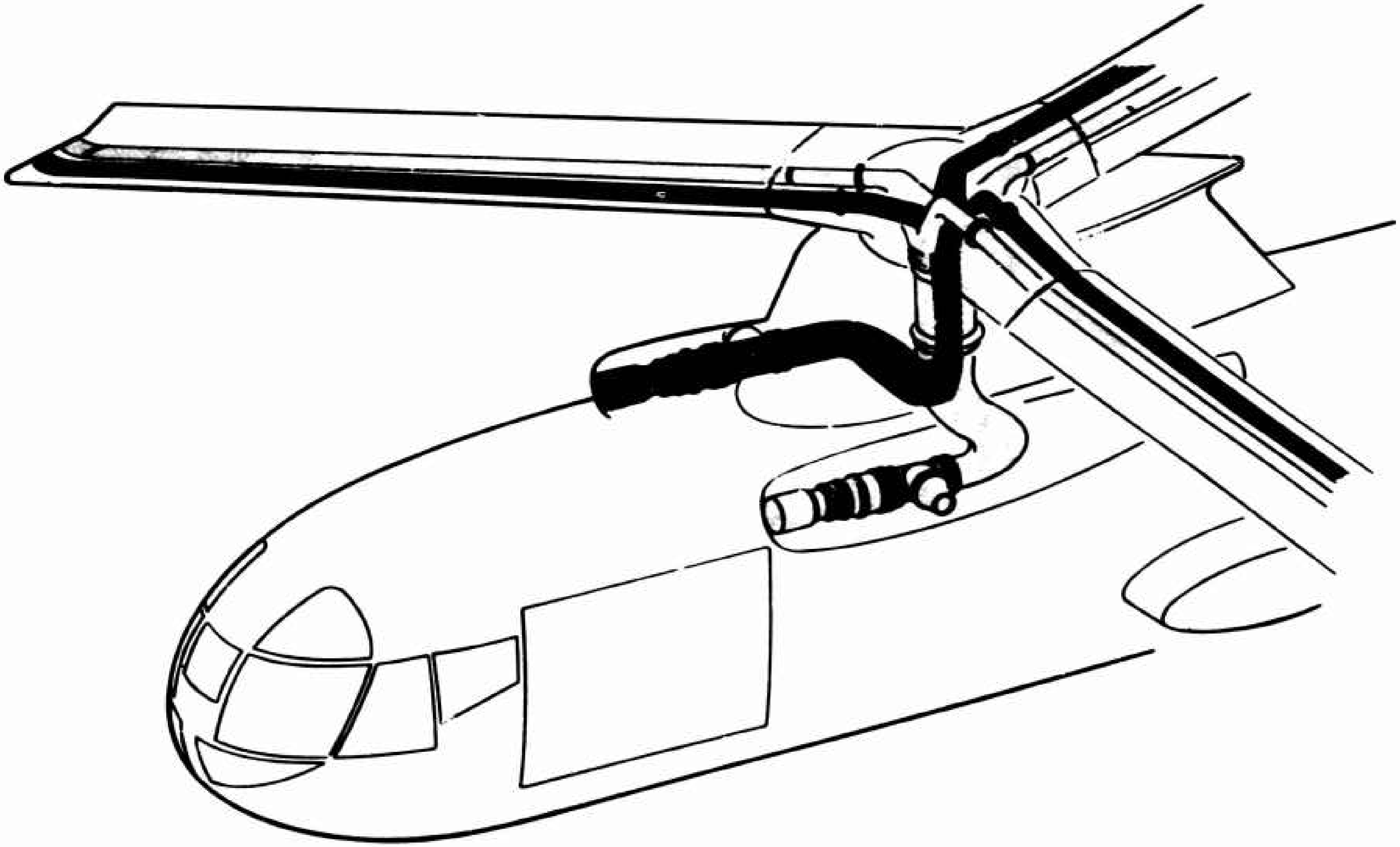


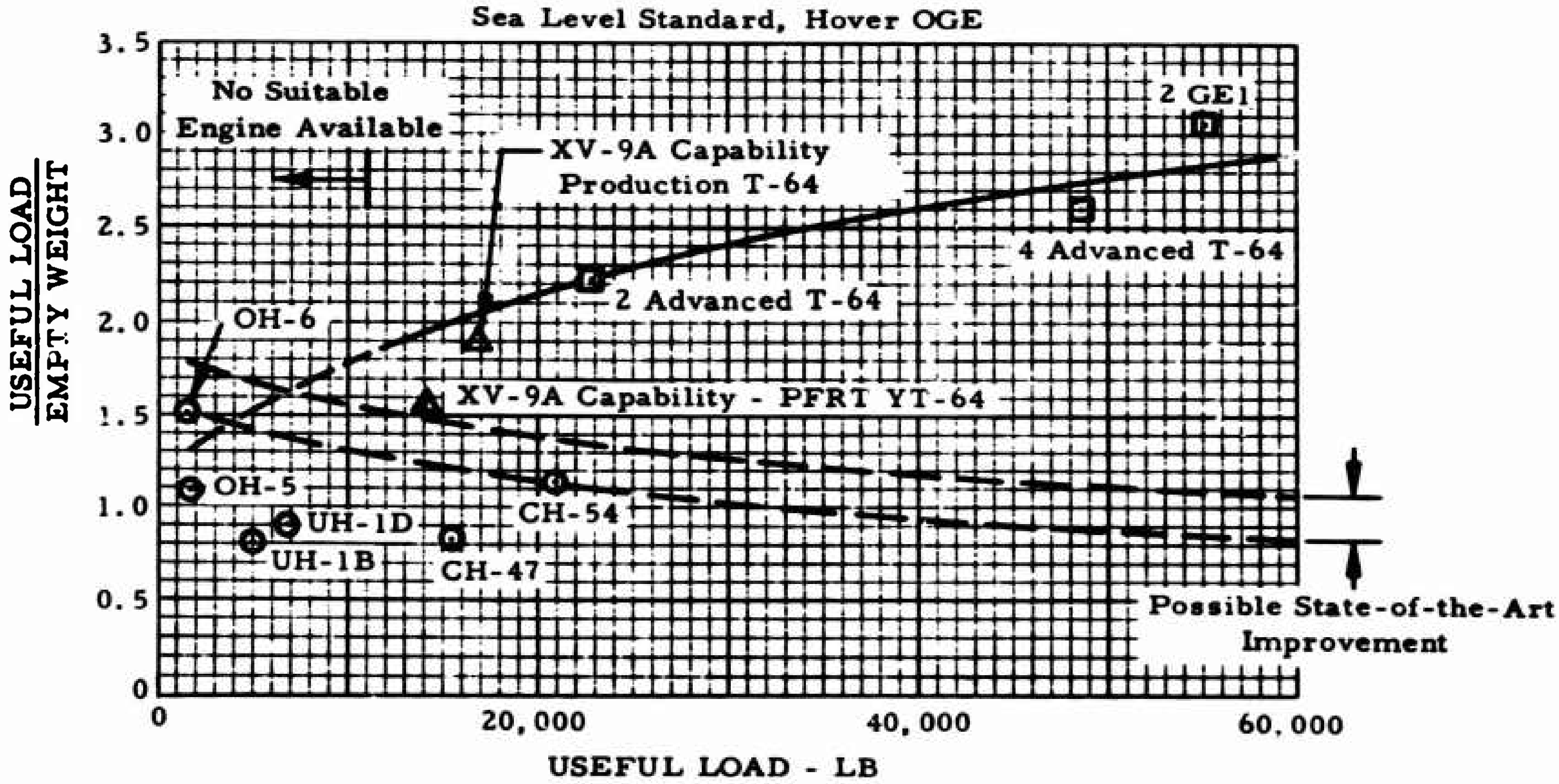
Figure 2. Propulsion System.

in-plane chordwise restraint (akin to a rigid rotor). The rigid type of rotor investigated weighed almost twice as much as the articulated rotor of the same size. Design layouts, structural design and weight analysis, and stability and control studies were completed on this selected rotor. The basic characteristics of this rotor are shown in Table IX in the Rotor Section.

The rotor is powered by the Hot Cycle propulsion system. As shown in Figure 2, the Hot Cycle system transmits power pneumatically by lightweight ducting that directs high-energy gas from turbine engines to the rotor blade tips to drive the rotor as a large reaction turbine. The Hot Cycle rotor is suited to the transport and heavy-lift missions of 12 to 20 tons and up. The favorable characteristics of this rotor are the direct result of the simplicity and light weight inherent in the Hot Cycle propulsion system, which eliminates the weight and complexity of power turbines, shafts, large gearboxes, and clutches. Since there is no rotor shaft drive torque reaction on the fuselage, there is no need for a large antitorque tail rotor; directional control is provided by a small yaw fan located in the vertical stabilizer. The resulting low empty weight, and thus high payload to empty weight ratio, cannot be attained by the conventional shaft-driven rotors with their inherently heavier complex dynamic components. A plot of useful load/empty weight versus useful load (Figure 3) clearly shows an ever-widening gap in favor of the Hot Cycle system over the shaft-driven concept as useful load is increased.

To demonstrate the adaptability of the Hot Cycle principle, the selected optimum rotor in this study is shown installed on a number of helicopter configurations: the minimum-size streamlined conventional fuselage (configuration 2) carrying all cargo externally, a larger conventional streamlined fuselage with a 12-ton internal capacity (configuration 3), and a crane type (configuration 4) with the capability to carry payloads externally or in pods. In addition, the larger conventional fuselage configuration is also shown as a compound helicopter (configuration 5), so that the features required and the benefits obtained by substantially raising the cruise speed by this means can be identified. The parametric study also included a configuration 1 that was identical with configuration 4 except that a pod was included in the empty weight. Configuration 1 was not considered in the preliminary design, because it was not compatible with other heavy-lift studies for comparison purposes.

The selected optimum rotor has overload payload capabilities considerably in excess of those payloads specified in the heavy-lift requirements, as can be seen in Table I. The characteristics of this Hot Cycle rotor provide good hovering and cruise flight efficiency, low noise level, low downwash velocities, and good flying qualities.



⊡ Heavy-lift helicopter - Hot Cycle system detail design studies

⊙ Current Army turboshaft helicopters. Data from "Comparative Army Aviation Characteristics", Office of the Director of Army Aviation, 3 March 1965

Figure 3. Useful Load Comparison.

## TABLE I
## PERFORMANCE AND WEIGHT SUMMARY

| Item | Heavy-Lift Performance Requirements | Hot Cycle Heavy-Lift Capability Configuration | | |
|---|---|---|---|---|
| | | 2 | 3 | 4 |
| **Performance** | | | | |
| Transport mission (100-nmi radius) | | | | |
| Hover capability with 12-ton payload (95°F OGE) | 6,000 ft | 8,200 | 7,800 | 7,30 |
| Payload capacity (6,000 ft 95°F) | 12 ton | 14.12 | 13.79 | 13.0 |
| Outbound cruise speed (12-ton payload) | 110 kn | 110 | 137* | 114 |
| Inbound cruise speed (no payload-optimum) | 130 kn | 134 | 132 | 130 |
| Heavy-lift mission (20-nmi radius) | | | | |
| Hover capability with 20-ton payload (std OGE) | SL | 6,000 | 5,000 | 4,90 |
| Payload capability (SL 59° OGE) | 20 ton | 26.25 | 25.31 | 25.1 |
| Outbound cruise speed (20-ton payload) | 95 kn | 104 | 103 | 98 |
| Inbound cruise speed (no payload) | 130 kn | 134 | 132 | 130 |
| Ferry mission (at 2-g load factor) | 1,500 nmi | 2,172 | 2,040 | 1,90 |
| Max ferry range (STOL takeoff with load factor reduced to approx 1.75g) | - | 2,308 | 2,203 | 2,03 |
| Max speed capability (normal power at lightweight condition) | - | 179 | 178 | 175 |
| **Weights (lb)** | | | | |
| Empty weight | | 19,599 | 20,570 | 21,1 |
| Gross weight | | | | |
| Transport mission (12-ton payload) | | 52,260 | 52,234 | 54,1 |
| Heavy-lift mission (20-ton payload) | | 64,280 | 65,481 | 66,1 |
| Payload/empty weight ratio | | | | |
| Transport mission (max payload) | | 1.4 | 1.3 | 1.2 |
| Heavy-lift mission (max payload) | | 2.7 | 2.5 | 2.4 |

*Internal load for configuration 3.

The study of the compound helicopter was undertaken on a limited basis to identify the compromises in weight, size, complexity, and performance required to attain a substantial increase in cruise speed. Compounding was accomplished by the addition of wings and ducted fans for thrust. The study showed that the compound helicopter will provide a substantial increase in cruise speed and ferry range. The additional complexity of the compound is confined primarily to the wing and ducted thrust-fan installations, and the required implementation is well within the state of the art.

HTC-AD experience in the design and engineering of the Hot Cycle helicopter spans more than 10 years. The feasibility and attractiveness of the Hot Cycle propulsion system have been established through an extensive research and development program that culminated in the successful flight testing of the U.S. Army AVLABS XV-9A Hot Cycle Research Aircraft shown in Figure 4. During 160 hours of rotor operation and 35 hours of flight testing that was completed in August 1965, structural and mechanical design, weights, and cooling adequacy were verified. Gas leakage was found to be negligible (less than 1/5 of 1 percent) and noise was determined to be essentially equal to that of the quietest type of VTOL aircraft (turboshaft helicopter). The large reduction in maintenance requirements promised by the Hot Cycle system was illustrated by the low logistical requirements during XV-9A flight operations.

Figure 4. XV-9A Hot Cycle Research Aircraft.

## STUDY REQUIREMENTS

The preliminary design parametric and configuration study is based o the following vehicle and mission requirements.

### VEHICLE

The vehicle shall have the following characteristics:

1. Turbine power.
2. Safe autorotation at design gross weight.
3. Design vertical limit load factor of 2. 5 to -0. 5 g at design g weight.* For the integrated preliminary design, the design weight is interpreted to be the heavy-lift mission gross weig carrying a 20-ton payload.
4. Crew minimum of one pilot, one copilot, and one crew chief.
5. All components to be designed for 1, 200 hours between majo overhauls and 3, 600-hour service life.
6. Multiengine installation.

### MISSIONS - HELICOPTER

The aircraft shall be able to perform the following missions:

1. Transport mission
   a. Payload: 12 tons (outbound only)
   b. Radius: 100 nautical miles
   c. Cruise speed: 12-ton payload, 110 knots
   d. Cruise speed: no payload, 130 knots
   e. Hovering time: 3 minutes at takeoff; 2 minutes at midpo
   f. Reserve fuel: 10 percent of initial fuel
   g. Hover capability: 6, 000 feet 95°F (OGE)
   h. Cruise altitude: sea level standard atmosphere
   i. Fuel allowance for start, warmup, and takeoff per MIL-C-5011A

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*For the parametric study, the design gross weight was taken as the transport mission gross weight, with a resulting design limit load f of +2. 75 for compatibility with the ferry mission load factor of 2. 0.

2. Heavy-lift mission
    a. Payload: 20 tons (outbound only)
    b. Radius: 20 nautical miles
    c. Cruise speed: 20-ton payload, 95 knots
    d. Cruise speed: no payload, 130 knots
    e. Hovering time: 5 minutes at takeoff; 10 minutes at destination (with payload)
    f. Reserve fuel: 10 percent of initial fuel
    g. Hover capability: sea level 59°F (OGE)
    h. Cruise altitude: sea level standard atmosphere
    i. Fuel allowance for start, warmup, and takeoff per MIL-C-5011A
3. Ferry mission
    a. Ferry range: 1,500 nautical miles (no payload, STOL takeoff)
    b. Reserve fuel: 10 percent of initial fuel
    c. Fuel allowance for start, warmup, and takeoff per MIL-C-5011A
    d. Minimum design load factor of 2.0
    e. Best altitude for range
    f. Best speed for range

## MISSIONS - COMPOUND HELICOPTER

The following missions were selected for the compound study:

1. Transport mission
    a. Payload: both ways, weight to be determined
    b. Radius: 200, 300, and 500 nautical miles
    c. Cruise: 225 knots (minimum)
    d. Hovering time: 4 minutes at takeoff 2 minutes at destination (with payload)
    e. Reserve fuel: 10 percent of initial fuel
    f. Hover capability

| | |
|---|---|
| Basic | Hover OGE - initial takeoff at sea level, 59°F; cruise at sea level and best altitude |
| Altitude | Hover OGE - initial takeoff at 6,000 feet, 95°F; cruise at sea level and best altitude |
| Overload | Initial running takeoff at sea level, 59°F; hover OGE at destination at sea level, 59°F; cruise at best altitude and, alternatively, at sea level |

2. Ferry mission

a. Payload: none
b. $V_{cruise}$: for best range
c. Cruise altitude: for best range
d. Range: to be determined
e. Fuel reserve: 10 percent of initial fuel
f. Initial takeoff: STOL, sea level, 59° F

## AIRCRAFT CONFIGURATIONS STUDIED

A wide range of aircraft configurations has been considered in order to show the adaptability of the Hot Cycle rotor to any configuration that might be dictated by operational requirements. By installing the same rotor and propulsion system on the different airframes, the effect of configuration on mission effectiveness can be seen. A brief description of each of the helicopter configurations considered is given in the following paragraphs.

### MINIMUM-SIZE CONVENTIONAL FUSELAGE (Configuration 2) (Figure 5)

This configuration utilizes a conventional streamlined fuselage sized to carry the ferry fuel internally. A top-mounted engine installation has been utilized to reduce frontal area. This configuration has been included in the study because it represents the configuration having the lowest empty weight, highest payload-to-empty-weight ratio, and the longest ferry range capability. It is well to note that this configuration has the ability to meet the mission requirements with a rotor smaller than the selected optimum rotor and at a substantially lighter empty weight. The cargo compartment is approximately 6-1/2 feet wide, 7 feet high, and 45 feet long, and will accommodate six standard 54-by-88-inch pallets. Approximately 7 tons may be carried internally at a 10-pound-per-cubic-foot loading. Structural provisions have been included for the 7-ton internal load, though mission performance has been determined based on carrying the transport and heavy-lift mission payloads externally.

### CONVENTIONAL FUSELAGE WITH 12-TON INTERNAL CAPACITY (Configuration 3) (Figure 6)

A conventional streamlined fuselage has been used on this configuration, sized to carry 12 tons internally (at 10 pounds per cubic foot). The cargo compartment is approximately 8 feet wide, 7 feet high, and 46 feet long, and will accommodate six standard 88-by-108-inch pallets. The engines have been shoulder-mounted for accessibility and for ease of converting this configuration into a compound helicopter. Performance of this configuration has been determined assuming the transport mission payload to be carried internally and the heavy-lift mission payload externally.

### CRANE-TYPE (Configuration 4) (Figure 7)

This configuration is a crane type utilizing a straddle gear and sized to accommodate a pod with a cargo compartment 10 feet wide, 9 feet high, and 27 feet long. The cross section dimensions were chosen to be the same as those of the C-130 airplane cargo compartment to permit direct reloading between vehicles. The transport and heavy-lift payloads were assumed to be carried externally for the determination of performance. The fuel has been assumed to be carried in a faired pod for the ferry mission.

## COMPOUND HELICOPTER (Configuration 5) (Figure 8)

The study of the compound helicopter was undertaken on a limited basis to identify the compromises in weight, size, complexity, and performance required to attain a substantial increase in cruise speed. Configuration 5 is identical with the configuration 3 helicopter (conventional fuselage, 12-ton internal capacity) except that wings and ducted fans for thrust have been added for operation as a compound helicopter. To fly as a compound, the high-energy gas is diverted from the rotor to the ducted fans, with the wing acting to unload the rotor. The increased speed of the compound resulted in an appreciably higher productivity than that achieved by the configuration 3 helicopter.

## ENGINE INSTALLATION

Two engine installations were considered in the parametric study, one utilizing two GE1/J1 engines and the other utilizing four GE T64/S4B engines.

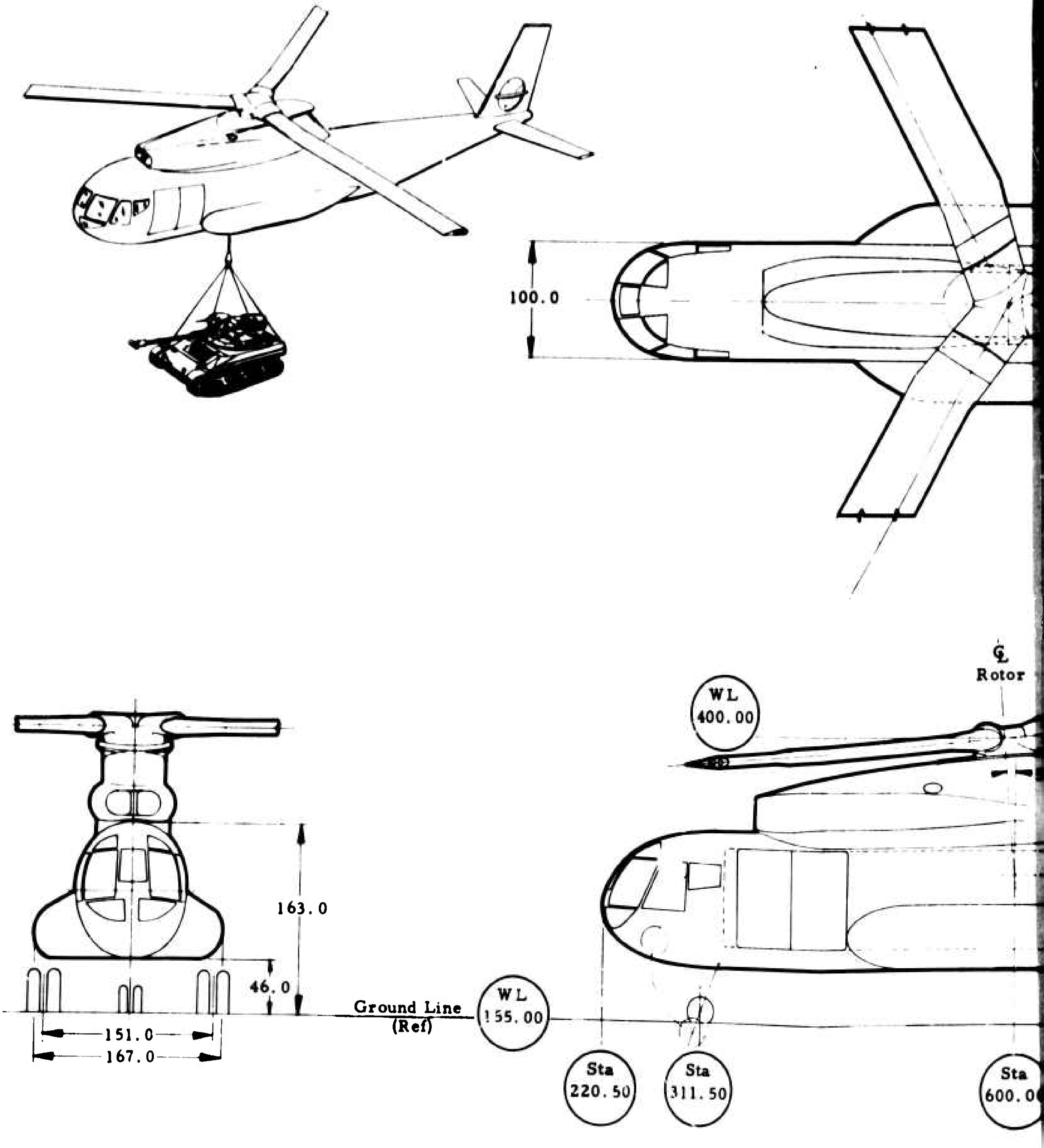
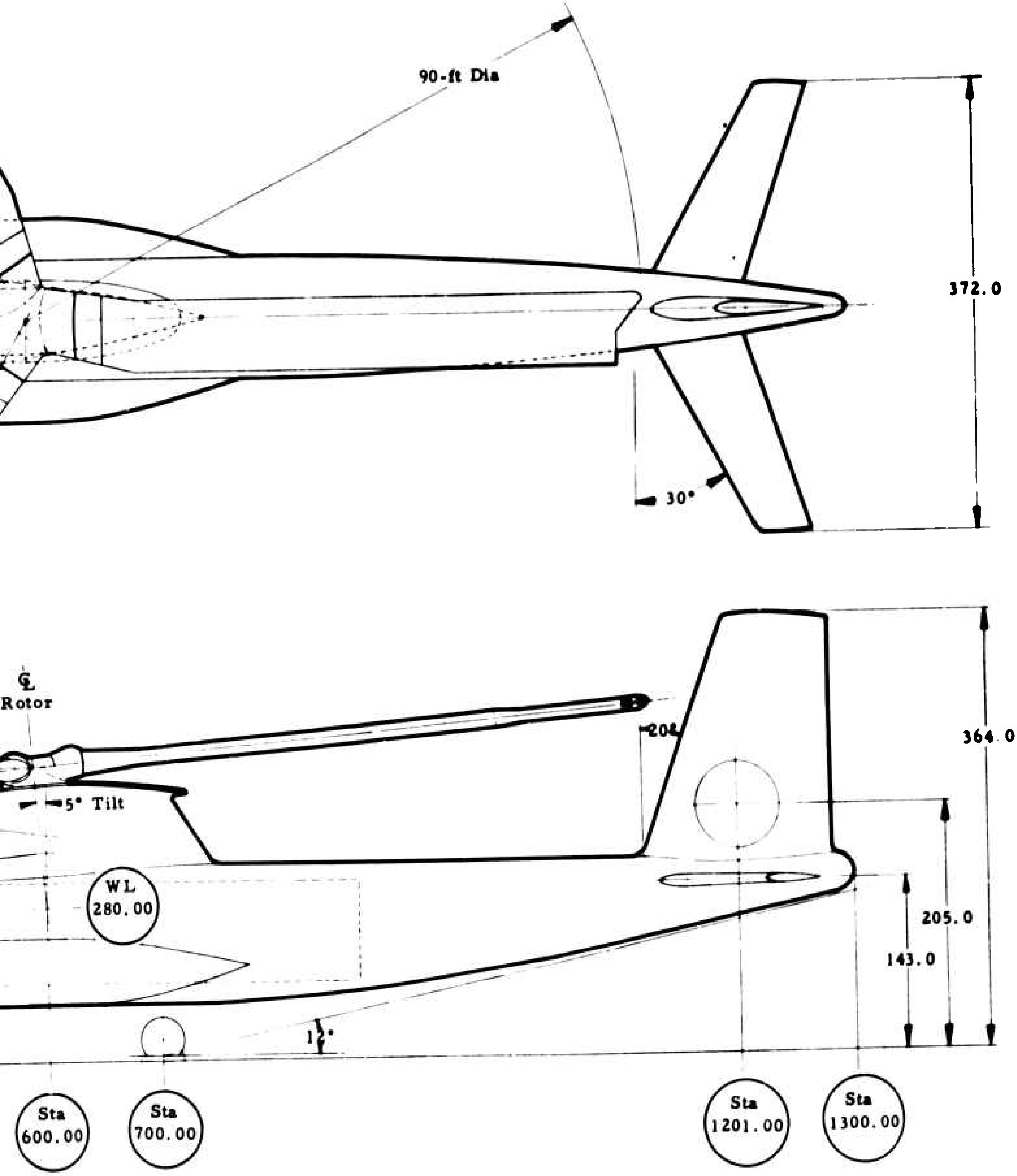


Figure 5. General Arrangement - Con



nt - Configuration 2 Helicopter.

B

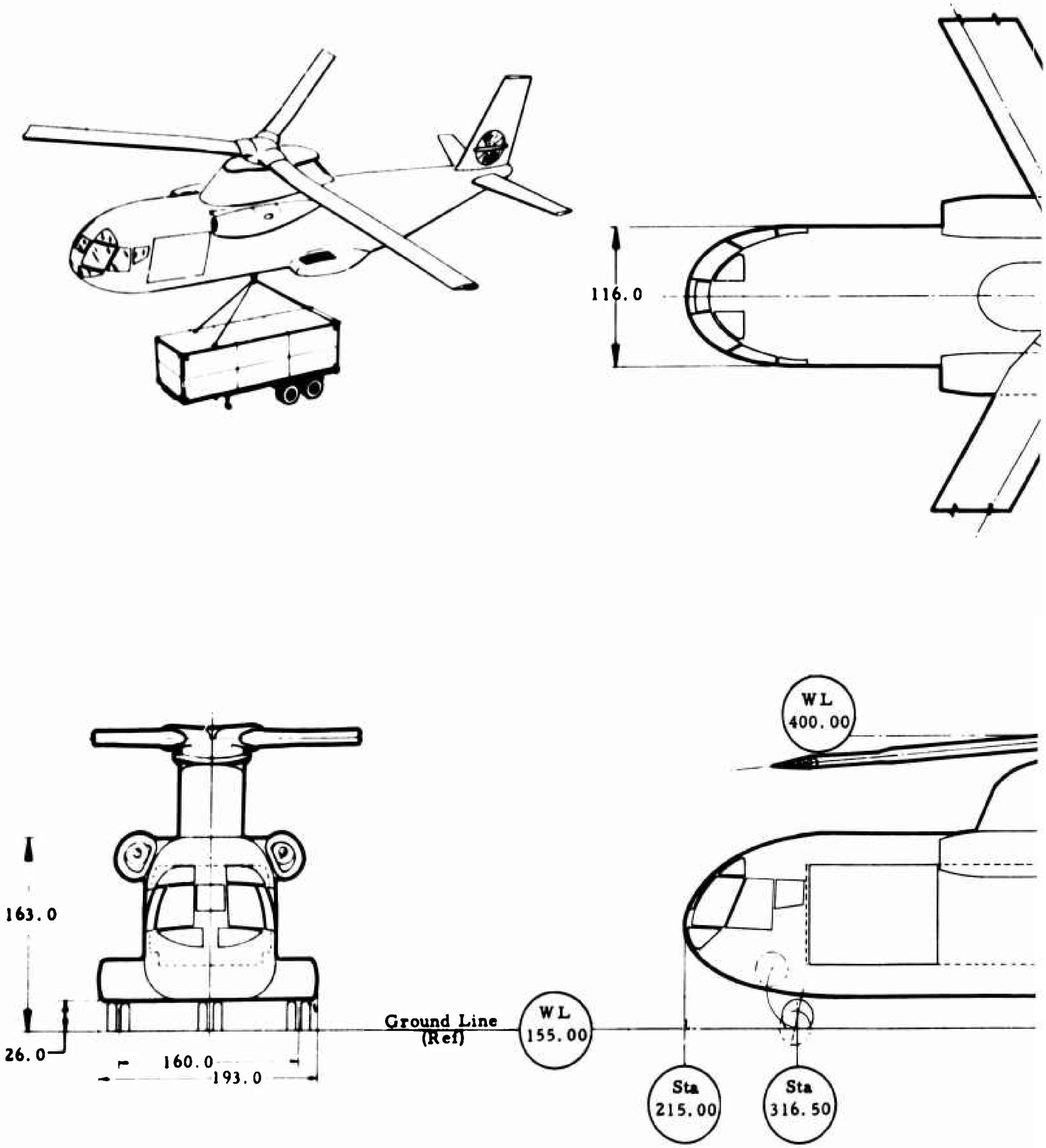
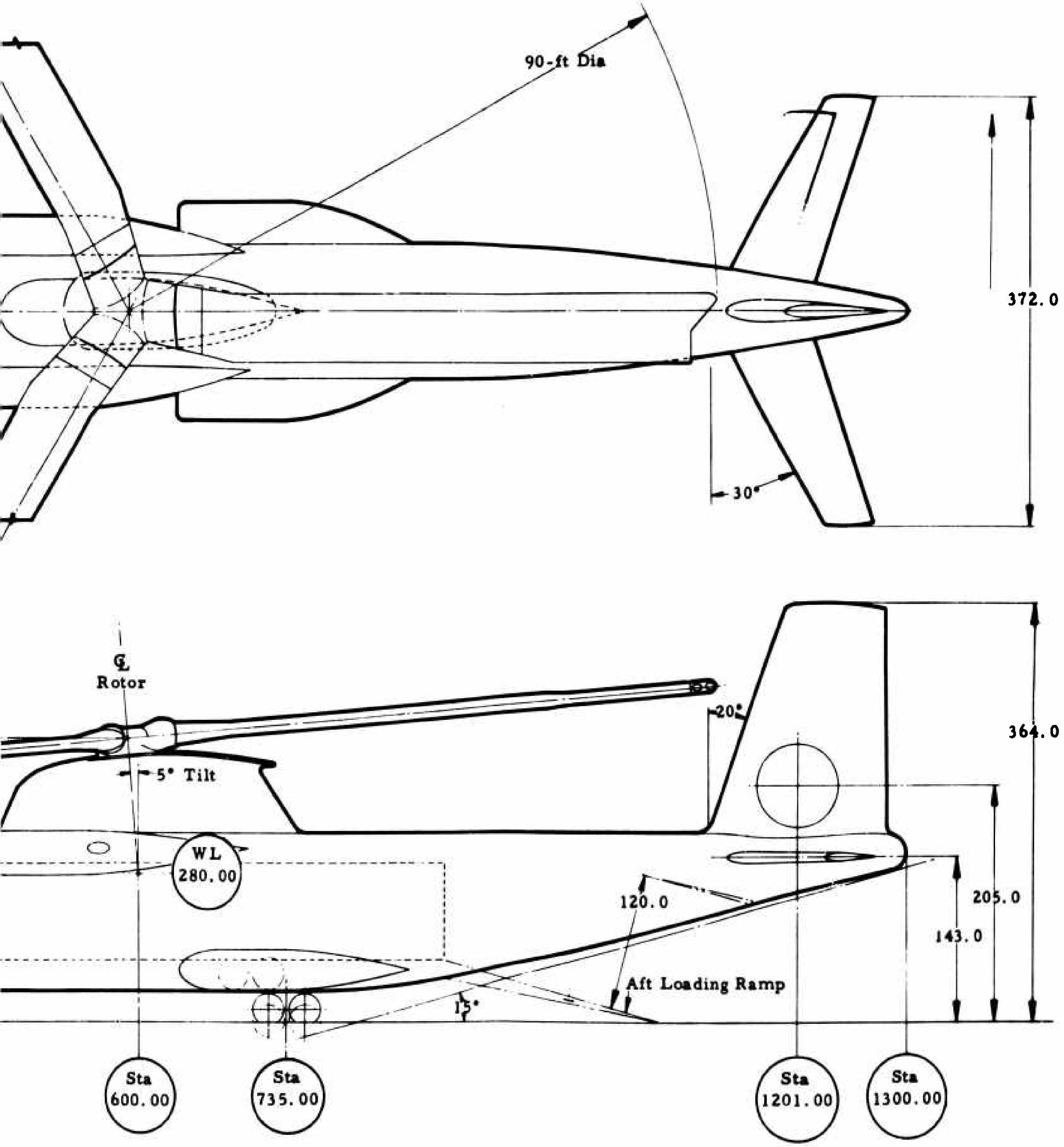


Figure 6. General Arrangeme



gement - Configuration 3 Helicopter.

B

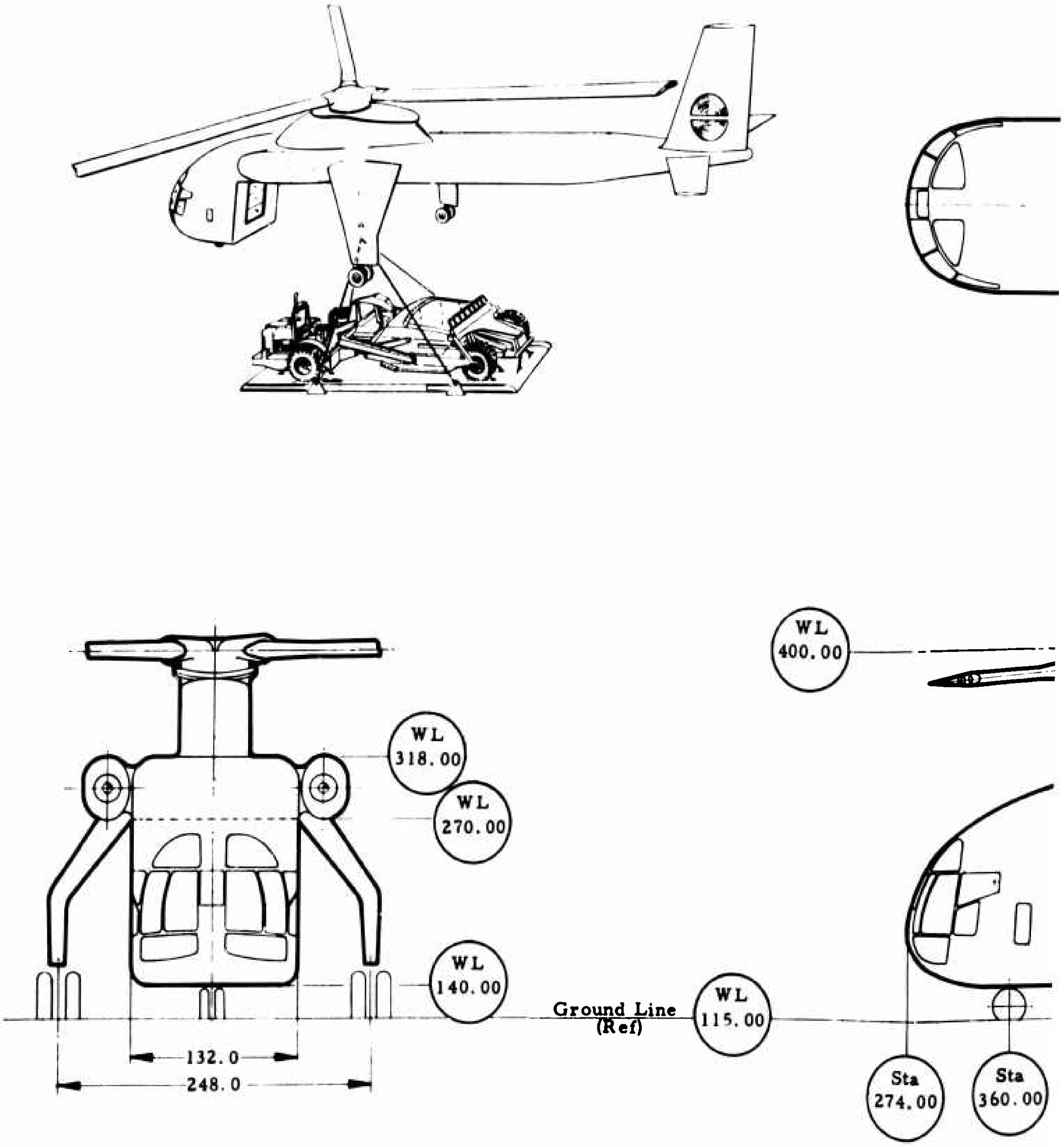
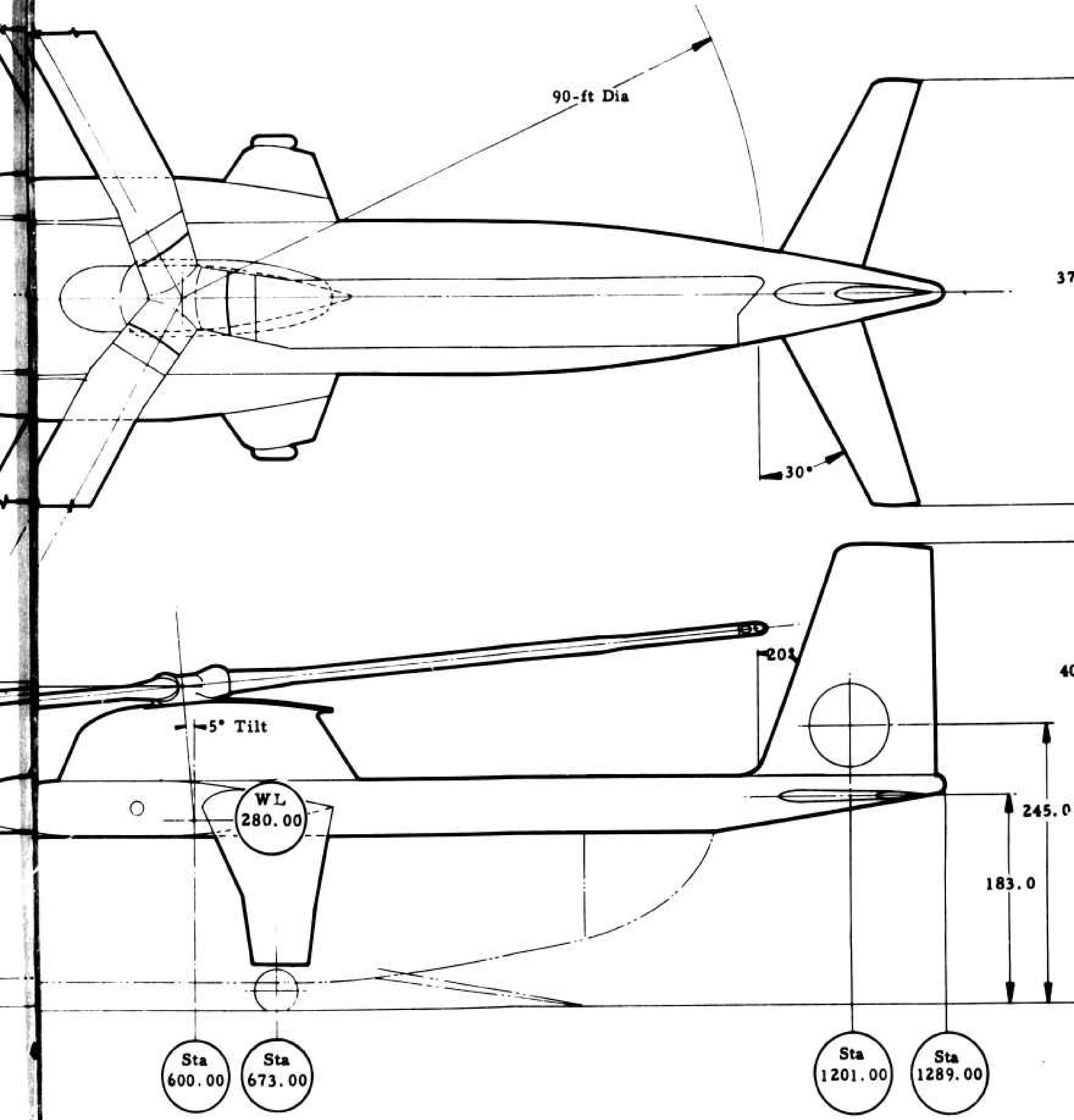


Figure 7. General /

A



gement - Configuration 4 Helicopter.

B

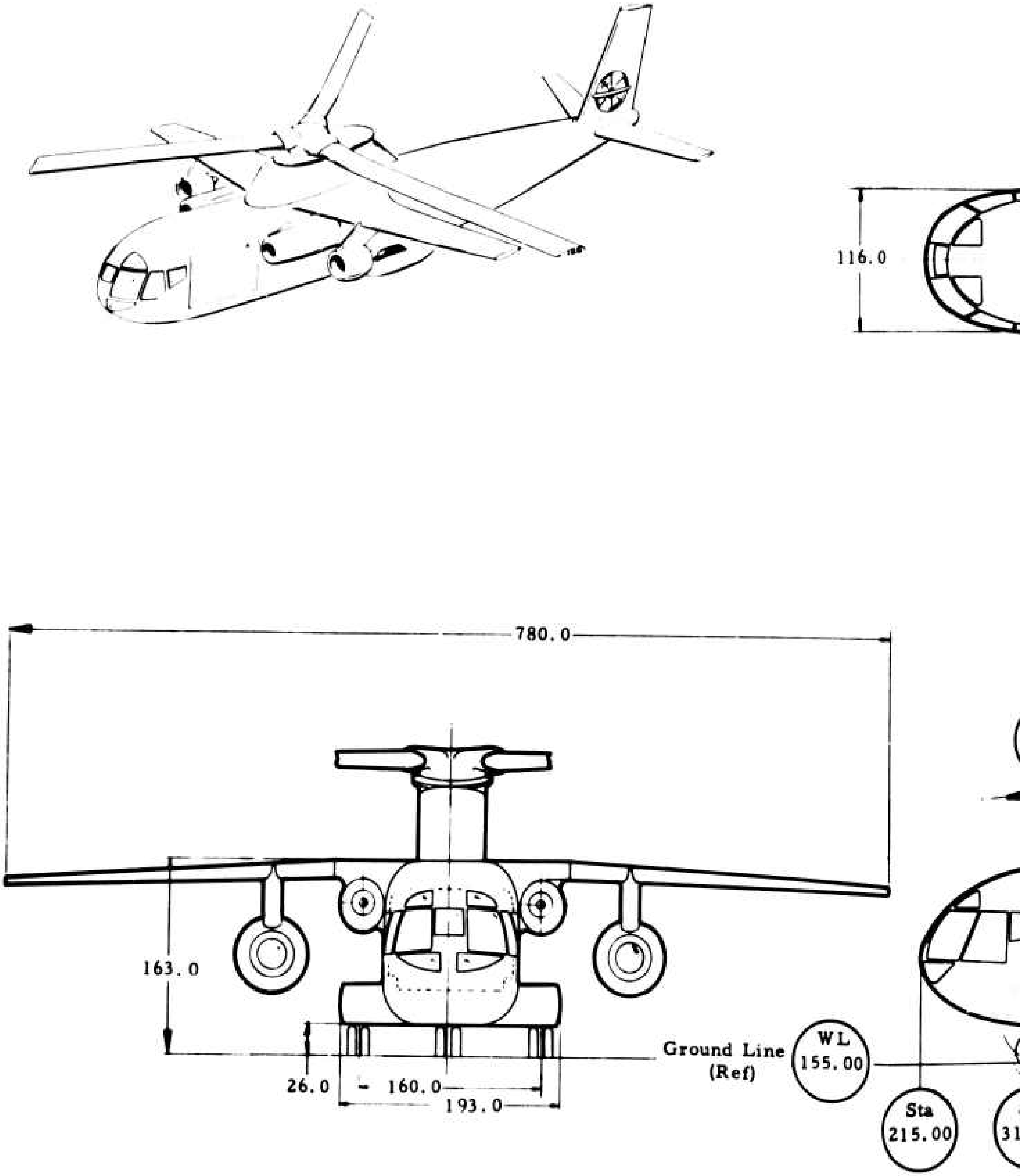
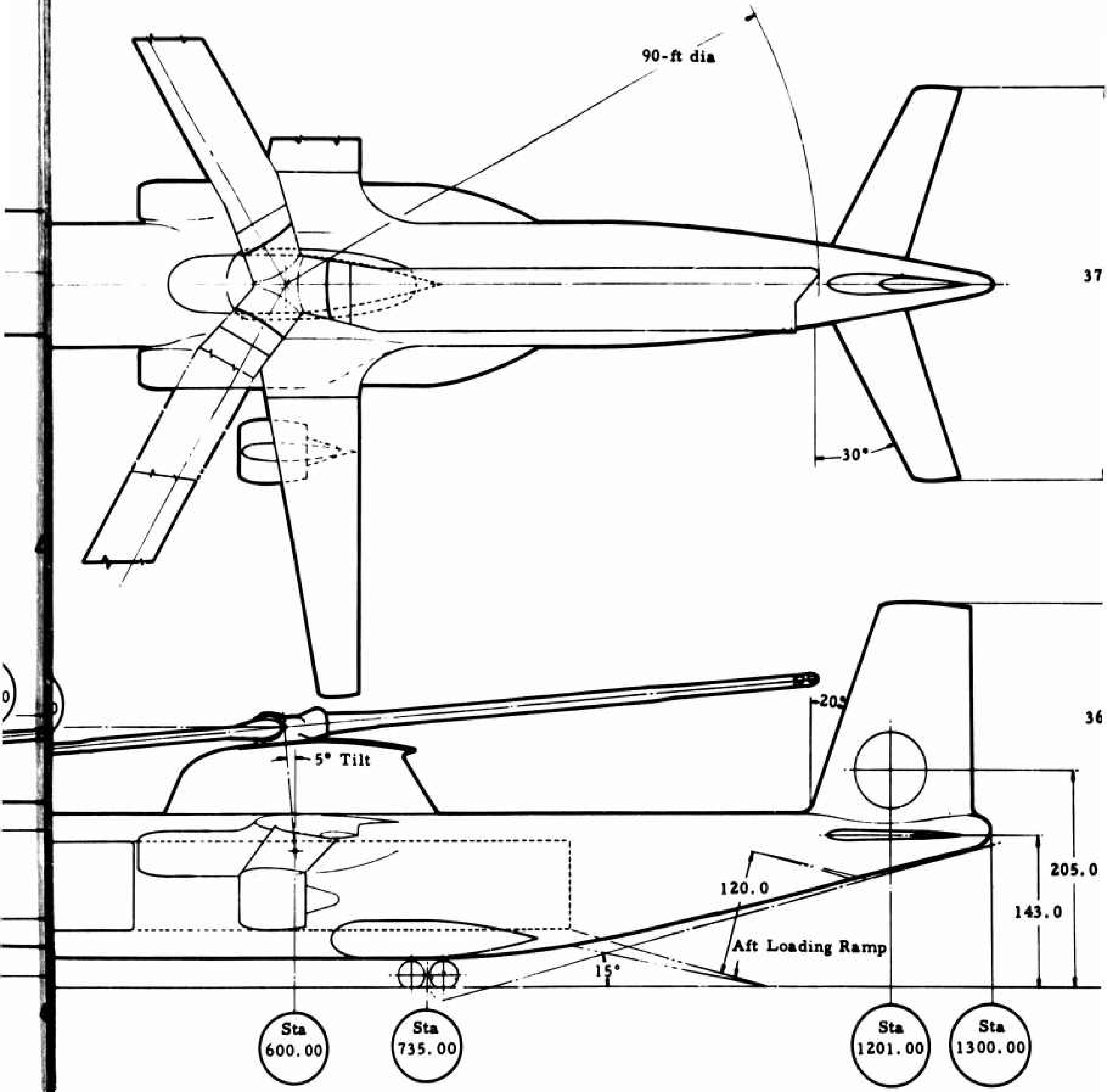


Figure 8. General A



gem ement - Configuration 5 Compound.

B

## PERFORMANCE

The performance of the Hot Cycle heavy-lift helicopter meets all mission requirements and exceeds most of the specified requirements by a substantial margin, as can be seen in Table I. Substantial improvement in fuel utilization efficiency over the best current turbine-powered helicopters is attained by the Hot Cycle propulsion system.

The selection of the optimum rotor was based on the results of the parametric study, wherein the effect of the many rotor variables was evaluated on several helicopter configurations. The rotor considered most nearly optimum for all the configurations studied was selected for the preliminary design. As can be seen by comparing the parametric study and preliminary design results (Tables I and XI), the performance of each aircraft configuration with its optimum rotor is somewhat superior to the performance of the same configuration utilizing the rotor selected for the preliminary design. Also contributing to the differences in performance are refinements to the weight and power-available equations as used in the parametric study. Subsequent to completion of the parametric study, the 20-ton heavy-lift mission was designated as the primary mission. This has resulted in a small increase in empty weight, as the design gross weight for the parametric study was initially assumed to be the transport mission gross weight.

## PERFORMANCE COMPUTATIONS

All power-required computations are based on standard computation methods developed by NASA, with additional corrections for blade stall and drag divergence. A complete discussion of the computation method is presented in Reference 3.

The induced power in hovering is computed using simple momentum theory, with corrections for tip loss, planform, and twist. The download on the fuselage is also estimated from the induced velocity. The profile power is based on the NACA polar for a 12-percent thickness airfoil, with corrections for blade thickness and practical construction.

The helicopter forward flight power required is computed using the NACA charts given in Reference 4. The profile power of these charts is corrected for thickness and practical construction. Profile power increase as a result of retreating tip stall and advancing tip drag divergence is also included, with the aid of NACA whirl tower model data, Reference 5.

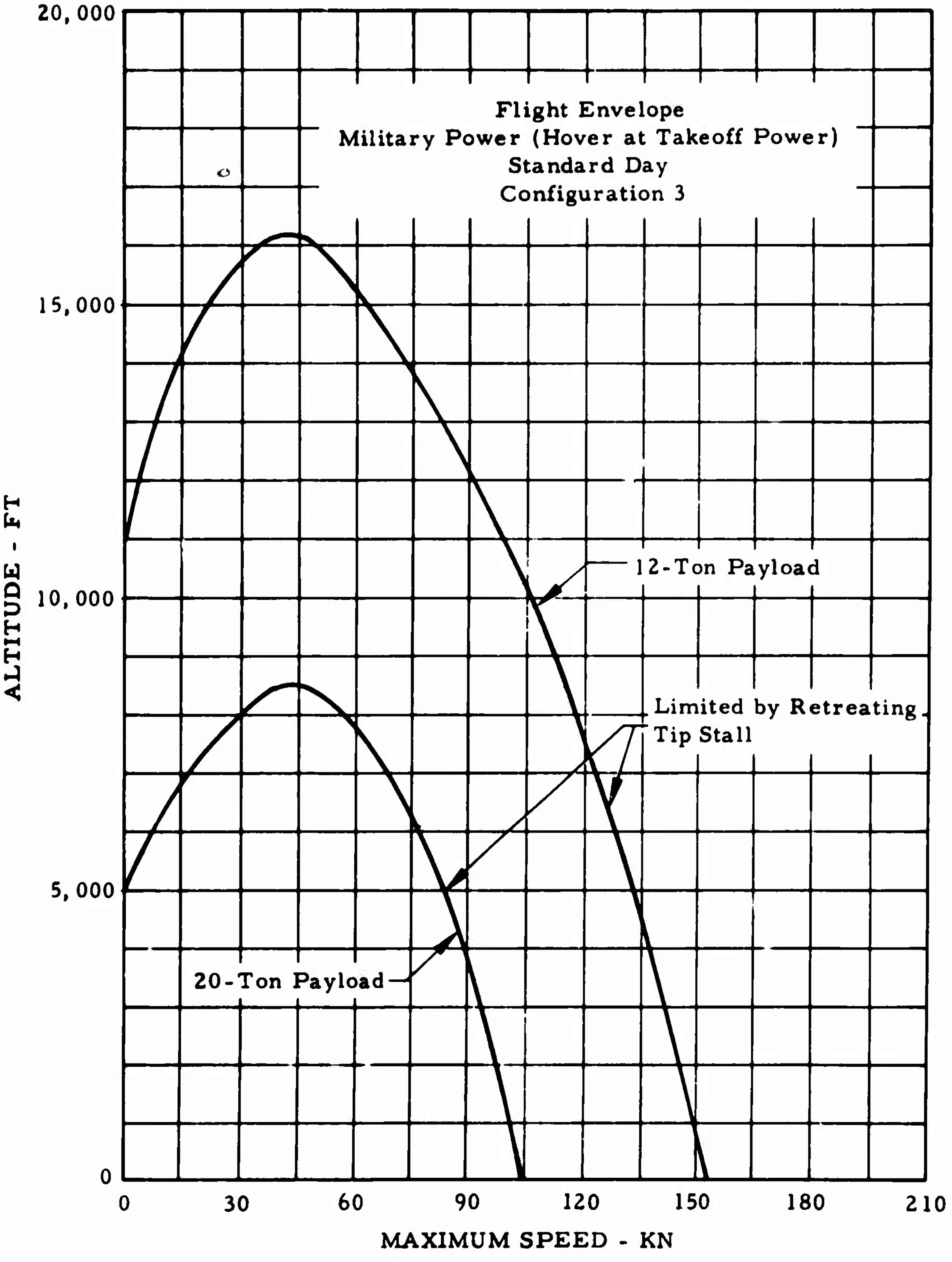


Figure 9. Flight Envelope.

The flight envelope, Figure 9, presents the maximum and minimum airspeeds as limited by military power or retreating tip stall. The retreating tip stall speed is determined as the speed at which the retreating tip drag coefficient is equal to 0. 0

Figure 10 presents the hover ceiling for standard ambient conditions in and out of ground effect as a function of gross weight. Takeoff power was used for the hover ceiling computation. A rotor height equal to one-half rotor diameter was assumed for the in-ground-effect calculations.

Figure 11 presents the maximum rate of climb with military power as a function of altitude.

Figure 12 shows the payload-range curve for sea level standard and 6, 000-foot 95° F hover conditions. Payload is outbound only: no return and no reserve fuel.

## PARASITE DRAG AREA ESTIMATION

Estimates were made of the parasite drag areas of the basic helicopter configurations with alternate hub arrangements. These estimates were based on References 6 and 7 and on sea level 59° F conditions, with velocity in the 95- to 130-knot range and gross weight in the 55, 000- to 90, 000-pound range. Results are presented in Table II.

The assumptions are as follows:

1. Fuselage angle of attack remains sufficiently low for all conditions to take it as zero for drag estimates.
2. Empennage parasite drag area (includes trim) constant at 3. 98 square feet.
3. Items such as rotor hub, pylon fairing, and landing gear have the same drag when used on fuselages of different configuration; that is, interference effects are taken as the same.
4. All fuselage corners have a radius at least 20 percent of width (or height). This assures lowest drag.
5. External payloads are constant-size cubes with a cargo density of 30 pounds per cubic foot; therefore, 12-ton payload = 9. 3 x 9. 3 x 9. 3. A 50-foot support cable is used.
6. Parasite drag areas for the compound helicopter include additional drag values of 0. 01 times wing area, 0. 6 square foot for fuselage-wing interference, 2 percent of fan thrust for nacelle drag, and 2 percent of fan thrust for nacelle-wing interference.

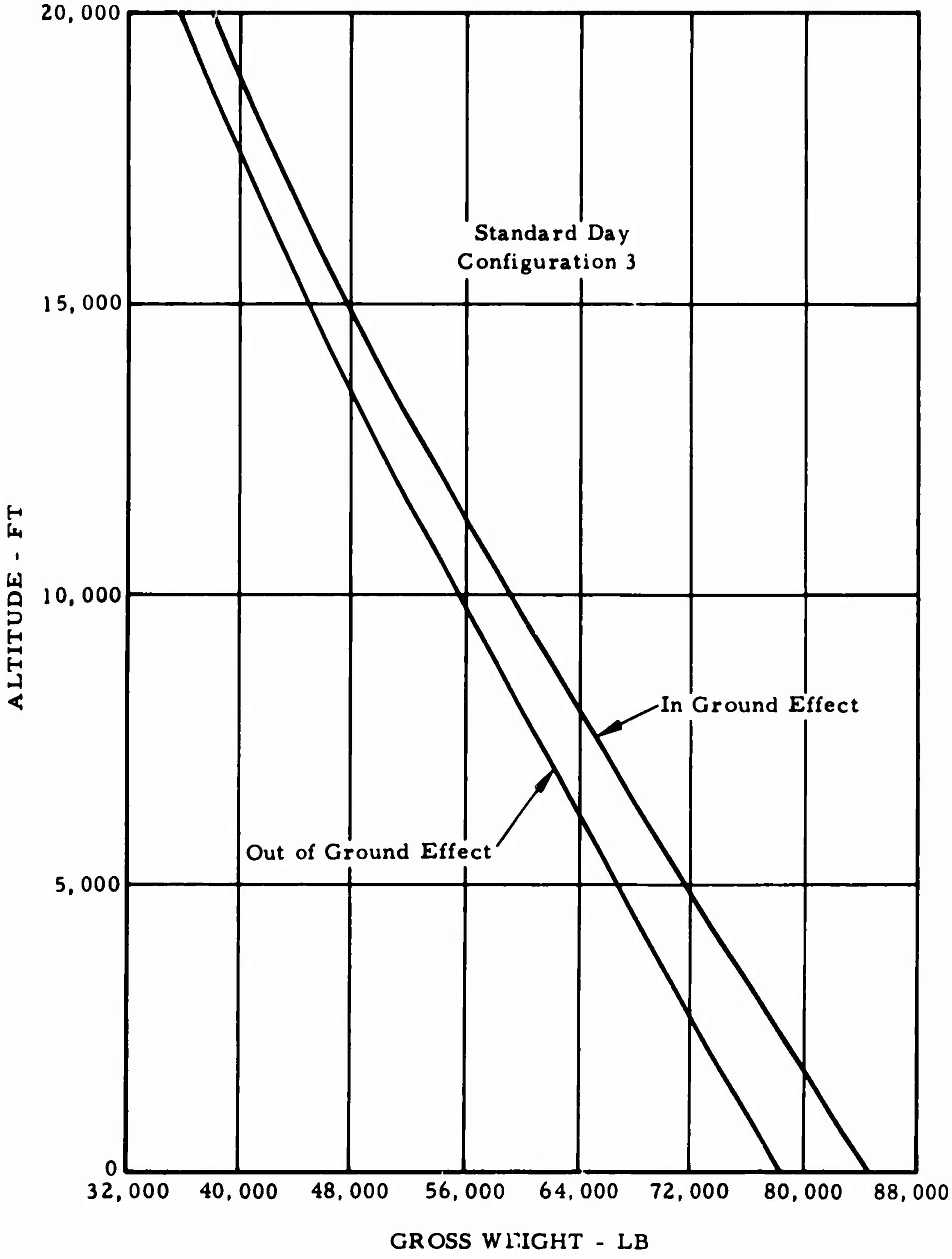


Figure 10. Hover Ceiling.

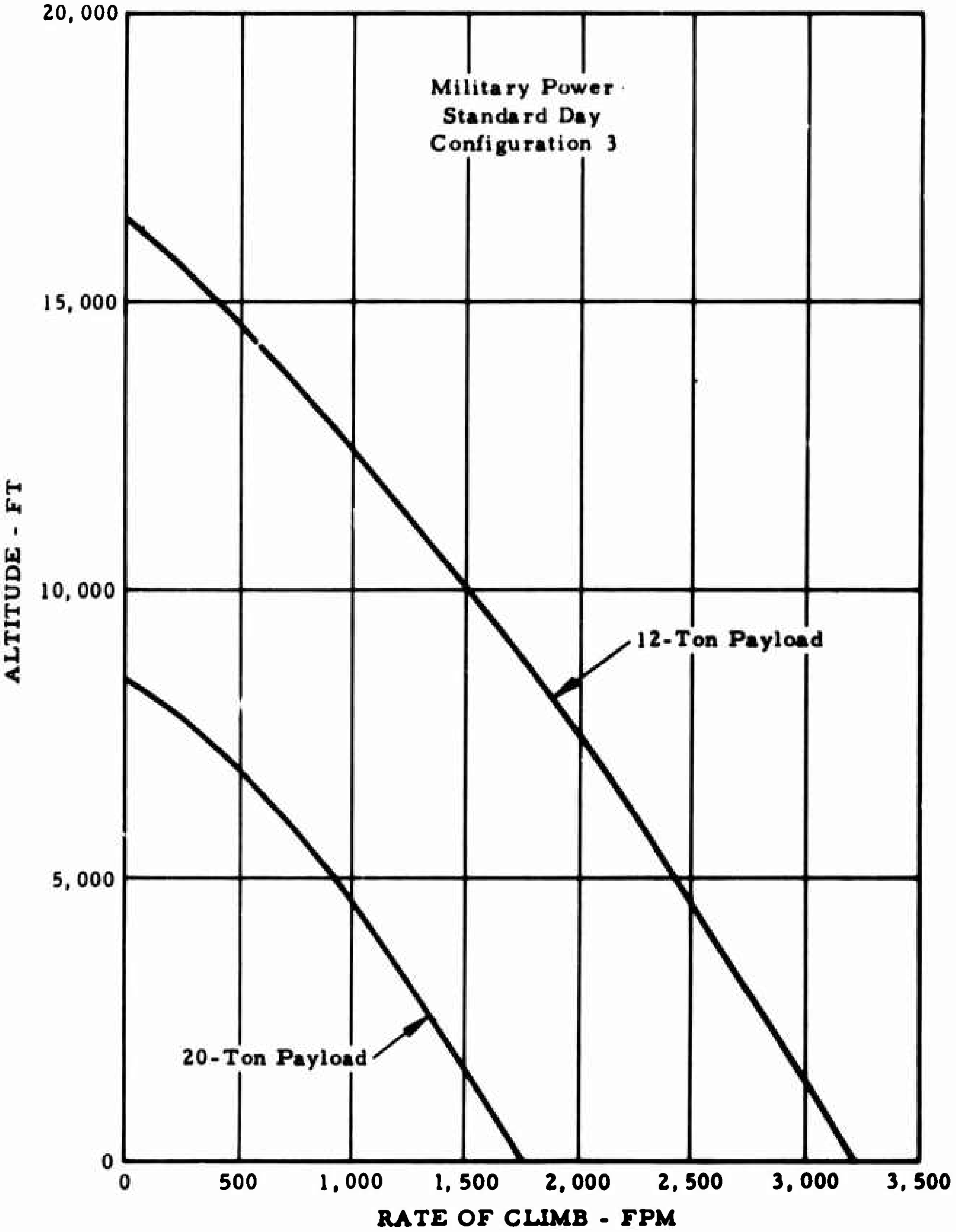


Figure 11. Maximum Rate of Climb Versus Altitude.

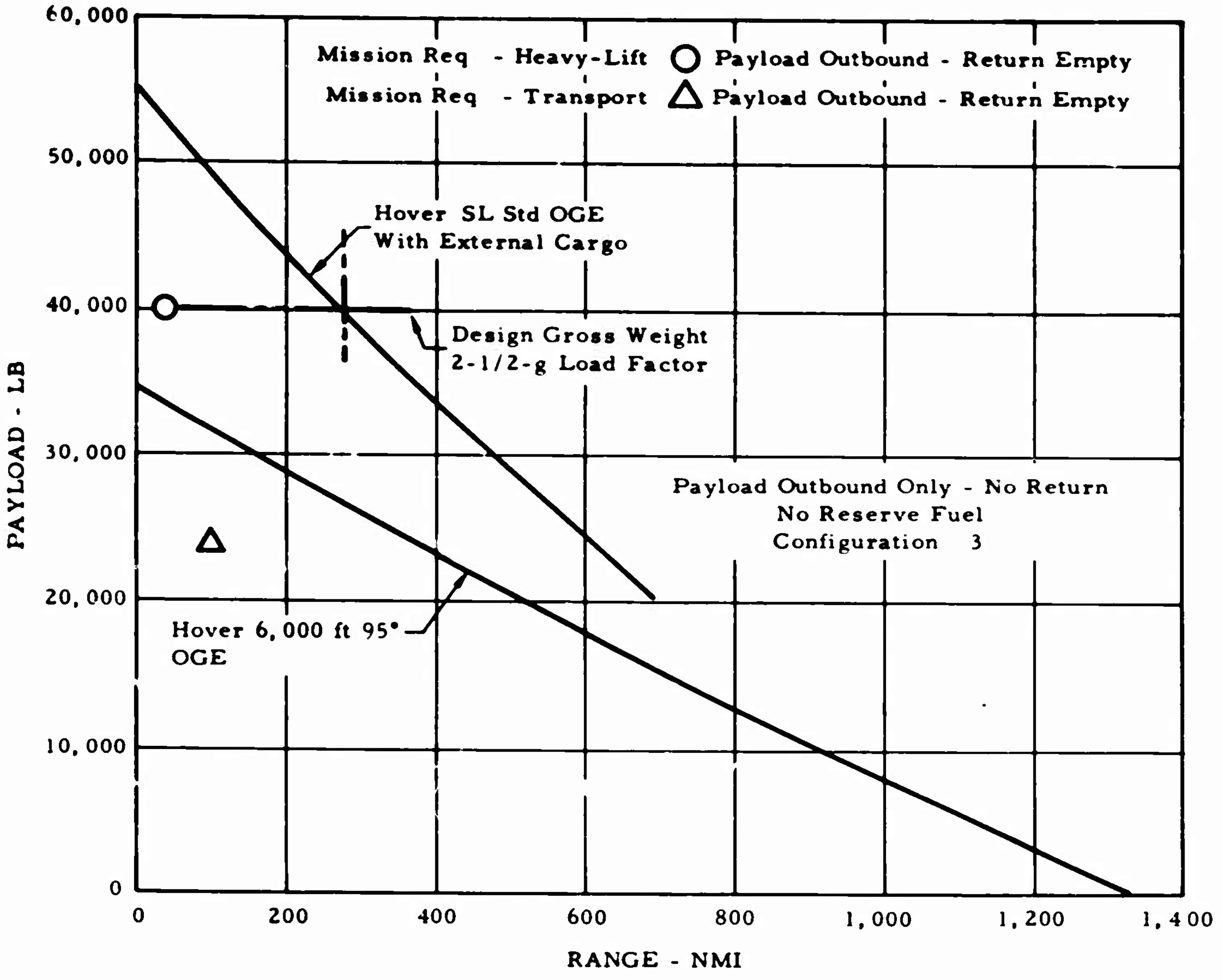


Figure 12. Payload Range - Sea Level Mission.

# TABLE II
# PRELIMINARY PARASITE DRAG ESTIMATE OF HEAVY-LIFT HELICOPTER CONFIGURATION

| | | Parasite Drag in Square Feet | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | Hub | | | | | | | | | | |
| | | Articulated | | Tilting | | Landing Gear | | External Payload | | Total | | |
| Configuration | Fuselage | Exterior Controls | Interior Controls | Exterior Controls | Interior Controls | Fixed | Retractable | 12-Ton(1) | 20-Ton(1) | Ferry Mission | Transport Mission | Heavy-Lift Mission |
| 1(2) | 26.39 | 8.06 | - | - | - | 3.33 | - | - | 139 | 37.78 | 37.78 | 176.78 |
| 1 | 26.39 | 8.06 | - | - | - | - | 0 | - | 139 | 34.45 | 34.45 | 173.45 |
| 1 | 26.39 | - | 10.86 | - | - | - | 0 | - | 139 | 37.25 | 37.25 | 176.25 |
| 1 | 26.39 | - | - | 18.76 | - | - | 0 | - | 139 | 45.15 | 45.15 | 184.15 |
| 1 | 26.39 | - | - | - | 23.01 | - | 0 | - | 139 | 49.40 | 49.40 | 188.4 |
| 4(3) | 48.75 | 8.06 | - | - | - | 3.33 | - | 33 | 139 | 60.14 | 93.14 | 199.14 |
| 4 | 48.75 | 8.06 | - | - | - | - | 0 | 33 | 139 | 56.81 | 89.81 | 195.81 |
| 2 | 11.03 | 8.06 | - | - | - | 3.99 | - | 99 | 139 | 23.08 | 122.08 | 162.08 |
| 2 | 11.03 | 8.06 | - | - | - | - | 1.95 | 99 | 139 | 21.04 | 120.04 | 160.04 |
| 2 | 11.03 | - | 10.86 | - | - | - | 1.95 | 99 | 139 | 23.84 | 122.84 | 162.64 |
| 2 | 11.16 | - | - | 18.76 | - | - | 1.95 | 99 | 139 | 31.87 | 130.87 | 170.87 |
| 2 | 11.16 | - | - | - | 23.01 | - | 1.95 | 99 | 139 | 36.12 | 135.12 | 175.12 |
| 3 | 17.80 | 8.06 | - | - | - | 2.50 | - | - | 139 | 28.36 | 28.36 | 167.36 |
| 3 | 17.80 | 8.06 | - | - | - | - | 0 | - | 139 | 25.86 | 25.86 | 164.86 |
| 3 | 17.80 | - | 10.86 | - | - | - | 0 | - | 139 | 28.66 | 28.66 | 167.66 |
| 3 | 17.80 | - | - | 18.76 | - | - | 0 | - | 139 | 36.56 | 36.56 | 175.56 |
| 3 | 17.80 | - | - | - | 23.01 | - | 0 | - | 139 | 40.81 | 40.81 | 179.81 |
| 5(4) | | | | | | | | | | | | |

(1) Drags based on 30-lb/cu-ft density. A decrease to 20 lb/cu-ft would increase $\left(\frac{D}{q_0}\right)$ 23 sq ft for 12-ton load and 36 sq ft for 20-ton load.

(2) Includes pod.

(3) Without pod.

(4) Configuration 5 compound drags may be obtained by adding 0.01 times wing area, 0.6 square foot for fuselage-wing interference to the configuration 3 drags. An allowance of 2-percent fan thrust must also be made for nacelle drag and another 2-percent fan thrust for nacelle-wing interference.

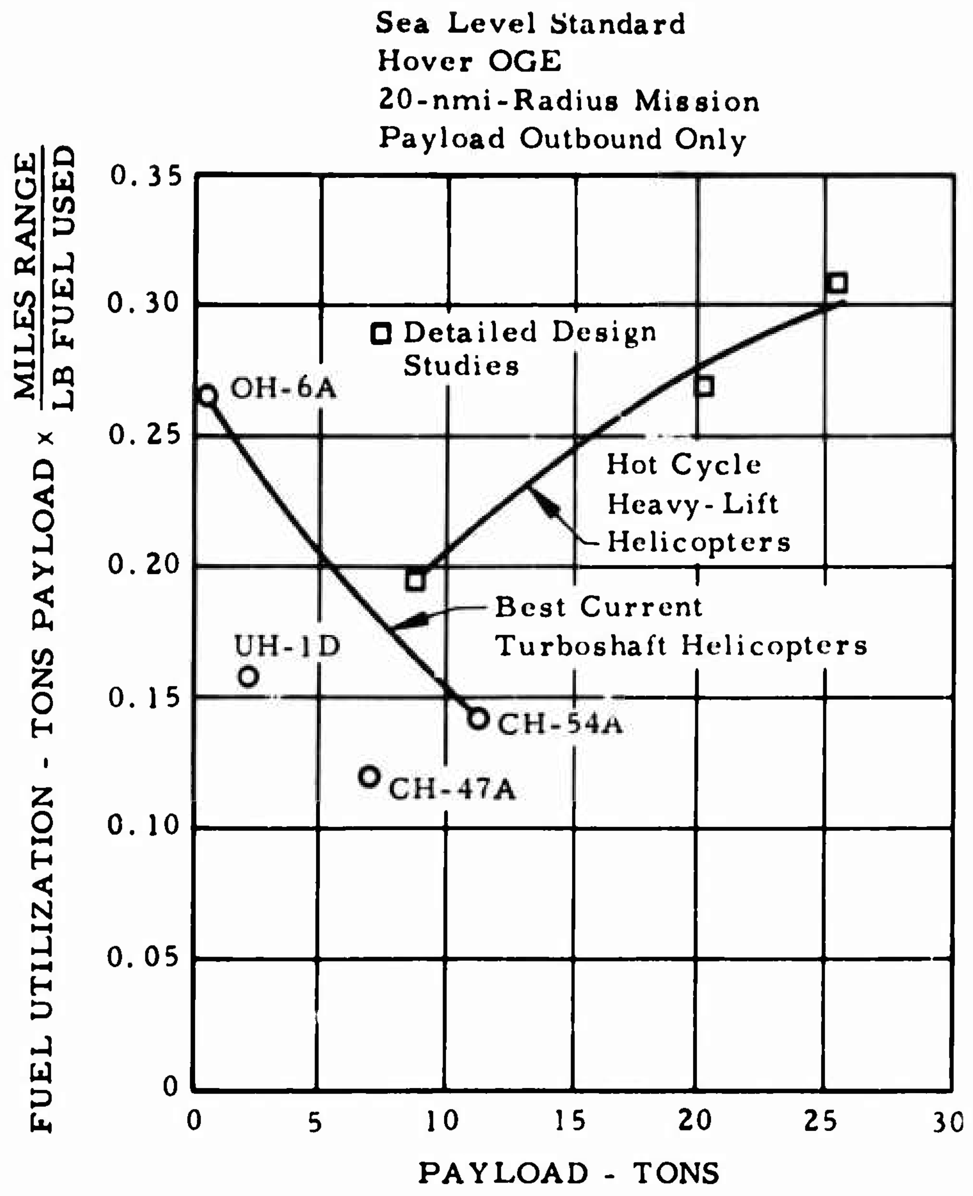


Figure 13. Fuel Utilization Versus Payload - Heavy-Lift Mission.

## DISC LOADING

Though not a required part of the study, disc loadings were taken into consideration. The selected optimum rotor installed on the airframe configurations considered in this study results in disc loadings of approximately 10 pounds per square foot for the heavy-lift mission (20-ton payload). Even taking advantage of the large overload capability of the optimum Hot Cycle rotor for the heavy-lift mission results in disc loadings of approximately 12 pounds per square foot. However, this is an external-load condition and downwash hazards are minimized, since the actual disc loading and resulting downwash velocity are very low during hookup and until lift-off. For the transport mission, disc loadings are much more modest.

## FUEL UTILIZATION

The results of the fuel consumption study indicate that a breakthrough for the economy of helicopter transports can be expected using the Hot Cycle propulsion system. The fuel utilization (payload ton-mile/pound of fuel) was calculated for the various configurations and missions and is shown in Figures 13 and 14 for the heavy-lift and transport missions, respectively. Fuel utilization based on payload, as opposed to specific fuel consumption, fuel flow/gross weight, and other parameters, is of direct importance for estimating actual fuel costs of specific helicopter operations. These comparisons indicate that for heavy-lift payloads the Hot Cycle offers substantial improvements over the best present turbine-powered helicopters (References 8 through 12). This excellent fuel utilization efficiency of the Hot Cycle helicopter is mainly the result of its excellent payload/empty weight ratio, the empty weight of the helicopter being greatly reduced.

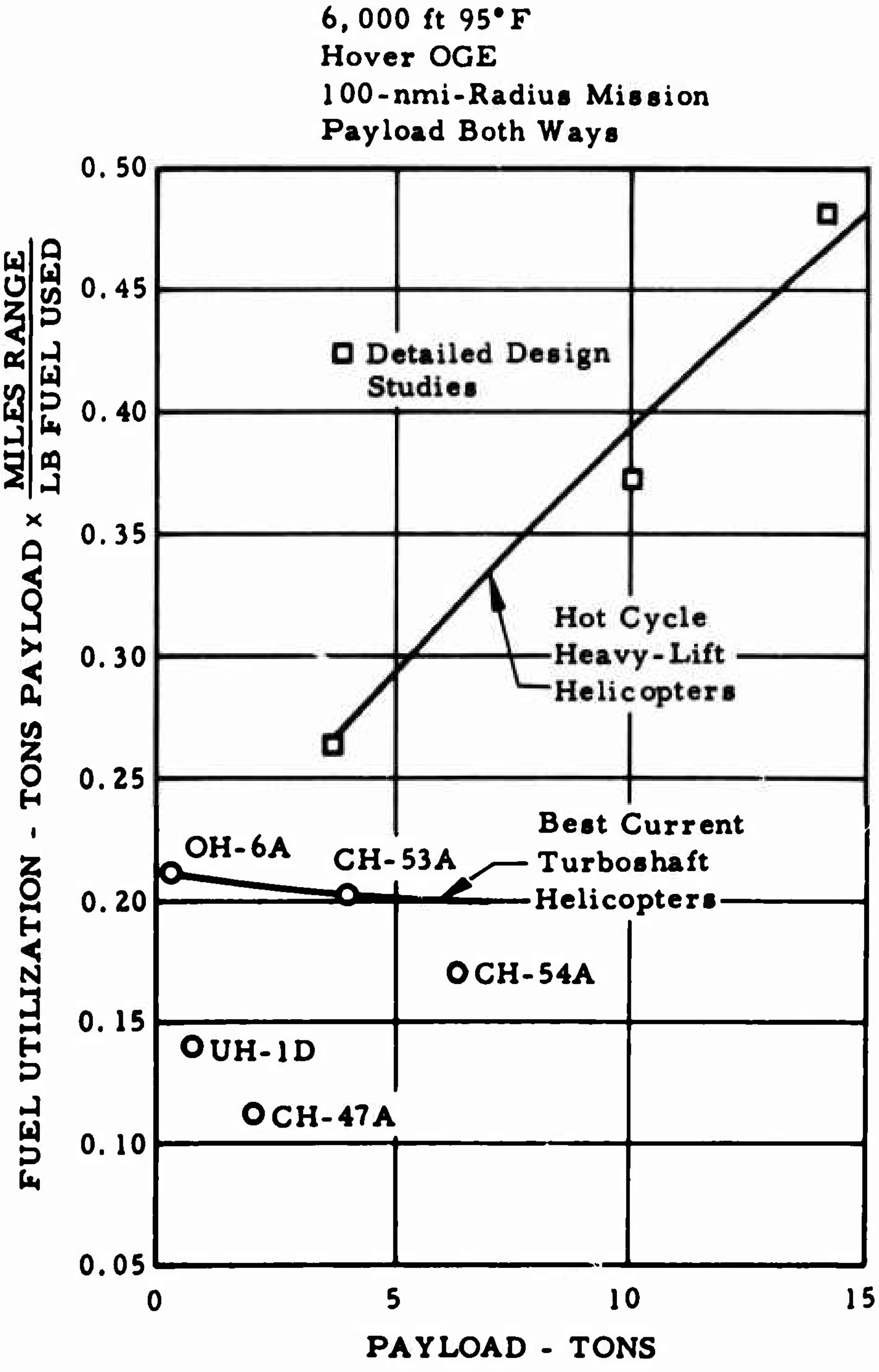


Figure 14. Fuel Utilization Versus Payload - Transport Mission.

## STABILITY AND CONTROL CHARACTERISTICS

This section presents the stability and control characteristics of three basic configurations considered for the Hot Cycle heavy-lift helicopter shown previously in Figures 5, 6, and 7.

Efficient utilization of the heavy-lift helicopter, particularly in the external load-carrying conditions requiring pickup, transport, and precise placement of large and heavy loads, dictates that the helicopter possess good handling characteristics under various flight conditions. To ensure this capability of precision flying for the Hot Cycle heavy-lift helicopter, the stability and control requirements of MIL-H-8501A have been considered as a minimum for this design study. The rotor system has been designed to incorporate large blade-flapping hinge offset (4.2-percent blade radius) to provide the high control power and rotor damping necessary for the required good handling characteristics.

In summary, the stability and control analysis has shown the following:

1. With the proposed Hot Cycle rotor design, the heavy-lift helicopter in hover and low-speed flight will possess excellent handling characteristics in pitch and roll, superior to those required by MIL-H-8501A.

2. For cruise flight, the horizontal stabilizer has been sized to provide good longitudinal static and maneuver stability characteristics.

3. The vertical stabilizer has been sized to provide stable directional stability in cruise flight. In hover and forward flight, the proposed yaw fan thrust of 700 pounds per inch of pedal will provide excellent yaw response, superior to that required by MIL-H-8501A.

Since the handling characteristics of each configuration are interdependent on its loading condition (internal or external loading), the two primary mission modes have been considered for each configuration. The configurations and loading conditions investigated for this design study are as follows:

1. Minimum streamline fuselage (close packaged engines on top of the fuselage - configuration 2)

    a. 20-ton external loading (single-point sling)
    b. 7-ton - internal loading capability

2. Streamline fuselage with laterally-located pylon-mounted engines (configuration 3)

    a. 20-ton external loading (single-point sling)
    b. 12-ton transport - internal loading

3. Crane-type fuselage (configuration 4)
    20-ton external loading (single-point sling)

Tables III and IV present the helicopter dimensional data (common to all configurations) and mass properties used in the stability and control analysis of the design configurations considered for the Hot Cycle heavy-lift helicopter.

## HOVER FLIGHT CHARACTERISTICS

### HANDLING CHARACTERISTICS IN PITCH

Table V presents the hover handling characteristics in pitch for the three basic configurations of the heavy-lift helicopter. The results are also compared with the handling requirements of MIL-H-8501A. As can be seen, the angular velocity damping of the heavy-lift helicopter is superior to that required by MIL-H-8501A for all configurations investigated. It can also be seen that the angular response in pitch per inch of control displacement is three to four times greater than the MIL-H-8501A requirements. For full control displacement from trim, the ratio of angular response available to that required is even greater. This is primarily because of the high control power provided by the large blade flapping hinge offset of the proposed Hot Cycle rotor design. The combination of high rotor damping and control power will provide the heavy-lift helicopter (HLH) with excellent handling characteristics in pitch.

### HANDLING CHARACTERISTICS IN ROLL

Table VI presents the hover handling characteristics in roll for the heavy-lift helicopter and compares the results with MIL-H-8501A. Again, as in pitch, the angular velocity damping in roll for the HLH is far superior to that required by MIL-H-8501A. In fact, for all configurations investigated, the damping in roll is approximately twice that required by MIL-H-8501A.

# TABLE III
## DIMENSIONAL DATA

| | |
|---|---|
| **Rotor** | |
| Diameter | 90.0 ft |
| Disc area | 6,359 sq ft |
| Chord | 60.0 in. |
| Solidity | 0.106 |
| Blade twist | -8° |
| Number of blades | 3 |
| $\delta_3$ | 0 |
| Flapping hinge offset (% blade radius) | 4.2% |
| Rotor shaft tilt, line ⊥ to fuselage WL | 5° fwd |
| Centrifugal force of rotor blade ($\gamma_T$ = 750 ft/sec) | 221,040 lb/blade |
| Airfoil section | NACA 0018 from root to 75% radius;<br>NACA 0014 from 75% radius to blade tip. |
| Rotor tip speed, hovering | 750 ft/sec |
| Rotor tip speed, forward flight | 675 ft/sec |
| **Horizontal Tail** | |
| Span | 324 in. |
| Tip chord | 44.4 in. |
| Root chord | 88.8 in. |
| Area | 150.0 sq ft |
| Leading edge sweep | 30° |
| Geometric aspect ratio | 4.8 |
| Incidence of tail with respect to fuselage WL | -5° (nose down) |
| Airfoil section | NACA 0012 |
| **Vertical Tail** | |
| Span | 200.0 in. |
| Tip chord | 87.8 in. |
| Root chord | 163.0 in. |
| Area | 175.0 sq ft |
| Leading edge sweep | 20° |
| Geometric aspect ratio | 1.58 |
| Airfoil section | NACA 0012 |
| **Control Travel** | |
| **Longitudinal Stick** | |
| Full aft to full forward | 12.0 in. |
| Cyclic pitch range | 14° fwd, 14° aft |
| **Lateral Stick** | |
| Full left to full right | 12.0 in. |
| Cyclic pitch range | 6° left, 6° right |
| **Collective Pitch Control Stick** | |
| Full down to full up | 9.5 in. |
| Cyclic pitch range (at 0.75R) | 1° to 14° |
| **Pedals** | |
| Full right to full left | ±3.25 in. |
| Pitch range | ±25° |

## TABLE IV
## HEAVY-LIFT HELICOPTER MASS PROPERTIES

| | | Center of Gravity (in.) | | | Inertia (slug feet$^2$) | | |
|---|---|---|---|---|---|---|---|
| Condition | Weight (lb) | Fuselage Station | Butt Line | Water Line | Pitch | Roll | Yaw |
| Configuration 2 | | | | | | | |
| 20-ton external - sling | 62,900 | 377.2 | 0 | 115.9 | 203,361 | 37,418 | 172,970 |
| 7-ton transport - internal | 39,900 | 377.6 | 0 | 121.3 | 279,097 | 39,562 | 250,988 |
| Configuration 3 | | | | | | | |
| 20-ton external - sling | 65,700 | 385.8 | 0 | 113.9 | 232,812 | 50,383 | 204,233 |
| 12-ton transport - internal | 52,700 | 386.8 | 0 | 110.4 | 364,095 | 56,994 | 335,869 |
| Configuration 4 | | | | | | | |
| 20-ton external - sling | 66,900 | 330.2 | 0 | 174.1 | 235,555 | 49,303 | 210,644 |

## TABLE V
## HOVER HANDLING CHARACTERISTICS IN PITCH

| | Angular Velocity Damping (ft-lb/rad/sec) | | Angular Response (Degree of Angular Displacement at End of One Second Per Inch Control) | |
|---|---|---|---|---|
| Heavy-Lift Helicopter Configuration | Heavy-Lift Helicopter | Minimum Requirement per MIL-H-8501A | Heavy-Lift Helicopter | Minimum Requirement per MIL-H-8501A |
| Configuration 2 | | | | |
| 20-ton external load - sling | 73,630 | 41,995 | 3.88 | 1.13 |
| 7-ton transport - internal | 58,386 | 51,896 | 3.57 | 1.31 |
| Configuration 3 | | | | |
| 20-ton external load - sling | 76,060 | 46,200 | 5.39 | 1.11 |
| 12-ton transport - internal | 67,570 | 63,510 | 3.24 | 1.19 |
| Configuration 4 | | | | |
| 20-ton external load - sling | 71,170 | 46,750 | 4.78 | 1.10 |

## TABLE VI
## HOVER HANDLING CHARACTERISTICS IN ROLL

| | Angular Velocity Damping (ft-lb/rad/sec) | | Angular Response (Degree of Angular Displacement at End of One Second Per Inch Control) | |
|---|---|---|---|---|
| **Heavy-Lift Helicopter Configuration** | Heavy-Lift Helicopter | Minimum Requirement per MIL-H-8501A | Heavy-Lift Helicopter | Minimum Requirement per MIL-H-8501A |
| Configuration 2 | | | | |
| 20-ton external load - sling | 73,630 | 28,750 | 3.38 | 0.68 |
| 7-ton transport - internal | 58,386 | 29,736 | 2.69 | 0.78 |
| Configuration 3 | | | | |
| 20-ton external load - sling | 76,060 | 35,450 | 2.76 | 0.67 |
| 12-ton transport - internal | 67,570 | 38,510 | 2.31 | 0.72 |
| Configuration 4 | | | | |
| 20-ton external load - sling | 71,170 | 34,950 | 2.53 | 0.66 |

The angular response in roll for the HLH for all conditions investigated is approximately three times greater than the roll response requirements of MIL-H-8501A, and yet does not exceed the maximum allowable roll rate of 20 degrees per second per inch of stick of that specification. For full control displacement from trim, the roll response available is again superior to MIL-H-8501A requirements. This high control power and corresponding high rotor damping will provide the Hot Cycle HLH with excellent handling characteristics in roll.

## HANDLING CHARACTERISTICS IN YAW

Table VII presents the angular response in yaw for the three basic configurations of the HLH based on a common yaw fan thrust of 700 pounds per inch of pedal. As can be seen, in all configurations investigated, the angular response in yaw of the HLH exceeds the MIL-H-8501A requirements. Analysis also shows that the yaw response at the most critical azimuth angle, relative to a 35-knot wind, is superior to MIL-H-8501A requirements.

TABLE VII
HOVER HANDLING CHARACTERISTICS IN YAW

| Heavy-Lift Helicopter Configuration | Angular Response (Degree of Angular Displacement at End of One Second Per Inch Control)<br>Heavy-Lift Helicopter | <br>Minimum Requirement per MIL-H-8501A |
|---|---|---|
| Configuration 2 | | |
| 20-ton external load - sling | 5.41 | 2.75 |
| 7-ton transport - internal | 3.79 | 3.19 |
| Configuration 3 | | |
| 20-ton external load - sling | 4.61 | 2.71 |
| 12-ton transport - internal | 2.92 | 2.91 |
| Configuration 4 | | |
| 20-ton external load - sling | 4.46 | 2.70 |

The angular velocity damping in yaw for the Hot Cycle heavy-lift helicopters is low because of the relatively small size of the yaw fan, which is required only for yaw control. This characteristic, which is typical for all tip-driven helicopters, is not expected to produce any adverse handling characteristics based on company experience with the tip-driven XV-9A helicopter. The heavy-lift helicopter utilizing a yaw fan will have damping superior to that of the XV-9A (with yaw jet control) and will result in greatly improved handling characteristics.

## FORWARD FLIGHT CHARACTERISTICS

### LONGITUDINAL MANEUVER STABILITY

The longitudinal maneuver stability characteristics of the three basic configurations considered for the Hot Cycle heavy-lift helicopter were determined with the aid of Reference 13. Since the maneuver stability parameter angle of attack stability ($M_\alpha$) is dependent on cg location, the critical condition of maximum aft cg was considered. Figure 15 presents the results of the maneuver stability analysis for the HLH at $\mu = 0.30$ (forward flight speed of approximately 120 knots). As can be seen, the results show that all three configurations of the heavy-lift helicopter remain on the stable side of the boundary line for all of the representative flight conditions. Thus, the HLH will have good maneuver characteristics. This excellent longitudinal stability is primarily attributed to the relatively large horizontal tail provided in the design.

### DIRECTIONAL STABILITY

The combination of the yaw fan control and large vertical tail will provide the Hot Cycle heavy-lift helicopter with good directional stability and control characteristics in cruise flight.

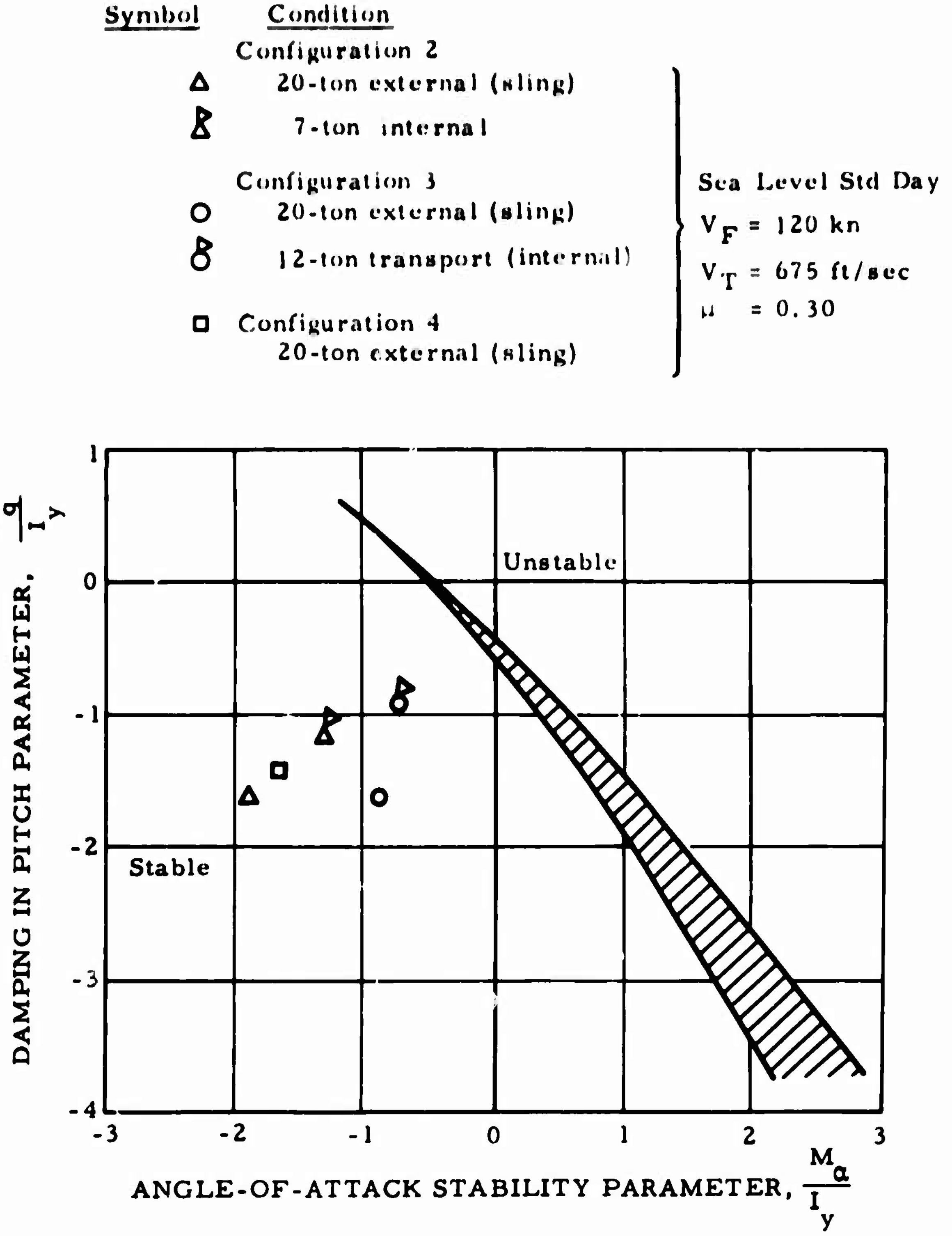


Figure 15. Longitudinal Maneuver Stability Criterion.

## ROTOR SYSTEM

The primary objective of this program was to define the optimum Hot Cycle rotor for the heavy-lift helicopter. This objective has been achieved by analyzing the results of a parametric study to determine the effect of varying the many characteristics of the rotor, such as rotor diameter, tilting or articulated type of hub, number and size of blades, tip speed, blade structural arrangement, internal or external flight controls, and airfoil shape. The rotor systems were further evaluated by considering them installed on several aircraft configurations. The results upon which the selection of the optimum rotor was based are discussed in detail in the Parametric Study section of this report. The study has indicated that a rotor as small as 80 feet in diameter, when installed on the minimum airframe, will result in a helicopter that will weigh approximately 18,000 pounds empty and will exceed all mission requirements of range and payload by 20 to 30 percent. However, for the integrated preliminary design, a larger diameter rotor -- 90 feet -- has been selected for disc loading considerations and as the rotor that is more nearly optimum for all of the aircraft configurations studied. The selected rotor is defined in Table VIII.

TABLE VIII
SELECTED OPTIMUM ROTOR CHARACTERISTICS

| | |
|---|---|
| Type of hub | Articulated |
| Flapping hinge offset | 22-1/2 in. |
| Lead-lag hinge | Blade station 66-1/2 |
| Controls | External |
| Rotor diameter | 90 ft |
| Blade chord | 60 in. |
| Blade section | |
| Root to 0.75R | NACA 0018 |
| 0.75R to tip | NACA 0014 |
| Blade spar location | 25% chord |
| Blade duct configuration | Figure-8 |
| Rotor tip speed | |
| Hover | 750 ft/sec |
| Cruise | 675 ft/sec |

## HUB DESIGN

The parametric study considered two basic types of hub: the fully articulated hub with offset flapping hinges and the tilting type as used on the

XV-9A Hot Cycle helicopter with hub restraint added to provide the necessary control power. Two variations of these basic types -- namely, with internal or external controls -- were also evaluated. A rigid-type rotor was also considered but was abandoned because of its inherent structural problems and the resultant weight increase. The hubs evaluated are described in more detail in the Parametric Study section of this report. The articulated hub was selected for the optimum rotor because of its clear-cut advantage over the tilting type in the areas of light weight and lower drag. The external controls were selected because they were determined to be lighter, less complicated, and more rigid.

The selected hub is shown in Figure 16. This configuration allows coaxial gas ducts to be routed uninterrupted up through the center of the hub assembly. As the ducts approach the blade level, they are split off into three pairs of ducts, one duct from each engine. This arrangement allows the engine output to be completely separated from gas generator to blade tip nozzles.

Located just outside of the coaxial ducts is the rotating housing portion of the hub. Thermal protection is provided by insulation applied to the ducts and by centrifugally pumped cooling airflow between the insulation and housing. A ring gear is installed on the lower rim of this housing to drive the accessory gearbox.

A pair of angular contact bearings offset vertically is used to carry rotor lift loads and moments from the rotating housing into the stationary mast, which in turn is attached to the fuselage through a tubular truss. The vertical offset of the bearings, plus the additional effective distance supplied by the contact angle, provides a generous couple arm to accommodate applied rotor moments. Lift is taken by the lower pair of bearings, and any download is reacted through the single upper bearing. Bearings are lubricated by a circulating oil system.

The stationary mast, in addition to its function as the rotor support, acts as the guide and sliding surface for the spherical bearing on which the swashplate tilts for cyclic inputs and moves vertically for collective motion. The swashplate is of conventional configuration, utilizing an angular contact bearing assembly to provide for the loads between the rotating and stationary swashplates.

The main structural members in the blade retention system and hub assembly provide a direct load path for the centrifugal force and lift loads from the three blades. The retention system consists of a lead-lag strap pack that attaches the inboard end of each blade spar to the flapping

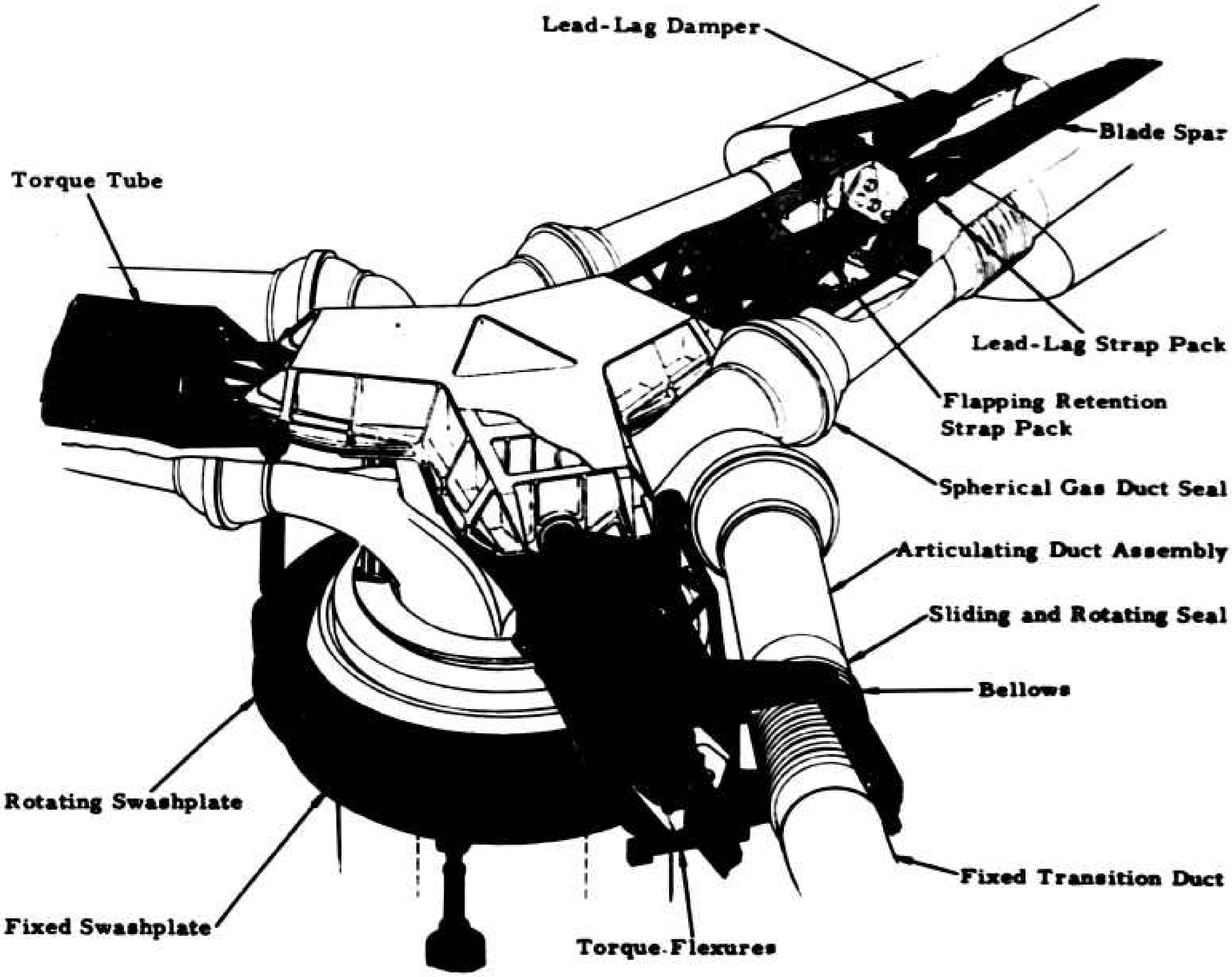


Figure 16. Articulated Hub With External Controls.

retention strap pack and allows lead-lag motion of the blade. The flapping strap packs are attached to the hub plate, wherein the centrifugal loads from the three baldes are effectively cancelled out. Only the unbalanced lift and in-plane loads remain to be carried through the support attachment to the mast. A torque box extends from the flapping axis out to the lead-lag hinge point at each blade. This torque box transmits feathering motion from the swashplate to the blade. The torque box is connected torsionally to the blade across the lead-lag hinge points through two flexures offset vertically to provide the torsional load path.

The lead-lag hydraulic damper is installed between the blade leading edge structural member and the torque box. Three stages of damping are provided, so that damping is increased in steps as the lead-lag oscillation increases.

## BLADE DESIGN

The blade designs considered in the parametric study were dictated to a large extent by duct configuration. Essentially, the parametric study resolved the tradeoff between duct area and blade weight for the different duct shapes evaluated. Three basic duct shapes were considered. The first configuration considered was the elliptical-shaped ducts as used on the XV-9A Hot Cycle helicopter, where the ducts were an integral part of the blade segment; the second, round ducts; and third, figure-8 ducts. The figure-8 ducts were selected as the most efficient configuration. A more detailed description of these blade duct configurations is to be found in the Parametric Study section of this report.

The structural arrangement of the selected blade is made up of a single spar, leading edge member, and segmented assemblies of sandwich-type skin and ribs joined spanwise by flexible couplings (Figure 17). The pairs of gas ducts are routed through the blades, one forward of the spar and one aft. A segmented trailing edge fairing completes the blade structure. The spar is located on the 25-percent chord and extends the full length of the blade from the lead-lag flexure on the inboard end to the cascade at the blade tip.

The spar area required at each spanwise blade station is determined by and is proportional to the centrifugal force. The flapwise stiffness, that is, moment of inertia, required at each spanwise station is determined by the ground flapping condition. The spar area is apportioned at each spanwise station to meet, but not exceed, the required stiffness. Exceeding the required flapwise stiffness would result in undesirably high in-flight flapwise bending moments. All flapwise shear and moments are taken by the spar. The required chordwise balance weight, located in

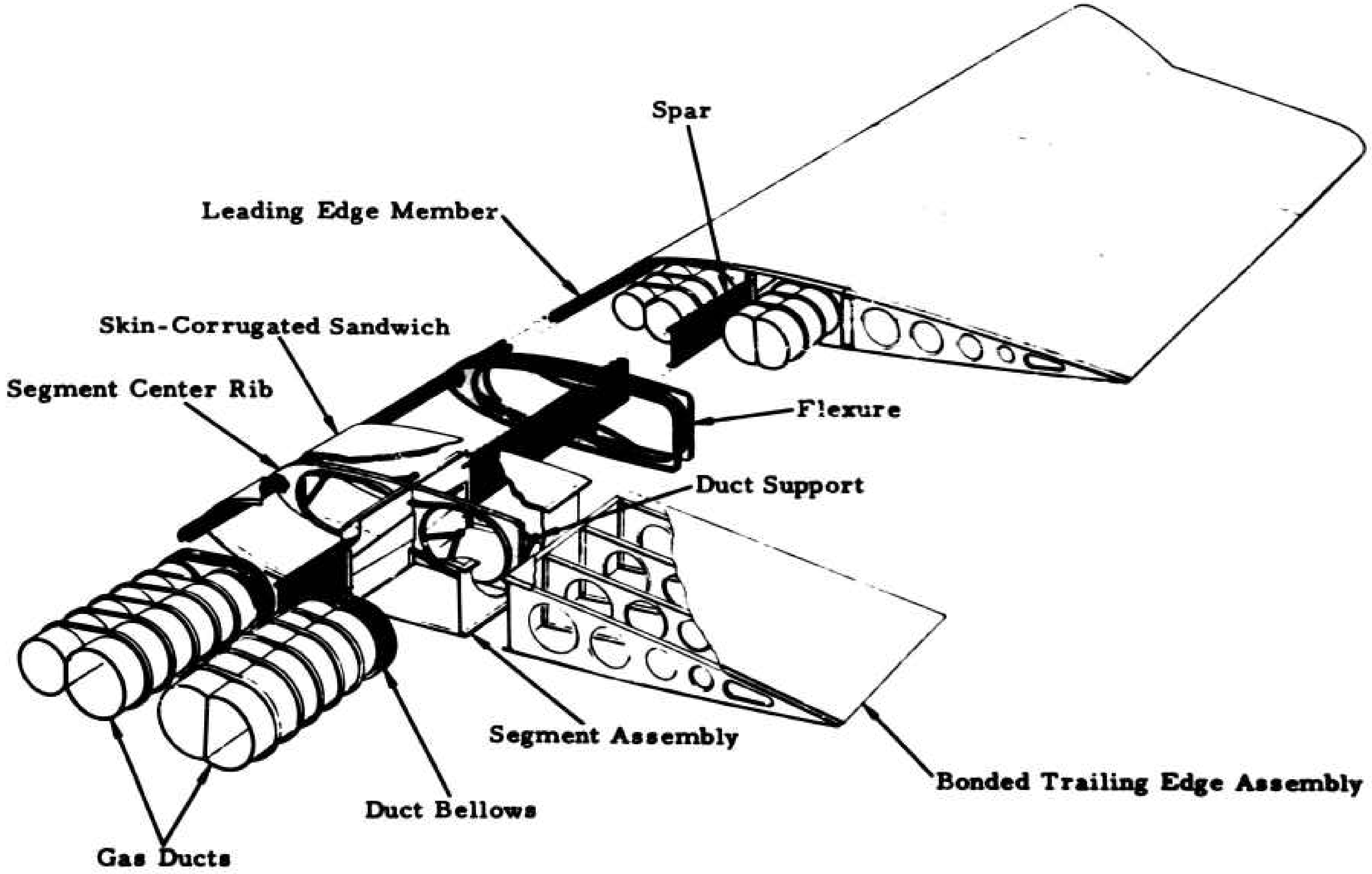


Figure 17. Blade Assembly.

the leading edge, is utilized as a continuous structural member extending from the lead-lag damper at the inboard end to the blade tip. The leading edge member is designed to take chordwise shear and, coupled with the spar, provides a load path for the blade chordwise moment.

The spanwise segments are approximately 20 inches in length and are made up of corrugated titanium skin assemblies and Inconel 718 ribs. The flexures join the segments to each other to provide a load path for blade torsion as well as to provide the necessary flexibility to prevent bending stresses from being induced into the skin panels. Trailing edge fairing segment assemblies are also interrupted spanwise for the same reason, and are fabricated from thin-gage aluminum skin bonded to internal ribs, a configuration similar to the XV-9A.

## FLIGHT CONTROL DESIGN

Two basic flight control configurations were considered. One configuration utilized a swashplate located below the rotor with the push rods extending up through the center of the gas ducts to walking beams that were connected to the blade lift links. The other configuration used a swashplate assembly large enough to be installed outside of the ducts and hub structure with the lift link attached directly between the swashplate and blade pitch arm. This latter configuration was selected because of its greater rigidity, simplicity, and resulting lower weight. It also required a smaller fairing, inasmuch as the walking beams increased the size of the required fairing. Three hydraulic servo-controlled cylinders power the flight controls. They are operated in such a manner that for collective pitch they act in unison and for cyclic pitch they act differentially.

# PROPULSION SYSTEM

The design of the propulsion system places emphasis on simplicity, reliability, and safety in an easily maintainable twin-engine installation. These factors are inherent in the Hot Cycle propulsion system, in which high-energy gas is diverted from the engine exhaust up through the hub to the tip of each blade, where it is exhausted to drive the rotor (Figure 18). Conversion of the basic helicopter propulsion system to the compound helicopter propulsion system can be accomplished in the manner shown by Figure 19.

## DESIGN CHARACTERISTICS

The primary advantage of the Hot Cycle propulsion system is its simplicity, with the resulting advantages of light weight and reliability gained by the elimination of many heavy and complex dynamic components required by other types of propulsion systems. The increased reliability achieved is a significant feature of the Hot Cycle propulsion system. The extreme scatter of failure lifetimes found in conventional drive system elements, such as bearings, gears, couplings, shafts, and clutches, is well recognized throughout the rotary-wing and propulsion industries. Conversely, the low incidence of failure with conservatively designed ducted propulsion systems has been well established, particularly in jet-engine technology. Thus, comparison with the more complex shaft-driven helicopters using the many complex dynamic components emphasizes the simplicity and resulting increase in reliability and safety of the Hot Cycle rotor.

## HOT GAS DUCT SYSTEM

The knowledge and experience gained from the successful XV-9A Hot Cycle program have been utilized in the design of the hot gas system. Additional factors of safety have been applied to the design of all pressurized hot gas ducting, and only materials with excellent corrosion resistance and crack-propagation resistance are used. Isolation of both thermal and structural strains is provided in the design of the hot gas ducting system, through proper design of mounts, reinforcements, and flexible joints. In addition to the isolation of both hot and cold components from a structural viewpoint, insulation and cooling airflow preclude any possible detrimental effects from the interaction of the hot and cold components. Further, thermal differential expansion in the primary structure is reduced by using materials of similar thermal expansion rates. Transient thermal effects in the hot gas system are minimized by detail design to assure even heat-up and cool-down of the components. The materials used in

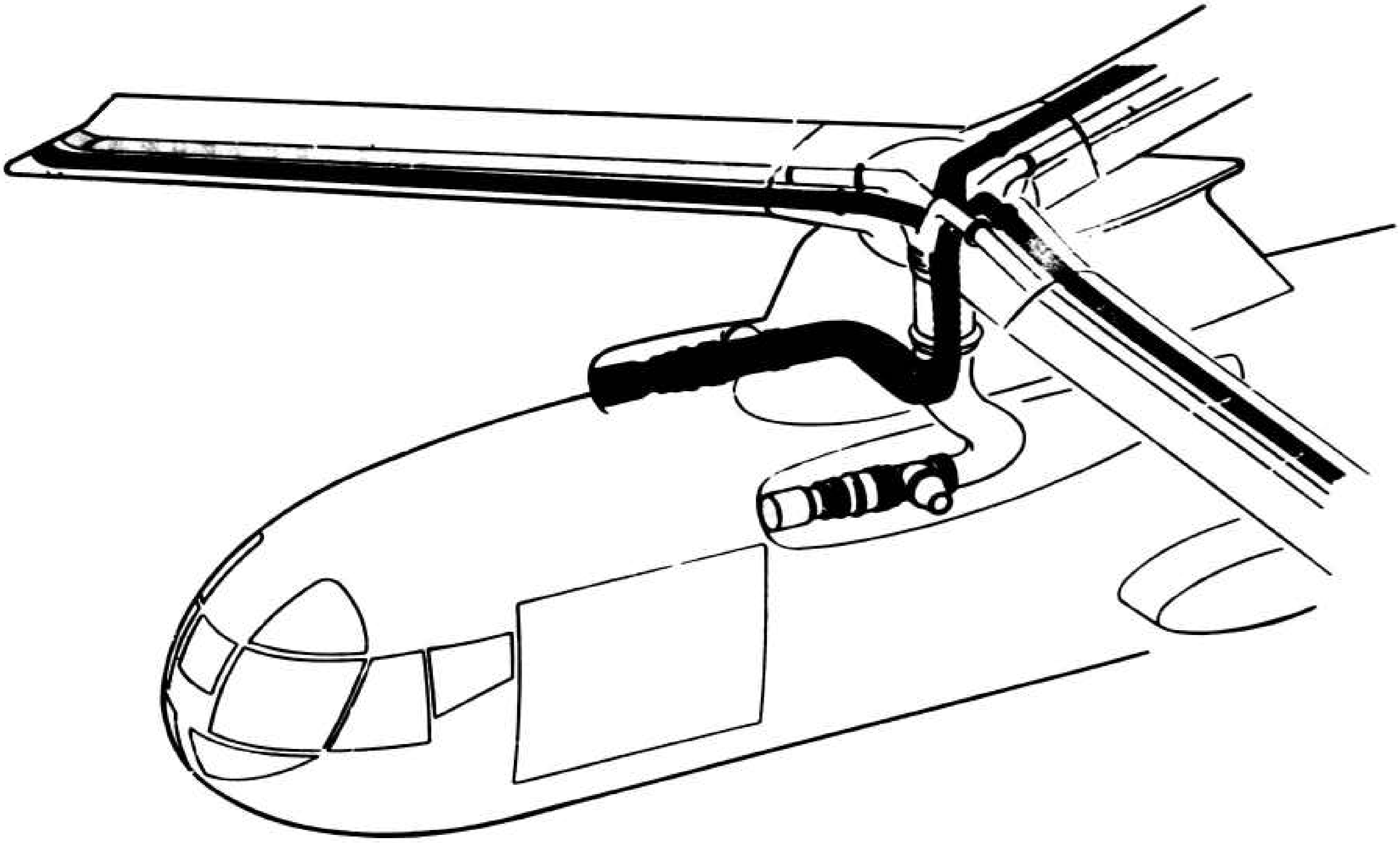


Figure 18. Propulsion System - Helicopter.

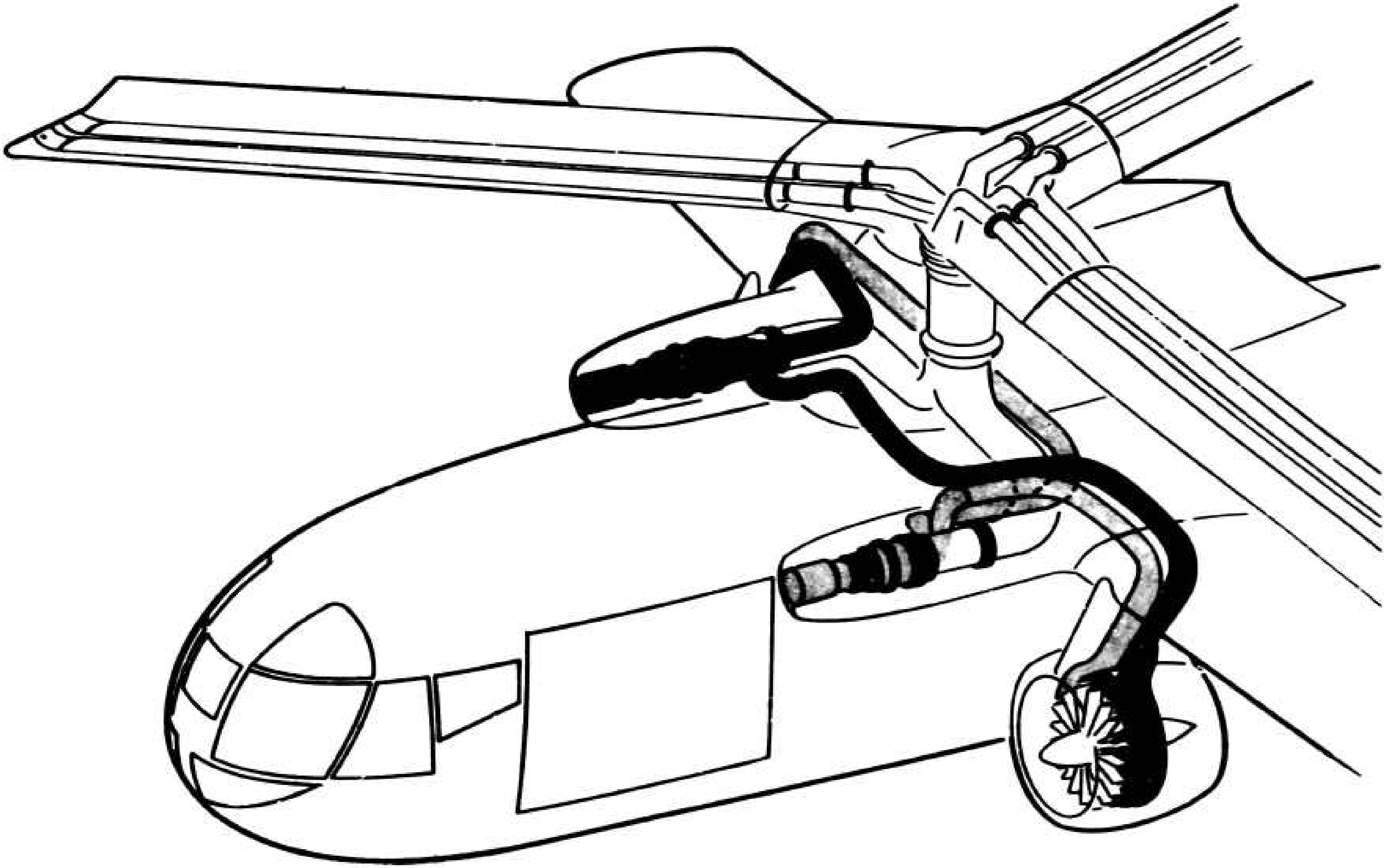
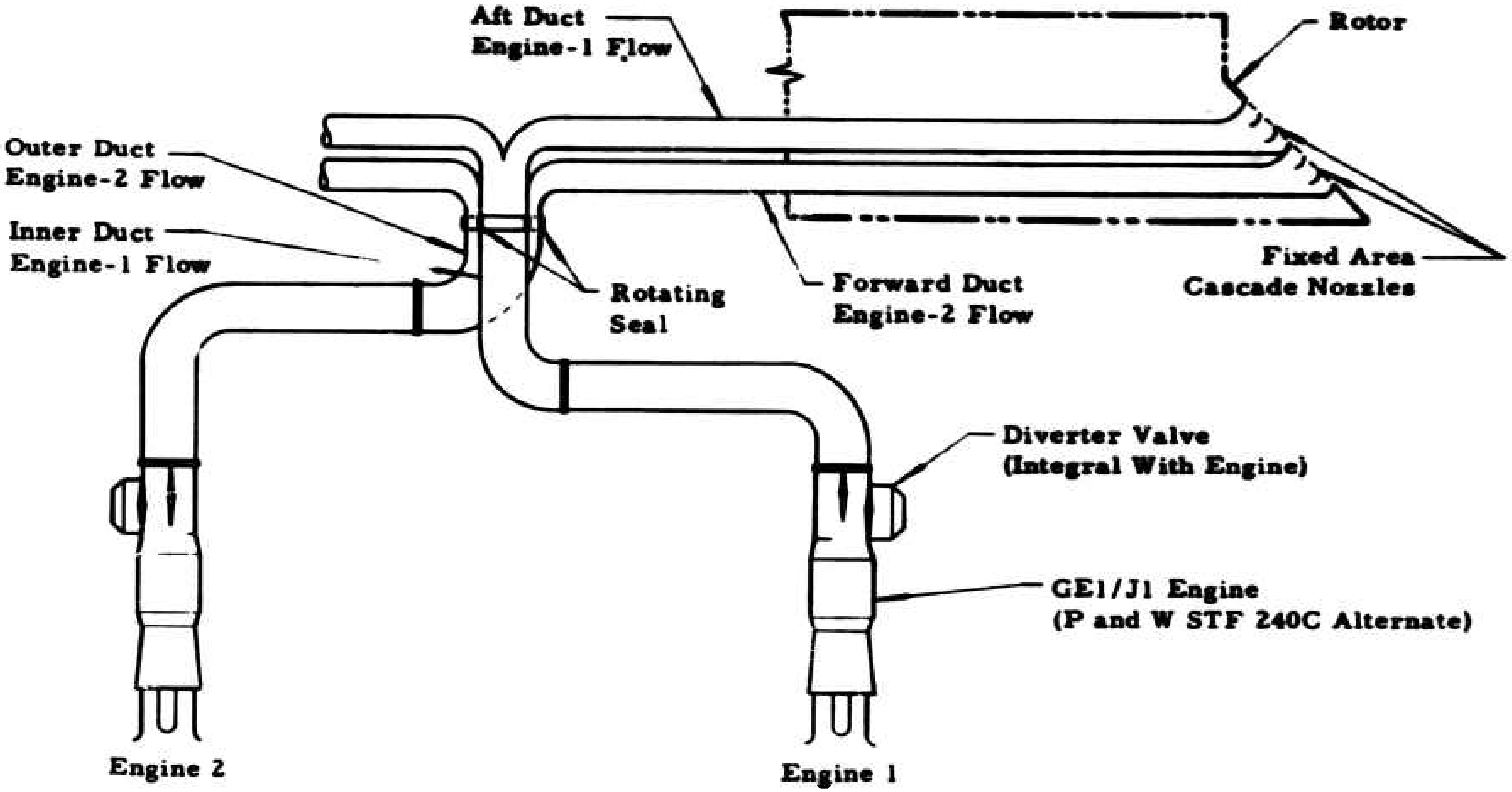


Figure 19. Propulsion System - Compound Helicopter.



Note: Diverter valves are positioned for rotor flow.

Figure 20. Hot Cycle Propulsion System Schematic.

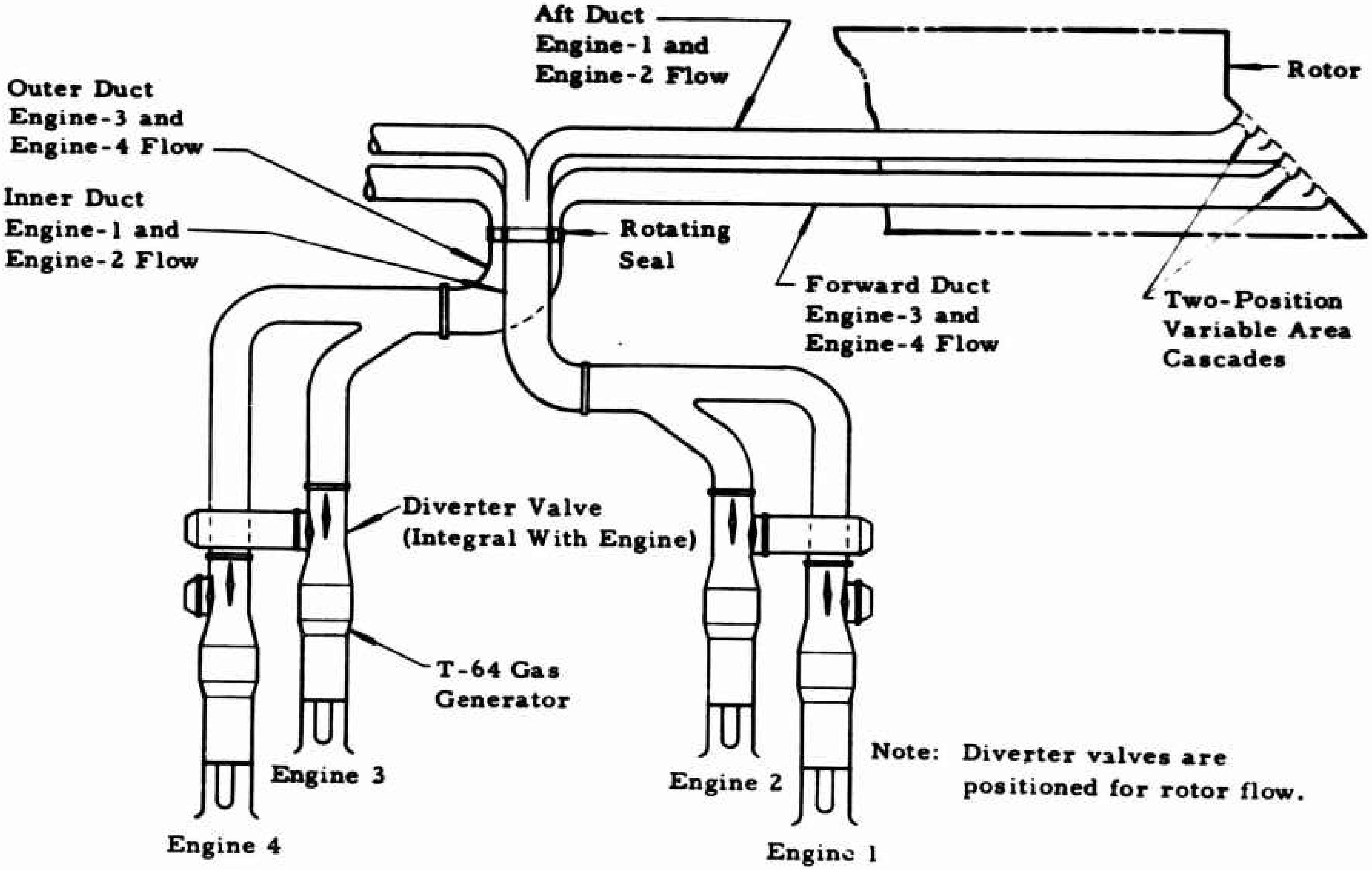
the hot components are standard production materials having wide usage in the jet engine industry and do not require the development of new technology.

The gas output of each engine for the two-engine configuration is ducted separately from engine to blade tip by coaxial gas ducts through the hub and through separate ducting in the blades, shown schematically in Figure 20. The use of separate outlets negates the problems associated with engine mismatch and thereby eliminates the necessity for power matching of engines and the need for blade-tip closure valves.

The exhaust gas flows from each engine through diverter valves that either divert the flow overboard for engine starting or direct the flow up through hub and blades for rotor operation. The engine and diverter valve are an integral unit. The seal above the diverter valve permits rotation between the stationary duct and its counterpart in the rotating system. As it emerges from the hub, the gas flows out three pairs of parallel ducts, separated to provide the necessary clearance for the hub and blade retention straps, through a transition section, and into the blade constant section. At the blade tip, the gas is turned 90 degrees by the cascade vanes and ejected at the trailing edge. All the ducts are insulated to reduce heat flux; bellows are utilized to allow for thermal expansion; articulating ducts and seal assemblies at the blade root are installed to permit blade feathering, flapping, and lead-lag motion.

## ENGINE INSTALLATION

Two engine installations were evaluated in the parametric study. The primary power source considered utilized two General Electric GE1/J1 gas generators, shown in Figure 20. An alternate installation utilizing four General Electric GE T64/S4B gas generators was also surveyed and is shown schematically in Figure 21. Subsequent to completion of the parametric study, data on the Pratt and Whitney STF240C gas generator has become available, and it appears to be interchangeable with the GE1/J1 without any major changes to the propulsion system installation or aircraft configuration.



Note: Diverter valves are positioned for rotor flow.

Figure 21. Propulsion System Schematic - Four-Engine.

# WEIGHTS

The favorable performance of the Hot Cycle rotor is a direct result of the simplicity and inherent light weight of this propulsion system. Since the propulsion system is lighter, the gross weight is lower and requires a smaller rotor, which results in an even lower gross weight. This compounding effect produces a low empty weight and a high payload-to-empty weight ratio.

The weight estimation for the Hot Cycle heavy-lift helicopter configurations noted in Table IX has been based on data compiled from analytical and statistical studies and was carried out in two parts. The first task was to develop weight equations for the parametric study from existing statistical data and preliminary layouts. The second task was to calculate the weight of the selected optimum rotor detail design. Upon completion of these tasks, it was found that the rotor weight as obtained by the equation using an estimated running blade weight was higher than the rotor weight as obtained by the detailed analysis. This difference was the result of refinement and optimization of blade design subsequent to the development of equations for the parametric study. Good agreement is obtained when the lower running blade weight of the optimized design is used in the weight equation. The detailed weight analysis summarized in Table X shows the selected rotor weight to be 5,440 pounds, and applying the parametric equations to the same rotor results in a weight of 5,475 pounds when the calculated running blade weight is used. The tail group, flight controls, and propulsion group weights for the preliminary design have been changed from those used in the parametric study to reflect a more realistic distribution of weight. A summary weight statement per MIL-STD-451 Part I may be found in Appendix I.

## SUBSTANTIATION OF WEIGHT EQUATIONS - HELICOPTER

The helicopter group weights and equations used in the parametric study and preliminary design are based on data compiled from analytical and statistical studies of numerous production and proposed helicopters. Conventional methods have been employed in arranging the various parameters used to obtain meaningful expressions that result in reasonable weight estimates. Also used to the greatest extent possible was the invaluable data and experience gained in the development of the Hot Cycle XV-9A research vehicle. The success achieved in obtaining reasonable correlation with actual data has verified the validity of the equations developed and presented herein.

## TABLE IX
## CONFIGURATION WEIGHT SUMMARY

Rotor radius = 45 ft
Chord = 60 in.
Design tip speed, $V_t$ = 750 fps
Ultimate load factor = 3.75 (heavy-lift mission)

| | Configuration | | |
|---|---|---|---|
| | 2 | 3 | 4 |
| Rotor group | 5,440 | 5,440 | 5,440 |
| Tail group | 970 | 992 | 998 |
| Hover-yaw group | 193 | 197 | 198 |
| Fuselage | 2,843 | 3,615 | 3,575 |
| Alighting gear** | 2,185 | 2,300 | 2,852 |
| | | (2,810*) | |
| Flight controls | 1,414 | 1,445 | 1,454 |
| Hydraulic and pneumatic | 711 | 731 | 735 |
| Electrical | 742 | 749 | 752 |
| Propulsion (includes 2 each GE-1 engines) | 2,971 | 2,971 | 2,971 |
| Instruments | 180 | 180 | 180 |
| Electronics | 150 | 150 | 150 |
| Furnishings and equipment | 300 | 300 | 300 |
| Air conditioning and anti-icing | 100 | 100 | 100 |
| Cargo-handling equipment | 1,400 | 1,400 | 1,400 |
| WEIGHT EMPTY | 19,599 | 20,570 | 21,105 |
| | | (21,080*) | |
| Crew (3-man) | 600 | 600 | 600 |
| Crew kits | 50 | 50 | 50 |
| Oil | 30 | 30 | 30 |
| Unusable fuel | 100 | 100 | 100 |
| OPERATING WEIGHT | 20,379 | 21,350 | 21,885 |
| | | (21,860*) | |
| Heavy-Lift Mission: | | | |
| Payload (20-ton) | 40,000 | 40,000 | 40,000 |
| Fuel | 3,901 | 4,131 | 4,312 |
| GROSS WEIGHT | 64,280 | 65,481 | 66,197 |
| | | (65,991*) | |
| Transport Mission: | | | |
| Payload (12-ton) | 24,000 | 24,000 | 24,000 |
| Fuel | 7,881 | 6,884 | 8,272 |
| GROSS WEIGHT | 52,260 | 52,234 | 54,157 |
| | | (52,744*) | |

*Retractable landing gear.

**Landing gear weight was based on the maximum gross weight associated with a limit load factor of 2-1/2 g. This was obtained by dividing the product of the mission gross weight x load factor by 2-1/2. In all cases, the ferry mission was critical.

## TABLE X
## ROTOR GROUP SUMMARY

| | | Weight (lb) |
|---|---|---|
| **Blade*** | | |
| Constant section | | 848. 8 |
| Transition section | | 57. 2 |
| Torque box | | 92. 5 |
| Tension strap (flapping) | | 40. 5 |
| Lead-lag flexure | | 21. 3 |
| Stub spar | | 28. 6 |
| Sealant, finish, etc | | 5. 3 |
| Blade to hub truss | | 26. 2 |
| Droop stop | | 18. 2 |
| Damper | | 66. 0 |
| Damper arm | | 4. 5 |
| Articulated duct | | 68. 0 |
| Fairing over torque box | | 16. 0 |
| Damper attachment | | 4. 8 |
| | Total 1 blade | 1, 298 |
| | | x 3 |
| | Total 3 blades | 3, 894 |
| **Hub and Shaft** | | |
| Hub | | 369 |
| Hub support | | 37 |
| Droop stop support | | 27 |
| Fixed shaft | | 334 |
| Rotating shaft | | 280 |
| Upper bearing, seal, retainer | | 101 |
| Lower bearing, seal, retainer | | 356 |
| Feathering bearings | | 17 |
| Hub fairing | | 24 |
| | Total hub | 1, 545 |
| | Total rotor group | 5, 439 |

*Blade balanced chordwise 23 percent at the tip to 28 percent at the root.

## MAIN ROTOR GROUP WEIGHT EQUATION

The main rotor group equation is based on the statistical and analytical study performed by HTC-AD and published in Reference 14. The equation developed is a power function expression relating total rotor group weight to the total "idealized" blade weight ($W_{BU}$) and rotor tip speed ($V_t$). The "idealized" blade weight is defined as the weight of the blade less the weight of the retention system, root fittings, doublers, and so forth. These data were obtained or determined from published detailed weight statement reports of numerous helicopters, based on actual or calculated weights.

A power function analysis was performed on the basis of these data, resulting in the following equation that gives the best fit curve for the plotted points of Figure 22:

$$W_r = B\left(\frac{bW_{BU}}{1,000}\right)^{0.896}\left(\frac{V_t}{700}\right)^{0.80} \qquad (1)$$

where

$W_r$ = total rotor group weight, lb

b = number of blades

$W_{BU}$ = ideal blade weight, lb per blade

$V_t$ = rotor tip speed, maximum power on, ft per sec

B = 2,282 = constant for best fit of statistical data

This equation is used as the basis for establishing the relationship of the total rotor group weight to blade weight for each of the Hot Cycle rotor configurations investigated in the parametric study. An estimated rotor size of 94-foot diameter was chosen; and through detailed design layouts and analysis, estimated weights were obtained for the blades, retention system, hub, and rotating controls. From this data, with idealized blade weight ($W_{BU}$) and total rotor group weight ($W_r$) being known quantities, the specific value of coefficient B was then determined as follows for the various hub and shaft configurations:

$$B = W_r \div \left(\frac{bW_{BU}}{1,000}\right)^{0.896} \qquad (2)$$

when tip speed ($V_t$) = 700 feet per second.

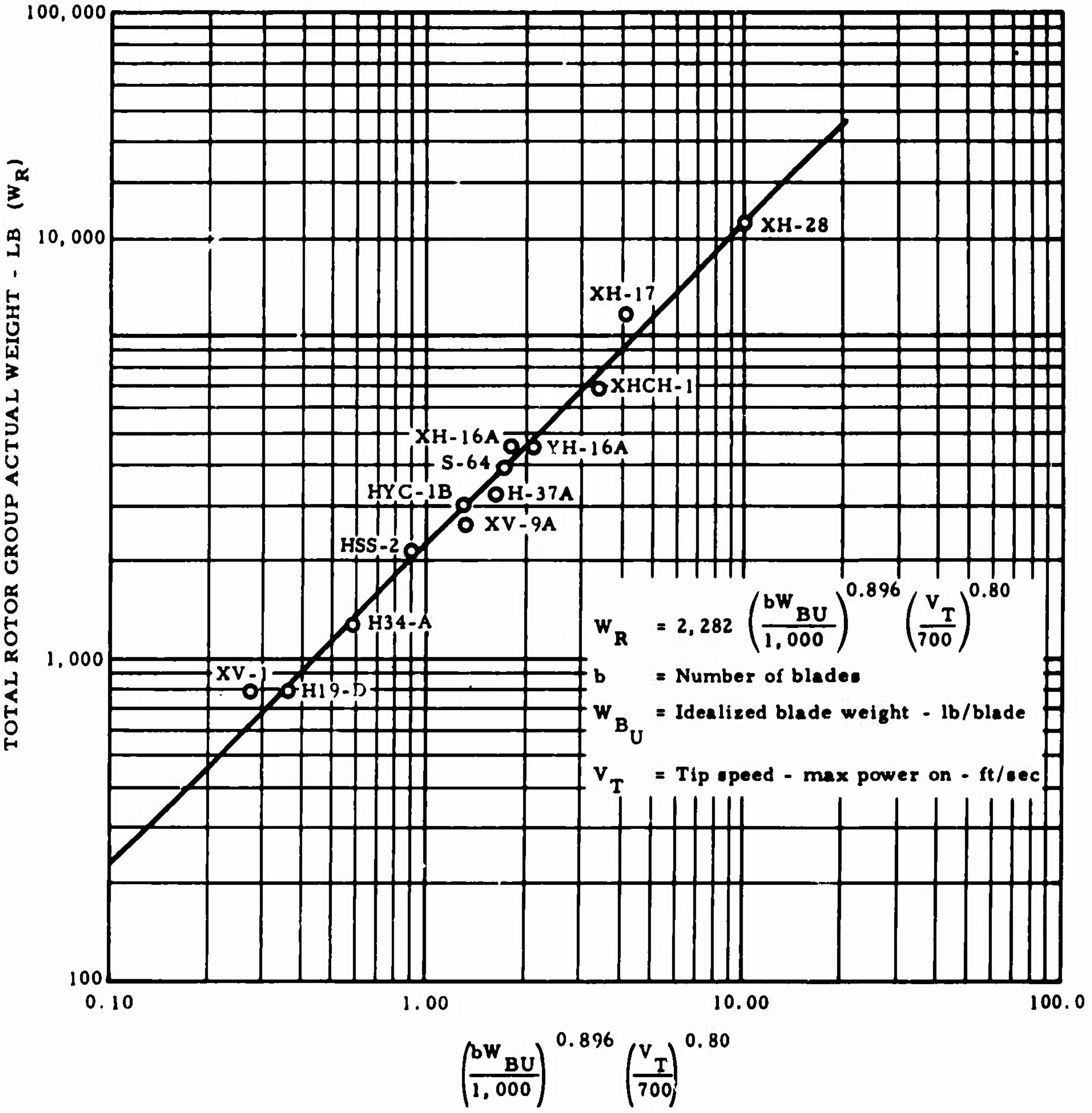


Figure 22. Total Rotor Group Actual Weight Versus Equation Results.

| Hub Type | Shaft | $bW_{BU}$* | $W_{hub + retention}$* | $W_r$ | $\left(\frac{bW_{BU}}{1,000}\right)^{0.896}$ | B |
|---|---|---|---|---|---|---|
| Tilting | Internal | 5,670 | 4,657 | 10,327 | 4.80 | 2,170 |
| Tilting | External | 5,670 | 5,957 | 11,627 | 4.80 | 2,440 |
| Articulated | Internal | 5,160 | 3,351 | 8,511 | 4.36 | 1,980 |
| Articulated | External | 5,160 | 3,190 | 8,350 | 4.36 | 1,940 |

The equivalent coefficient for the XV-9A unrestrained tilting-hub internal shaft rotor system is 2,130. The two-percent weight increase obtained in the tabulated value of B for the similar configuration above (tilting-internal) is attributed to the increased loads obtained in a restrained hub. The articulated hub system with its more direct load paths and lower chordwise loads is predictably lighter than the XV-9A system, by as much as nine percent.

## TAIL GROUP EQUATION

This equation includes only the weight of the horizontal and vertical surfaces required for flight stability and control.

Qualitative stability studies at HTC-AD coupled with actual experience derived in the testing of the XV-9A Hot Cycle research vehicle indicate that the total surface of the tail should be on the order of 5.50 square feet per 1,000 pounds of gross weight.

The equation used in the parametric study conservatively assumes a unit weight of 3.50 pounds per square foot and is derived as follows:

$$W_{tg} = \frac{5.50\ (3.50)}{1,000}\ W_g = 0.0193\ W_g \qquad (3)$$

where $W_g$ = design gross weight.

A later review of aircraft tail group data revealed that the unit weight used was too conservative. This conclusion is based on investigation of tail surface weights within the size, gross weight range, and speeds being considered. A more realistic unit weight of 2.75 pounds per square foot would result in a revision of the original equation as follows:

$$W_{tg} = \frac{5.50\ (2.75)}{1,000}\ W_g = 0.0151\ W_g \qquad (4)$$

---

*Calculations based on layouts (b = 3 blades).

## HOVER-YAW GROUP EQUATION

The equation used in the parametric study was derived by estimating the weight of a tail rotor system required for the tip-driven rotor vehicles being studied. The weights were sized from comparable components used on the OH-6A helicopter. The following data were used to obtain the estimated weight changes noted.

## DESIGN DATA

Comparison of Heavy-Lift Helicopter and OH-6A Hover-Yaw Systems

| | OH-6A | Heavy-Lift Helicopter |
|---|---|---|
| Rotor radius, ft | 2.13 | 4.00 |
| Number of blades | 2 | 6 |
| Design tip speed, fps | 694 | 720 |
| Blade chord, in. | 4.81 | 10.0 |
| Rotor solidity | 0.116 | 0.357 |
| Design gross weight, lb | 2,400 | 60,000 |
| Hover-yaw system weight, lb | 25.0 | 179.7 |

A rational analysis of the comparative data shown above was performed to obtain weights for the various heavy-lift helicopter hover-yaw components shown, based on the comparable OH-6A weights. The tail rotor and hub weights were determined from blade radius, solidity, and centrifugal force considerations. Drive shafting and coupling weights were based on ratios of transmitted torque and length. The gearbox weights were based on statistical weight studies performed by HTC-AD involving torque, gear ratios, and speeds as parameters. The resulting heavy-lift helicopter weights, obtained by the methods described, totaled 179.7 pounds.

Complexity in a hover-yaw group equation is not warranted, in view of its small influence on gross weight. Assuming, therefore, that the group weight varies directly with gross weight for the heavy-lift parametric study, the equation used is as follows:

$$W_{hy} = \frac{179.7}{60,000} W_g = 0.003 W_g \qquad (5)$$

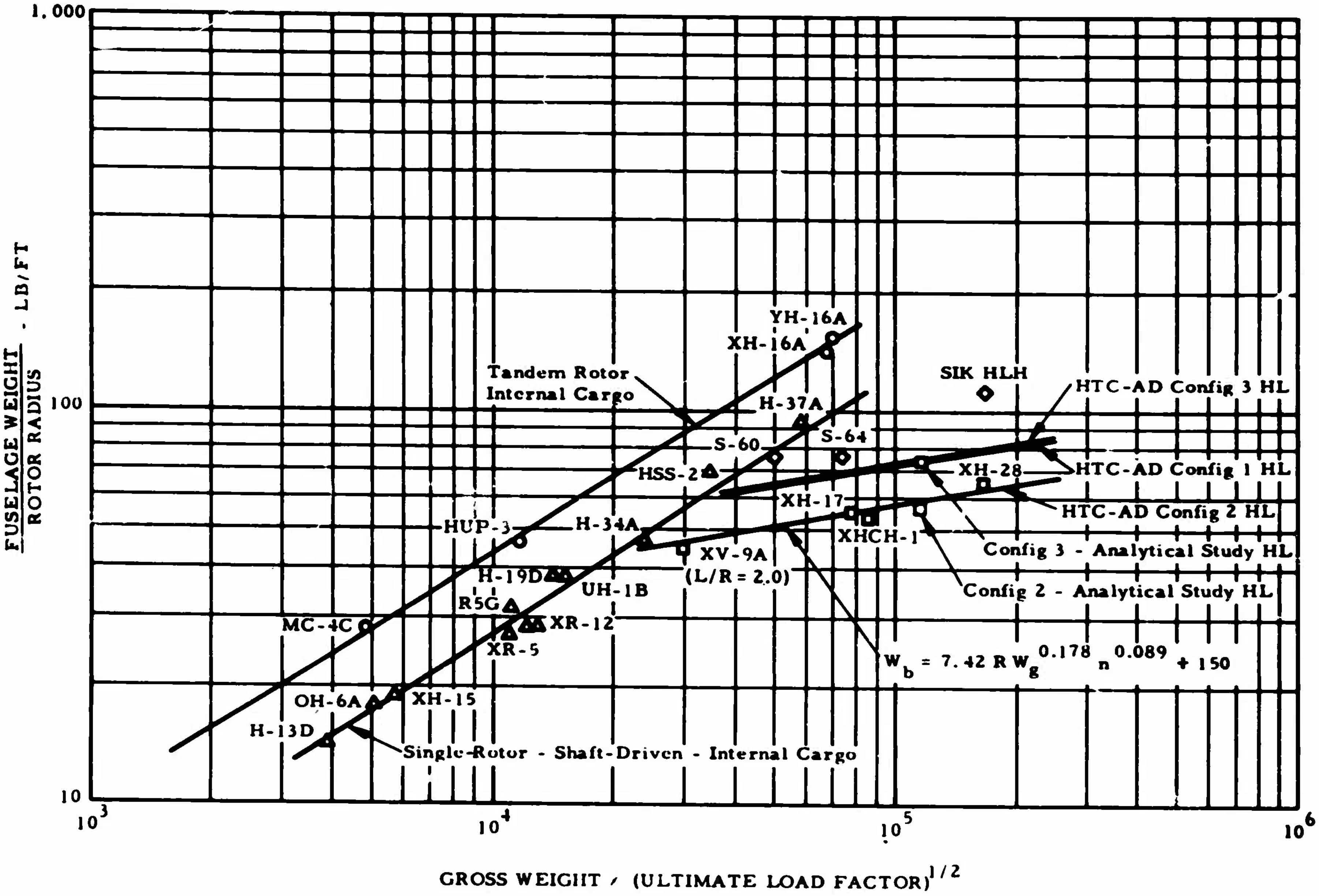


Figure 23. Fuselage Weight/Rotor Radius Versus Gross Weight x (Ultimate Load Factor)$^{1/2}$.

## FUSELAGE WEIGHT EQUATIONS

### Configuration 2 Fuselage Equation

The original development work of the fuselage weight equations used in the parametric study was performed by HTC-AD under contract AF33 (616)-3149 and published in Reference 15. This report illustrates the correlation between fuselage weight and the fundamental design parameters describing helicopter vehicles; namely, gross weight ($W_g$), rotor radius (R), and ultimate load factor (n). The three basic fuselage curves developed in the report are shown in Figure 23 for reference. The equation of interest in the parametric study for use on single-rotor tip-driven helicopters carrying cargo externally is:

$$W_b = 7.42\ R\ W_g^{0.178}\ n^{0.089} \tag{6}$$

The equation for configuration 2 streamlined fuselages with length-to-radius ratios of 2.0 was verified by a preliminary sizing from a structural analysis of the fuselage. A gross weight of 60,000 pounds and a rotor radius of 47 feet were assumed, using an ultimate load factor of 3.75. Floor weight was assumed as 1.5 pounds per square foot. The resultant weight distribution and integration are shown in Figure 24, and the results are plotted in Figure 23. The actual fuselage weight of the XV-9A Hot Cycle research vehicle is also plotted after being adjusted to a length-of-fuselage to rotor-radius ratio of 2.0 from 1.6. The points fall close to the fuselage equation curve, verifying its slope and intercept.

An additional 150 pounds was added to this, and the other equations, to account for the rotor mast fairing unaffected by parametric considerations. The final equation for the configuration 2 fuselage is then revised and used as follows:

$$W_b = 7.42\ R\ W_g^{0.178}\ n^{0.089} + 150 \tag{7}$$

The relatively low weight of the configuration 2 streamlined fuselage results primarily from the efficient structural shape. In addition, the fuselage is designed to transport a maximum of 7 tons of payload internally.

### Configuration 3 Fuselage Equation

The basic difference between this fuselage and configuration 2 is that the fuselage cross section is larger to allow for internal cargo capability to

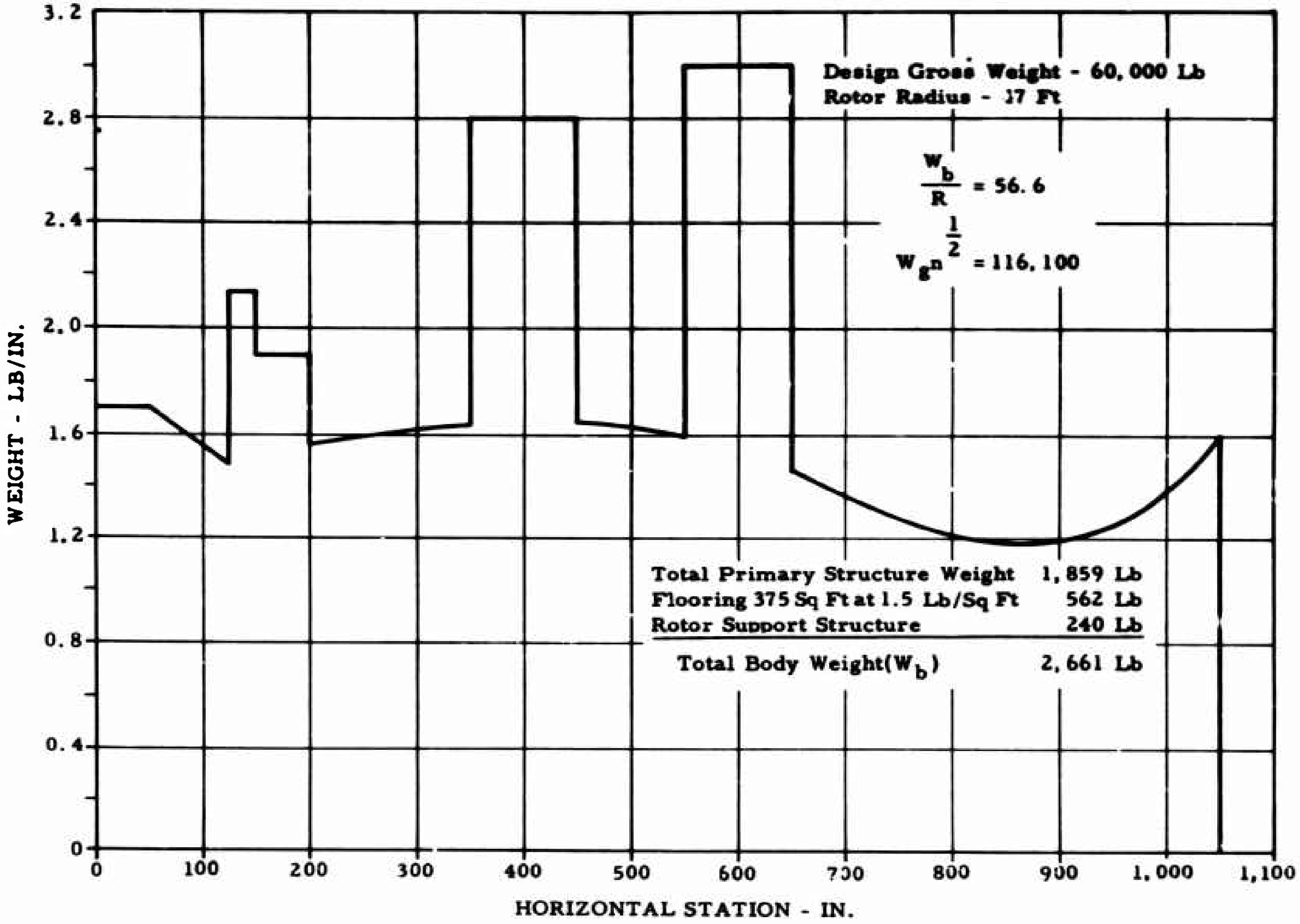


Figure 24. Primary Structure Weight Distribution - Configuration 2.

12 tons. The fuselage equation for configuration 3 was developed from the previous equation by including the effects of the larger fuselage. This was done on the basis of wetted area. On this basis, the configuration 3 fuselage weight would increase 23. 4 percent over that of a configuration 2 fuselage of similar length.

An increase in floor structure unit weight to 2. 00 pounds per square foot is allowed because of increased floor beam width and floor utilization.

The combined effects of these changes applied to the configuration 2 fuselage equation develop an equation for configuration 3 as follows:

$$W_b = 9.51 \, R \, W_g^{0.178} \, n^{0.089} + 150 \qquad (8)$$

An independent structural analysis similar to that performed on configuration 2 produced the weight distribution curve shown in Figure 25. A plot of this weight in Figure 23 shows that close agreement exists between the two methods employed to obtain a fuselage weight.

Configuration 1 and Configuration 4 Fuselage Equation

The configuration 1 and configuration 4 crane fuselages are identical in all respects except in the manner in which the mission payloads are carried; configuration 1 uses a detachable cargo pod in operation.

A weight comparison was made of three crane-type fuselages: the XH-17, S-60, and S-64. Also used was the weight data obtained from detailed design efforts by HTC-AD on the XH-28 heavy cargo crane. The weight study applied the same parameters used to develop the previously discussed equations. The results are plotted in Figure 23.

Two primary design differences are involved in these crane-type ships. First, the shaft-driven cranes require large tail rotors to provide the high reacting torques required to balance out the main rotor transmission torque. These larger tail rotors must be mounted high for ground clearance, and such mounting imposes high torsional loads in the fuselage. These high bending and torsional loads are not present in tip-driven rotor cranes. Second, the tip-driven helicopters used in this comparison have lower ratios of fuselage length to rotor radius than the shaft-driven cranes, since a large tail rotor is not required, and are therefore lighter.

These considerations resulted in the following equation:

$$W_b = 9.39 \, R \, W_g^{0.178} \, n^{0.089} + 150 \qquad (9)$$

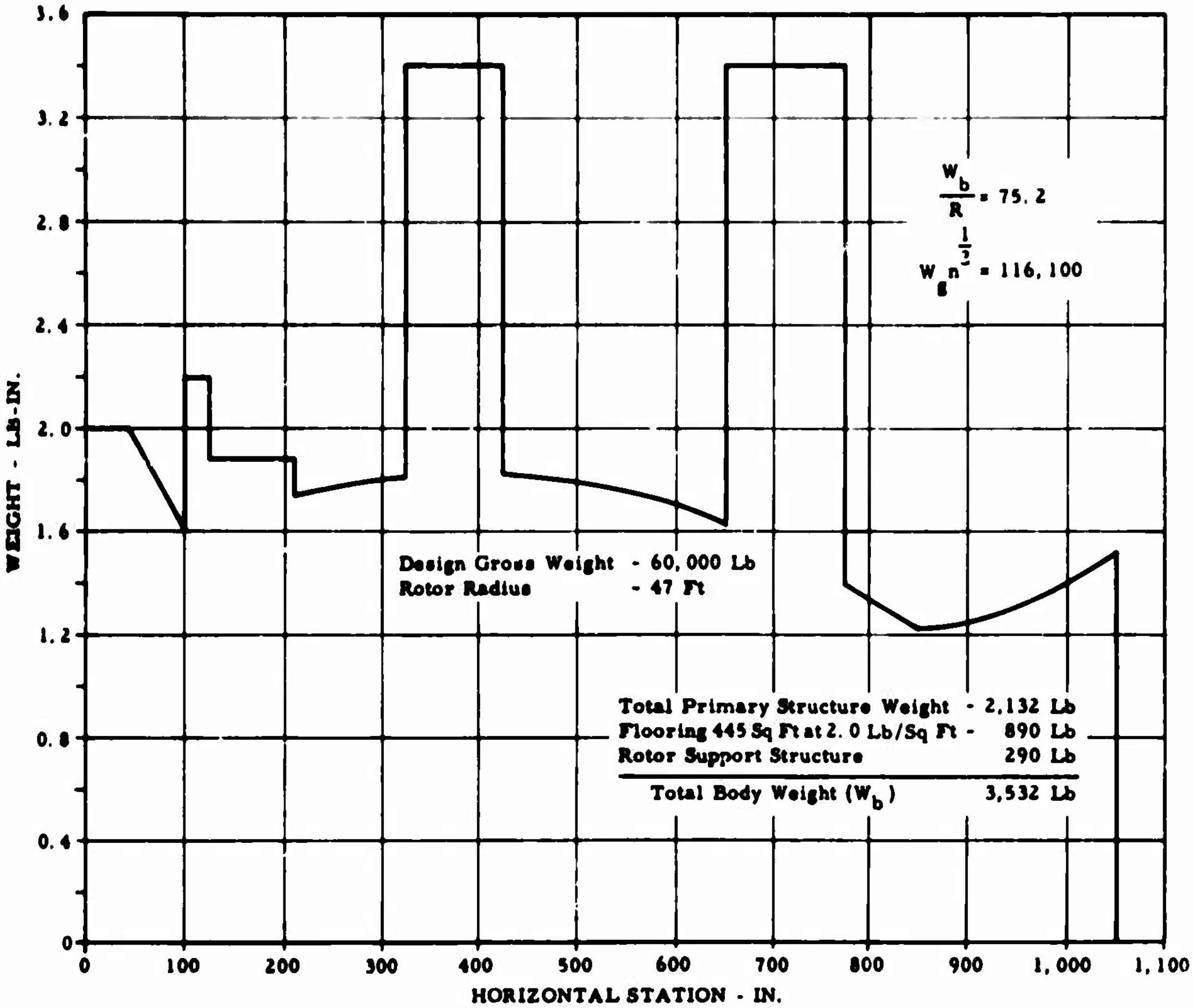


Figure 25. Primary Structure Weight Distribution - Configuration 3.

## ALIGHTING GEAR GROUP WEIGHT EQUATIONS

The conventional method of expressing alighting gear weight as a direct function of design gross weight was used in this report. Figure 26 establishes the validity of the fixed landing gear equations based on actual data, which are as follows:

Fixed long gear for configurations 1 and 4 $$W_{lg} = 0.046\ W_g \qquad (10)$$

Fixed short gear for configurations 2 and 3 $$W_{lg} = 0.035\ W_g \qquad (11)$$

The retractable landing gear equations assume a retraction system weight penalty of 0.010 $W_g$ and 0.013 $W_g$ for the short and long gears, respectively. The larger weight penalty for the long gear is based on the increased complexity of the retracting mechanism.

## FLIGHT CONTROLS EQUATION

The flight controls equation includes all cockpit controls, rotating and nonrotating rotor controls, and tail rotor and surface controls.

The equation used in the parametric sizing program reflects the preliminary weight estimates of the flight controls system, based on a 60,000-pound-gross-weight vehicle with a rotor radius of 47 feet. The equation assumes a direct relationship with gross weight and was based on design data available at the time that the computer program was being prepared. Since that time, these data have been reviewed, with some weight adjustments being made.

Summarized in the tabulation below is a comparison of this data as well as actual weights on the XV-9A helicopter.

| | XV-9A Actual Weight (lb) | | Parametric Study Controls Weight (lb) | | Preliminary Design Controls Weight (lb) | |
|---|---|---|---|---|---|---|
| Cockpit controls | | 29 | | 30 | | 30 |
| Intermediate linkages and controls - rotor and tail | | 92 | | 160 | | 160 |
| Hydraulic cylinders and mounts | | 89 | | 190 | | 340 |
| Rotor head controls | | 584 | | 620 | | 807 |
| Swashplate assembly | 104 | | 492 | | 679 | |
| Links, bellcranks, and supports | 480 | | 128 | | 128 | |
| Total flight controls | | 794 | | 1,000 | | 1,337 |

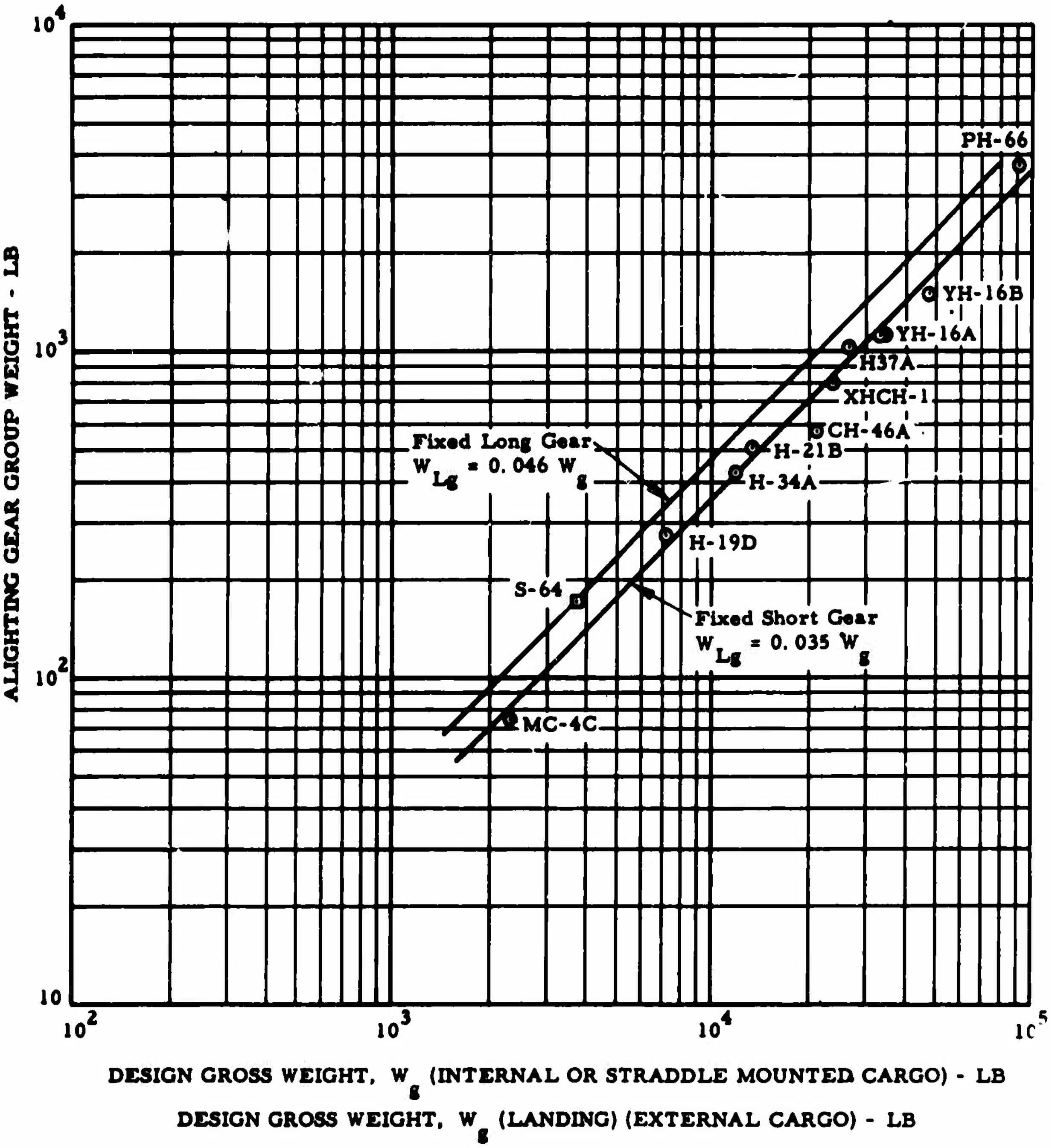


Figure 26. Fixed Alighting Gear Group Versus Design Gross Weight.

The parametric study flight control weight equation based on preliminary weight estimates was derived as follows:

$$W_{fc} = 1,000 \left( \frac{W_g}{60,000} \right) = 0.0167\ W_g \tag{12}$$

As a result of later weight data, the equation has been revised as follows for the preliminary design effort:

$$W_{fc} = 1,337 \left( \frac{W_g}{60,000} \right) = 0.022\ W_g \tag{13}$$

Figure 27 shows a plot of this equation as well as flight controls weights of articulated single-rotor shaft-driven helicopters.

## HYDRAULICS AND PNEUMATICS GROUP EQUATION

The hydraulics and pneumatics group weight equation was derived from weight data plotted versus design gross weight and shown on Figure 28. The equation of the best fit curve is:

$$W_h = 3.45 \left( \frac{W_g}{1,000} \right)^{1.28} \tag{14}$$

Examination of plotted data indicates that this equation adequately represents the weight trend of hydraulic systems with gross weight.

## ELECTRICAL GROUP WEIGHT EQUATION

As in the previous equation, the electrical group weights of numerous helicopters were plotted versus gross weight, as shown in Figure 29. The equation of the curve is expressed as follows:

$$W_{el} = 75 \left( \frac{W_g}{1,000} \right)^{0.55} \tag{15}$$

## FIXED-WEIGHT COMPONENTS

The following weights, common to all configurations, have been established from preliminary weight estimates and from comparisons with helicopters performing similar missions. These components are assumed to be of constant value for the range of gross weights under consideration.

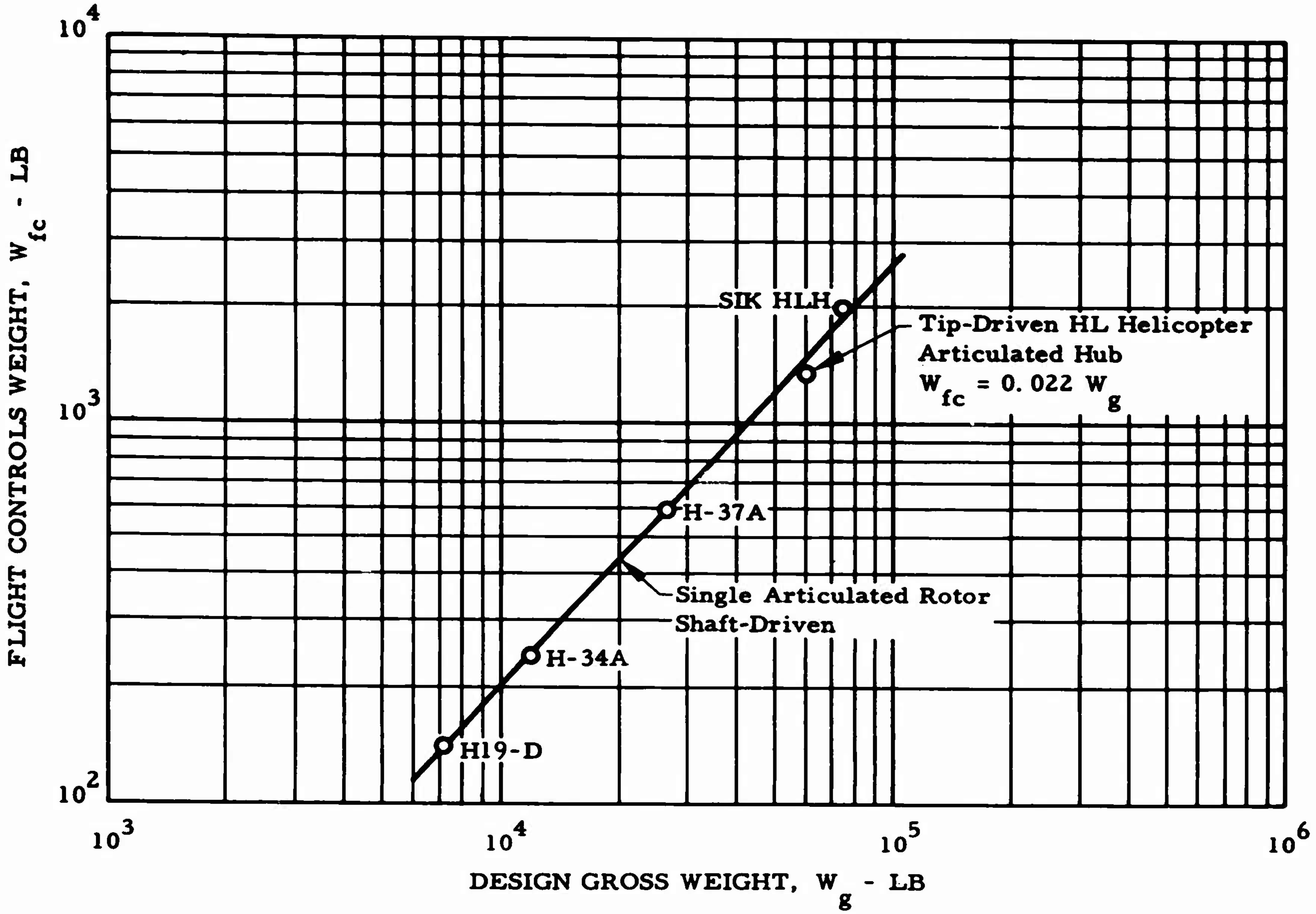


Figure 27. Flight Controls Versus Design Gross Weight.

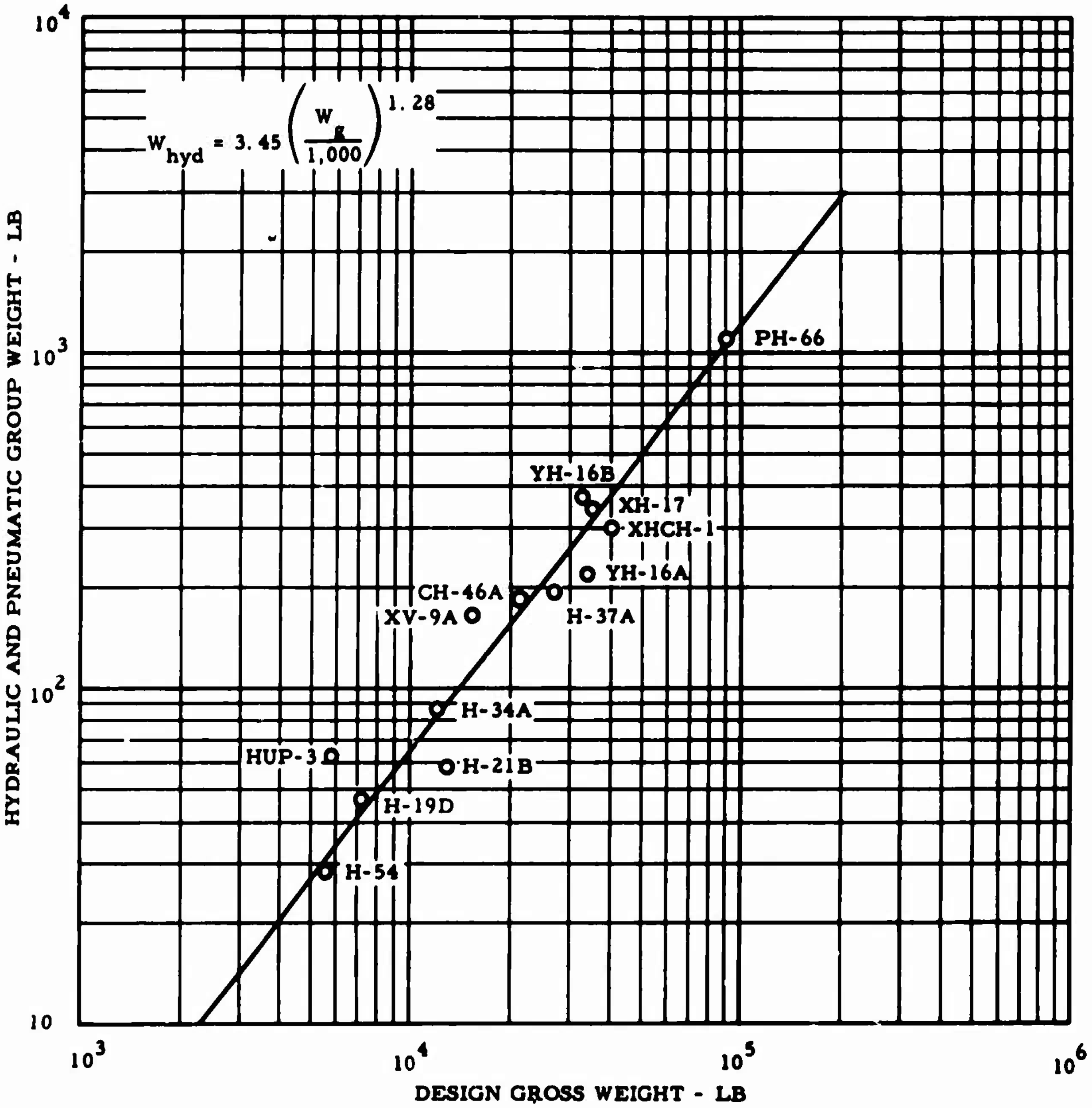


Figure 28. Hydraulic and Pneumatic Group Weight Versus Design Gross Weight.

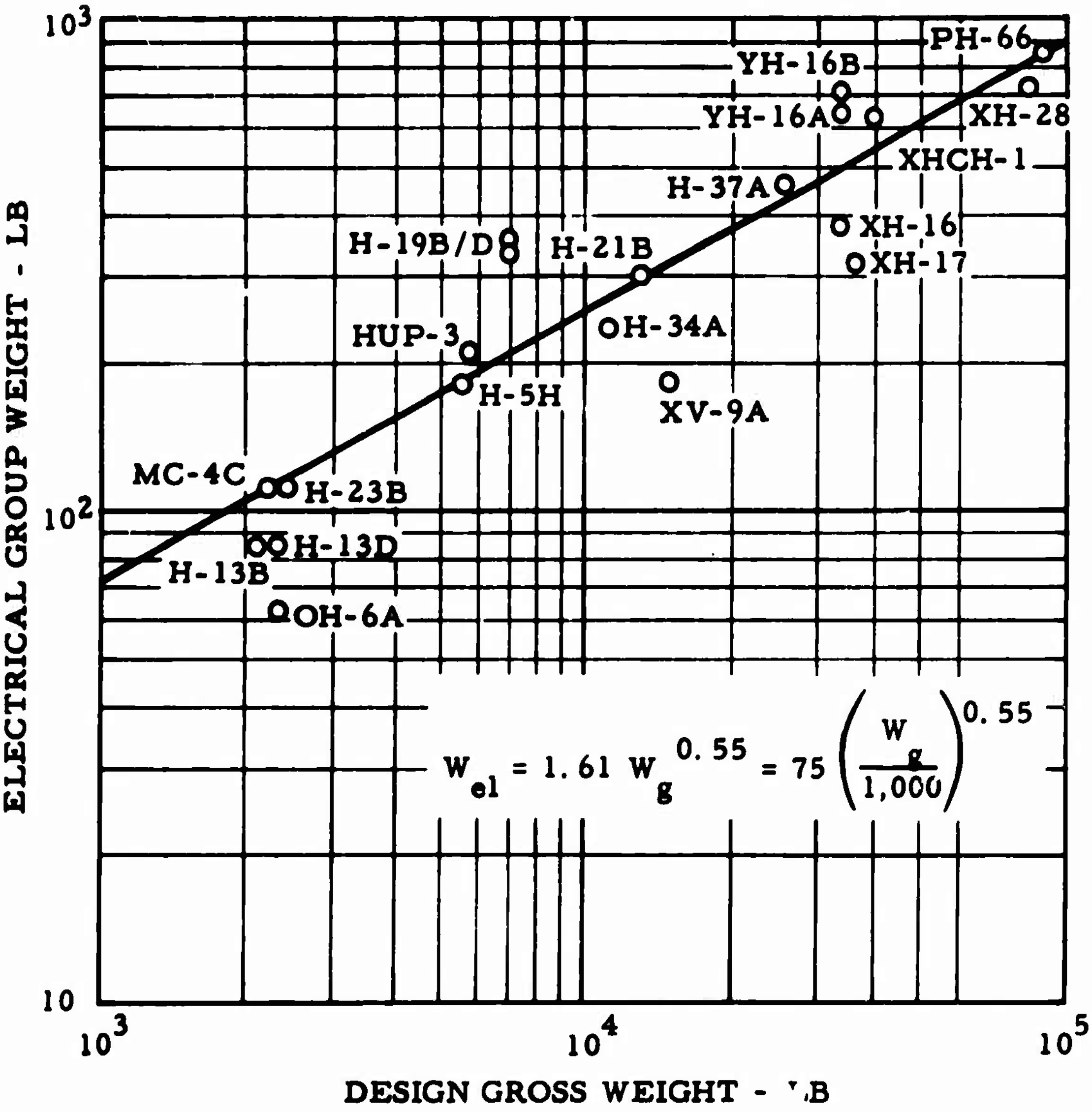


Figure 29. Electrical Group Weight Versus Gross Weight.

## PROPULSION GROUP WEIGHT

The propulsion group weight was established from preliminary weight estimates based on design layouts and from sizing of similar components used on the XV-9A helicopter. The weight estimates comprising the group weight of 3,466 pounds used in the parametric study have been revised for the preliminary design to reflect more current information.

A breakdown and comparison of these weights are given below. Included also are actual weights of the XV-9A propulsion group.

| | Actual XV-9A Weight (lb) | Parametric Study Weight (lb) | Preliminary Design Weight (lb) |
|---|---|---|---|
| Engine installation (includes engines, induction, exhaust, and fuel systems) | 1,537 | 1,627* | 1,627* |
| Accessory gearbox | 74 | 50 | 50 |
| Lubrication system | 64 | 64 | 64 |
| Starting system | 10 | 70 | 70 |
| Engine controls | 83 | 60 | 60 |
| Rotor drive system (includes diverter valves, ducting to rotor, joints, seals, and supports) | 420 | 825 | 480 |
| Engine section nacelles and supports | 683 | 610 | 460 |
| APU installation | | 160 | 160 |
| Total propulsion group | 2,865 | 3,466 | 2,971 |

Detailed calculations of the heavy-lift helicopter rotor drive system weights noted in the tabulation above are included in Appendix I.

A review of the propulsion group system using four T64/S4B engines has resulted in a weight change from 4,720 to 4,585 pounds.

## INSTRUMENTS AND NAVIGATIONAL EQUIPMENT

$$W_i = 180 \text{ pounds} \tag{16}$$

---

*Includes weight of GE1/J1 engines.

ELECTRONICS GROUP

$$W_{en} = 150 \text{ pounds} \tag{17}$$

FURNISHINGS AND EQUIPMENT GROUP

$$W_{fe} = 300 \text{ pounds} \tag{18}$$

AIR CONDITIONING AND ANTI-ICING

$$W_{ac} = 100 \text{ pounds} \tag{19}$$

CARGO-HANDLING EQUIPMENT

1 each 25-ton-capacity winch

$$W_c = 1,400 \text{ pounds} \tag{20}$$

Alternate: 4 each 6-ton-capacity winch

$$W_c = 1,700 \text{ pounds} \tag{21}$$

The above weights were based on a manufacturer's proposal for a 20-ton winch.

HELICOPTER EMPTY WEIGHT ($W_e$)

The empty weight, as defined in this study, is equal to the sum of the following groups:

| | |
|---|---|
| Main rotor group | $W_r$ |
| Tail group | $W_{tg}$ |
| Hover-yaw controls | $W_{hy}$ |
| Fuselage group | $W_b$ |
| Alighting gear group | $W_{lg}$ |
| Flight controls group | $W_{fc}$ |
| Hydraulics and pneumatics group | $W_h$ |
| Electrical group | $W_{el}$ |

| | |
|---|---|
| Propulsion group | $W_{pp}$ |
| Instruments and navigation equipment | $W_i$ |
| Electronics group | $W_{en}$ |
| Furnishings and equipment | $W_{fe}$ |
| Air-conditioning and anti-icing | $W_{ac}$ |
| Cargo-handling equipment | $W_c$ |

## USEFUL LOAD ITEMS

This group, as defined in this study, includes the following items.

| | |
|---|---|
| Crew (3-man) | 600 lb |
| Crew kits | 50 lb |
| Oil | 30 lb |
| Unusable fuel | 100 lb |
| Payload | As determined |
| Fuel | As required by study |
| External cargo pod (used on configuration 1, 12-ton maximum gross payload) | 4,300 lb |
| External fuel pod (optional for configurations 1 and 4; 50,000-lb capacity) | 2,500 lb |
| Internal fuel tank cells (optional for configurations 2 and 3; 50,000-lb capacity) | 2,500 lb |

## GROSS WEIGHT ($W_g$)

The gross weight of the mission being considered is equal to the empty weight ($W_e$) plus the applicable useful load items.

## WEIGHT EQUATIONS - COMPOUND HELICOPTER

The consideration of compound helicopter operation requires modification of some group weight constants and equations and the addition of new expressions reflecting this conversion. The equations listed below are applied as specified for flight configurations with or without wing installed.

## FIXED PROVISIONS

The following equations are used for the compound configuration, wing on or wing off:

### Tail Group

This equation replaces the pure helicopter equation for the tail group (3).

$$W_{tg} = 0.025\ W_g \tag{22}$$

### Surface Controls

Includes all controls required to operate tail surfaces plus wing surfaces controls inboard of the wing joint.

$$W_{scf} = 0.00833\ W_g \tag{23}$$

### Cruise Fan Duct System

$$W_{cff} = 230 \text{ lb} \tag{24}$$

### Hydraulics and Pneumatics System

This equation replaces the pure helicopter equation for the hydraulic and pneumatic systems (14).

$$W_h = 4.47 \left( \frac{W_g}{1,000} \right)^{1.28} \tag{25}$$

## REMOVABLE PROVISIONS

The following equations apply when the wing is installed:

### Wing Group

The equation shown is the reduced form of the wing weight equation developed by I. H. Driggs and is obtained by assuming the following constants:

$$\frac{C_t}{C_r} = 0.50$$

$$\frac{t_r}{C_r} = 0.21$$

$$\frac{t_t}{C_t} = 0.12$$

design stress factor, $f = 2.7 \times 10^3 \; W_g^{0.204}$

$$W_w = 0.43 \, n \, b \left(\frac{W_g}{10,000}\right)^{0.796} (4.95 + 0.465 \, AR) \qquad (26)$$

### Surface Controls

Includes all wing-mounted controls.

$$W_{scr} = 0.00333 \, W_g \qquad (27)$$

### Cruise Fan Installation

Includes wing-mounted cruise fan installation and removable ducting

$$W_{cfr} = 2,419 \text{ lb} \qquad (28)$$

## WEIGHT SAVING FEATURES OF THE HOT CYCLE ROTOR

The large difference between the weight of a heavy-lift helicopter with a tip-driven rotor and that of a heavy-lift helicopter with a shaft-driven rotor may be justified as follows:

1. Rotor Group
    a. The shaft-driven helicopter requires a larger rotor to support its higher gross weight, which is some 30 percent higher than the gross weight of any of the Hot Cycle configurations.
    b. The Hot Cycle utilizes strap retention instead of the heavier, more complex pitch housings and bearings.

c. The Hot Cycle does not have the high steady and cyclic torque loads to transmit through the rotor hub, since it is tip-driven

d. The Hot Cycle hub provides direct load paths that permit a simpler, lightweight structure.

e. Blade structure is optimized from root to tip on the Hot Cycle blade, where spar material is arranged to best satisfy the requirements for blade flight loads and ground flapping.

2. Body Group

a. The Hot Cycle configurations are smaller, with a substantially lower gross weight.

b. The Hot Cycle fuselages (configurations 2 and 3) are a more efficient structural shape.

c. The shaft-driven helicopter fuselage must support large gearboxes and associated high torque loads.

3. Landing Gear

a. The shaft-driven helicopter requires a heavier gear because of its higher gross weight.

b. The Hot Cycle landing gears are shorter because of the smaller fuselage and lower center of gravity.

4. Flight Controls

a. The Hot Cycle rotors are smaller, so the lower control loads result in a lighter system.

5. Propulsion Group

a. The very large and heavy transmission system for the shaft-driven helicopter is not required on the Hot Cycle configurations.

b. No main rotor shafting is required on the Hot Cycle helicopters -- only the lightweight shafting for the small yaw fan.

# STRUCTURES

## GENERAL PHILOSOPHY AND DESIGN OBJECTIVES

A discussion of the structural philosophy, features, materials, and design criteria is given in the following paragraphs. Design loads and the detail stress analysis may be found in Appendix II.

The general philosophy used in the design of the heavy-lift helicopter structure is the same as the philosophy used for the OH-6A, TH-55, Models 269 and 300, and the XV-9A Hot Cycle research aircraft. This design philosophy emphasizes simplicity, light weight consistent with desired strength and safety, fail-safe design, long service life, low maintenance, and conservative exploitation of the latest state of the art in materials, processes, and fabrication techniques.

## PRINCIPAL STRUCTURAL FEATURES

The following list summarizes the significant structural features of the Hot Cycle heavy-lift rotor:

1. Reliability gains and weight savings over shaft drives with their multiple dynamic elements -- due to the Hot Cycle ducted propulsion system; yaw fan required for maneuver only, since there is no main rotor drive shaft and resulting torque.

2. Fail-safe design features for improved level of safety.

3. Simplest possible functional and structural configuration with a minimum of discontinuities; direct load paths are provided.

4. Minimum structural weight consistent with safety and strength requirements and a conservative application of advanced design.

5. Isolation of hot and cold components.

6. Long service life -- all ducting designed to 0. 2 percent creep deformation for 3, 600-hour life.

7. No dynamic elements used in the rotor power transmission system; jet aircraft reliability of hot components.

## MATERIALS AND ALLOWABLE STRESSES

Materials chosen for the Hot Cycle rotor are fully proven materials with the highest strength-weight ratio for the temperature environment and the static and fatigue loadings to be encountered in this aircraft. Experience gained on the XV-9A has been used in the selection of the following materials.

### ALUMINUM ALLOY

Aluminum alloy has been selected as the material for the blade trailing edge fairings and all structural parts that are subject to less than a 200°F temperature environment. It will be used in any structure that is designed primarily by buckling stability, since in this application it is relatively lighter than steel or titanium. In all statically loaded and fatigue loaded structures in which the load is primarily tension, 2024 alloy is used rather than 7075 alloy, which not only has a higher static strength but also has a higher notch sensitivity. Adhesive bonding is used extensively in preference to rivet or bolt attachments to provide excellent fatigue life.

### STEEL

Carpenter 455 maraging stainless steel has been selected for the rotor blade spar material and for structural parts in a moderately elevated temperature environment requiring maximum static and fatigue strength properties. For fatigue applications, this steel is considered to be one of the best all-around materials tested to date, showing exceptionally consistent fatigue properties in both longitudinal and transverse grain direction for both smooth specimens and those with holes, and for sheet and bar.

This stainless maraging steel has good resistance to oxidation and pitting. It is highly resistant to stress corrosion cracking at high stress levels, being superior to the semiaustenitic precipitation hardened stainless steels.

The coefficient of linear expansion of Carpenter 455 is slightly higher than that of titanium alloys; therefore, it can be used in combination with titanium at moderately elevated temperatures without developing detrimental thermal stresses.

### TITANIUM ALLOY

Titanium alloy is used in sandwich construction for the blade skins and also is used for structural parts in slightly elevated temperature environment for applications that require high static- and fatigue-strength-to-density ratio.

Two titanium alloys are under consideration: Ti-6Al-6V-2Sn and B-120-VAC. Final selection awaits results of tests currently under way. Ti-6Al-6V-2Sn is similar in many respects to Ti-6Al-4V but has higher strength and greater depth hardenability. Considerably higher toughness with some sacrifice in static strength is attained by reducing the oxygen content. B-120 titanium is an all Beta alloy. It is supplied in the solution-treated condition. A desirable feature of this alloy is that after machining only aging is required to obtain the desired strength level. This alloy is superior to other titanium alloys in bending and cold-forming operations.

The problem of stress corrosion in titanium has also been considered, and it appears that there should be no problem in this application. Stress corrosion has been only a potential problem at temperatures of more than 450°F and a steady stress level of more than 45,000 psi. In the Hot Cycle heavy-lift helicopter application, the temperatures and stress levels are predicted to be well below these limits.

## RENÉ 41

René 41 is proposed as the material for the hot gas ducts. It has a superior strength-to-weight ratio for static strength as well as for 0.2-percent creep and rupture properties in the 1,400°F temperature environment. This material performed well as used for blade ducting on the XV-9A.

## INCONEL 718

Inconel 718 is proposed for fabrication of parts subjected to temperatures up to 1,200°F, which is higher than can be tolerated by titanium. It has superior static strength properties and elongation values up to 1,200°F. Long-time rupture and creep properties are also superior to those of René 41 up to 1,150°F. This material has good forming qualities with slow response to age hardening, which allows it to be welded in the annealed or aged condition without spontaneous hardening during heating and cooling. It has good corrosion resistance in a wide variety of environments. Inconel 718 proved to be an excellent material as used on the XV-9A for the large duct assemblies in the hub area.

### AM355

AM355 is a precipitation hardening stainless steel. This material has been selected for the flapping and lead-lag retention strap packs because of its high static strength, good elongation properties, and excellent fatigue strength. This material is very satisfactorily used for the same applications on the OH-6A and XV-9A.

## THERMAL STRESS

The aircraft employs hot gas jets at the rotor blade tips to provide the rotor driving torque. The typical cross section of the blade ducts is formed from the intersection of two circles forming a figure-8. This results in a lightweight system, as all the gas pressure loads are carried by hoop tension. Any additional weight that results from local stiffening is held to a minimum and occurs only in the transition areas where there is a departure from a circular cross section. Thermal considerations are solved in a straightforward, simple manner and do not require complicated or sophisticated systems.

The hot gas system is isolated from the aircraft structure to allow it to grow with temperature without loading the cold structure. The longitudinal elongation due to temperature is taken up by bellows that divide the ducting into appropriate lengths.

These bellows allow the ducting to expand with temperature without introducing high restraining forces in the ducting system. A system of links that fully support each length of duct allows unrestrained growth due to temperature both longitudinally and diametrically. A single thickness of metal forms the duct wall. This eliminates thermal gradients that would exist in a built-up wall having a hot inner wall and a cold outer wall.

The ducts are insulated, thereby reducing the thermal gradients in the primary structure. Isolation of the hot duct from the structure aids in lowering the temperature differential within the cold structure, since less heat is transferred by conduction. The cooling air flowing through the hub and into the blade alleviates the buildup of heat in these structures. The highest structural temperature gradient in the blade is estimated as 260°F, based on XV-9A experience.

## STRUCTURAL DESIGN CRITERIA

All limit loads derived from the following criteria shall be multiplied by 1. 5 to obtain ultimate loads. The requirements of MIL-S-8698 (ASG) have been used as a guide.

## LIMIT LOAD FACTORS

| Mission | Load Factor |
|---|---|
| Design gross weight | +2.5, -0.5 |
| Ferry | +2.00 |

## COMPONENT DESIGN CRITERIA

All components shall be designed for at least 1,200 hours between major overhauls and 3,600-hour service life.

## LIMIT DIVING SPEED

$1.11\ V_{ne} = 1.11 \times 135\ \text{kn} = 150\ \text{kn}$

## MAIN ROTOR

The basic rotor data used in the structural design criteria have been previously shown in Table VIII of the Rotor System section of this report.

## ROTOR TIP SPEED, DESIGN OPERATIONAL, POWER-ON OR POWER-OFF

The design operational rotor speed shall be consistent with a tip speed of 750 ft/sec in hover and 675 ft/sec in cruise.

## ROTOR TIP SPEED, MAXIMUM POWER-ON (RED LINE)

750 ft/sec

## ROTOR TIP SPEED, MINIMUM POWER-OFF (RED LINE)

590 ft/sec

## ROTOR TIP SPEED, DESIGN MINIMUM POWER-OFF

560 ft/sec

## ROTOR TIP SPEED, MAXIMUM POWER-OFF (RED LINE)

750 ft/sec

## ROTOR TIP SPEED, LIMIT, POWER-OFF OR POWER-ON (1 g)

786 ft/sec

## BLADE DESIGN REQUIREMENTS

Ground flapping - 2.5 g
Ground wind - 40 kn at $C_L$ = 1.0
Infinite life at weighted fatigue condition (1.2 x loads in maximum cruise level flight)

## HUB DESIGN REQUIREMENTS

Weighted fatigue flapping - ±5°
Weighted fatigue lead-lag - ±1.25°
Maximum lead-lag - ±3°
Weighted fatigue feathering - ±12°
Maximum flapping - +25°, -6°

## GAS TEMPERATURE CONSIDERATIONS

Duct wall and skin temperatures with and without insulation are to be based on the following tabulation:

| | Basic Structure No Insulation | Basic Structure 1/8-in. Refrasil Insulation |
|---|---|---|
| Heat flux - BTU/hr-sq ft | 9,500 | 3,000 |
| Duct wall temperature | 1,310°F | 1,390°F |
| Inner skin and inner rib flange temperature | 870°F | 400°F |
| Outer skin temperature | 225°F | 140°F |

(Reference - gas temperature = 1,425°F, ambient temperature = 70°F)

## GAS TEMPERATURE SPECTRUM

| Power Setting | Temperature (deg F) | Pressure (psig) | Remarks |
|---|---|---|---|
| Emergency | 1,443 | 39.6 | 10 seconds<br>12 times |
| Takeoff | 1,395 | 37.9 | 5 minutes |
| Military | 1,300 | 33.8 | 30 minutes |
| Maximum continuous power | 1,225 | 30.34 | |

## LIMIT STRUCTURAL DESIGN

| | |
|---|---|
| Maximum temperature<br>(occurs at Mach 0.2 SL 103°F) | 1,494°F (emergency)<br>1,415°F (takeoff) |
| Pressure at maximum temperature<br>(Mach 0.2 SL 103°F) | 37.7 psig (emergency)<br>36.3 psig (takeoff) |
| For 3,600 hours operational use | 2 minutes at 1,494°F<br>20 hours at 1,415°F<br>40 hours at 1,395°F<br>100 hours at 1,300°F |

See Figure 30 for remainder of spectrum.

| | 1,415 | 1,395 | 1,300 |
|---|---|---|---|
| Operational Use - 3,600 Hrs* | | | |
| Transport - 3,240 Hrs | 16 | - | 16 |
| Heavy-Lift - 360 Hrs | 4 | 40 | 84 |

*Includes 2 min at 1494°F

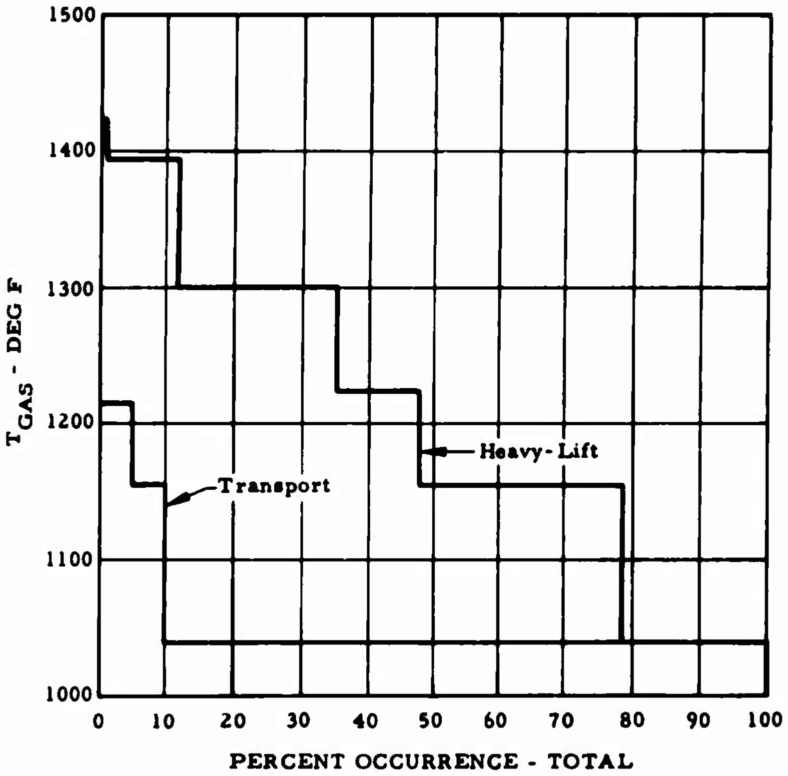


Figure 30. Time-Temperature Spectrum - GE1 Engine - Heavy-Lift Helicopter.

# PARAMETRIC STUDY

The parametric study was completed, summarized, and submitted in Reference 2. The following data and discussion of the parametric study are largely direct quotations of the material from that report. The areas covering the compound helicopter and fuel utilization have been revised to reflect the results of additional investigations.

The objective of the parametric and configuration study was to determine the optimum Hot Cycle rotor system for a 12- to 20-ton-payload heavy-lift helicopter, and to investigate, on a limited basis, the features required to increase its cruise speed by a substantial amount. The study indicates that a vehicle with a rotor as small as 80 feet in diameter and with an empty weight of approximately 18,000 pounds will exceed all mission requirements of range and payload by 20 to 30 percent, even though a larger-diameter rotor has been selected as optimum for other considerations.

Five aircraft configurations have been considered in the parametric study. Included are four helicopters and one compound helicopter, which has been investigated as a means of substantially increasing cruise speed. To accomplish the study, statistical and analytical data were integrated into a computer program, and the results were then cross-plotted to arrive at the optimum rotor.

The parametric variables used included four hub configurations; variations in number of blades and their thickness, chord, radius, and spar location; three duct configurations; tip speed; and fixed versus retractable landing gear.

The primary powerplant configuration studied consisted of two GE1/J1 engines; however, an alternate configuration using four T-64/S4B engines was also investigated in the parametric study.

## PARAMETRIC STUDY CONCLUSIONS

The following conclusions were reached from the results of the parametric study:

1. The optimum rotors for each of the various configurations, based on the results of the parametric study and on practical considerations, and their performances are shown in Table XI.

# TABLE XI
# SUMMARY - OPTIMUM ROTOR SIZE FOR CONFIGURATIONS STUDIED AND PERFORMANCE

| Parameter | Configuration | | | | |
|---|---|---|---|---|---|
| | 1 | 2 | 3 | 4 | 5** |
| Number of blades | 3 | 3 | 3 | 3 | 3 |
| Blade radius (ft) | 45 | 40 | 45 | 45 | 45 |
| Blade chord (in.) | 55 | 60 | 60 | 55 | 60 |
| Blade thickness (inboard 0.75R) (%c) | 18 | 18 | 18 | 18 | 18 |
| Blade thickness (outboard 0.25R) (%c) | 14 | 14 | 14 | 14 | 14 |
| Blade spar location (x/c) | 0.30 | 0.25 | 0.25 | 0.30 | 0.25 |
| Blade tip speed - hover (fps) | 750 | 750 | 725 | 750 | 725 |
| Blade tip speed - cruise (fps) | 675 | 675 | 675 | 675 | 675 |
| Blade duct configuration | Fig-8 | Fig-8 | Fig-8 | Fig-8 | Fig-8 |
| Hub configuration | * | * | * | * | * |
| Landing gear | Fixed | Fixed | Fixed | Fixed | Retractable |
| Ferry mission (nmi) | 1,816 | 2,065 | 2,034 | 1,816 | 2,416 |
| Payload (tons) | | | | | |
| Transport mission | 12.03 | 13.98 | 14.08 | 13.72 | (See |
| Heavy-lift mission | 25.66 | 25.40 | 25.55 | 25.65 | Fig 48) |
| Weight empty (lb) | | | | | |
| Transport mission | 25,898 | 17,832 | 20,887 | 21,598 | 28,011 |
| Heavy-lift mission | 21,598 | 17,832 | 20,887 | 21,598 | 28,011 |
| Gross weight (lb) | | | | | |
| Transport mission | 57,939 | 54,371 | 56,788 | 57,939 | 54,971 |
| Heavy-lift mission | 78,428 | 73,976 | 77,368 | 78,428 | 74,893 |
| Payload/empty weight ratio | | | | | |
| Transport mission | 0.9290 | 1.5680 | 1.345 | 1.2708 | - |
| Heavy-lift mission | 2.3758 | 2.8489 | 2.45 | 2.3758 | - |
| Disc loading (lb/sq ft) | | | | | |
| 12-ton payload | 9.10 | 10.42 | 8.25 | 8.58 | - |
| 20-ton payload | 10.66 | 12.60 | 10.39 | 10.66 | - |
| Computer run number | 1-31 | 2-15 | 3-12 | 4-4 | - |

*Articulated with external shaft.

**Identical with configuration 3, except that it has been converted to a compound by the addition of wings, fans, ducting, and other required revisions.

2. The rotor selected for the preliminary design for disc loading considerations and as most nearly optimum for all configurations has a 45-foot radius, a 60-inch chord, an 18-percent blade thickness from root to 0. 75R and 14-percent from 0. 75R to tip, an articulated hub, a 750-fps hover tip speed, and a 675-fps cruise tip speed.

3. The results of the fuel consumption study indicate that a substantial improvement in fuel economy can be expected using the Hot Cycle propulsion system. The numbers of ton-miles per pound of fuel show improvements in the order of 150 to 300 percent as compared with present conventional helicopters.

4. Configuration 1 performance would be improved if a pod of smaller cross section were used. The pod cross section on the ship studied was arbitrarily made the same as that of the C-130. By changing the cross section to one more nearly approximating configuration 3, the range and transport payload would be increased.

5. The weight of the pod is not offset by the saving in fuel from the resulting lower drag; therefore, a detrimental effect on the mission performance results, as shown in a comparison of configuration 1, which has a pod, and configuration 4, which is an identical ship carrying all payloads externally (see Table XI).

6. The optimum rotor size of configurations 2 and 3 could be substantially smaller, if either:
   a. Single-engine failure only was considered in the autorotational requirements, or
   b. For configuration 2, internal payload was limited to a lower value than the 7 tons assumed for the autorotational capability check.

7. The Hot Cycle heavy-lift rotor is readily adaptable to any configuration found desirable from an operational standpoint and will produce comparable results in performance and light weight.

8. It is concluded that the compound helicopter will provide a substantial increase in cruise speed and ferry range. It is further concluded that the additional complexity of the compound is confined primarily to the wing and ducted thrust-fan installations, and that the required implementation is within the state of the art.

## AIRCRAFT CONFIGURATIONS STUDIED

Five aircraft configurations were studied and are described in the following paragraphs. The propulsion for all configurations is provided by two General Electric GE1/J1 gas generators. An alternate installation using four T-64 engines was also considered. Conversion of the basic helicopter propulsion system to the compound helicopter propulsion system has been accomplished by the addition of wings and ducted fans for thrust.

### CONFIGURATION 1 (Figure 31)

This is a crane-pod ship. Its overall size is determined primarily by the cargo compartment, which is 10 feet wide, 9 feet high, and 27 feet long, for a total of approximately 2,400 cubic feet. The cargo compartment cross section dimensions were chosen to be the same as those of a C-130 airplane cargo compartment to permit direct reloading between the vehicles. At 10 pounds per cubic foot, this compartment permits a loading of 12 tons and has cargo floor space for four of the standard 88-by-108-inch pallets. For the parametric study, it is assumed that the 12-ton transport mission payload is carried internally and that the 20-ton heavy-lift payload is carried externally. The engines are shoulder-mounted.

### CONFIGURATION 2 (Figure 32)

This aircraft utilizes a streamline body sized to carry the fuel for the ferry mission internally. The cargo compartment is approximately 6-1/2 feet wide (5 feet wide at the floor line), 7 feet high, and 45 feet long. It can carry approximately 7 tons internally at 10 pounds per cubic foot, with a cargo floor area that will accommodate six standard 54-by-88-inch pallets. For the parametric study, it is assumed that both the total transport and heavy-lift payloads are carried externally. The engines on this configuration are top-mounted to minimize frontal area.

### CONFIGURATION 3 (Figure 33)

This aircraft utilizes a streamline body sized to permit loading of the transport mission payload of 12 tons (at 10 pounds per cubic foot) internally, with a cargo floor area for six standard 88-by-108-inch pallets. The compartment is approximately 8 feet wide by 7 feet high by 46 feet long. The total heavy-lift mission payload is carried externally. The engines on this configuration are shoulder-mounted to provide for accessibility and for retraction of the landing gear into the nacelle fairing.

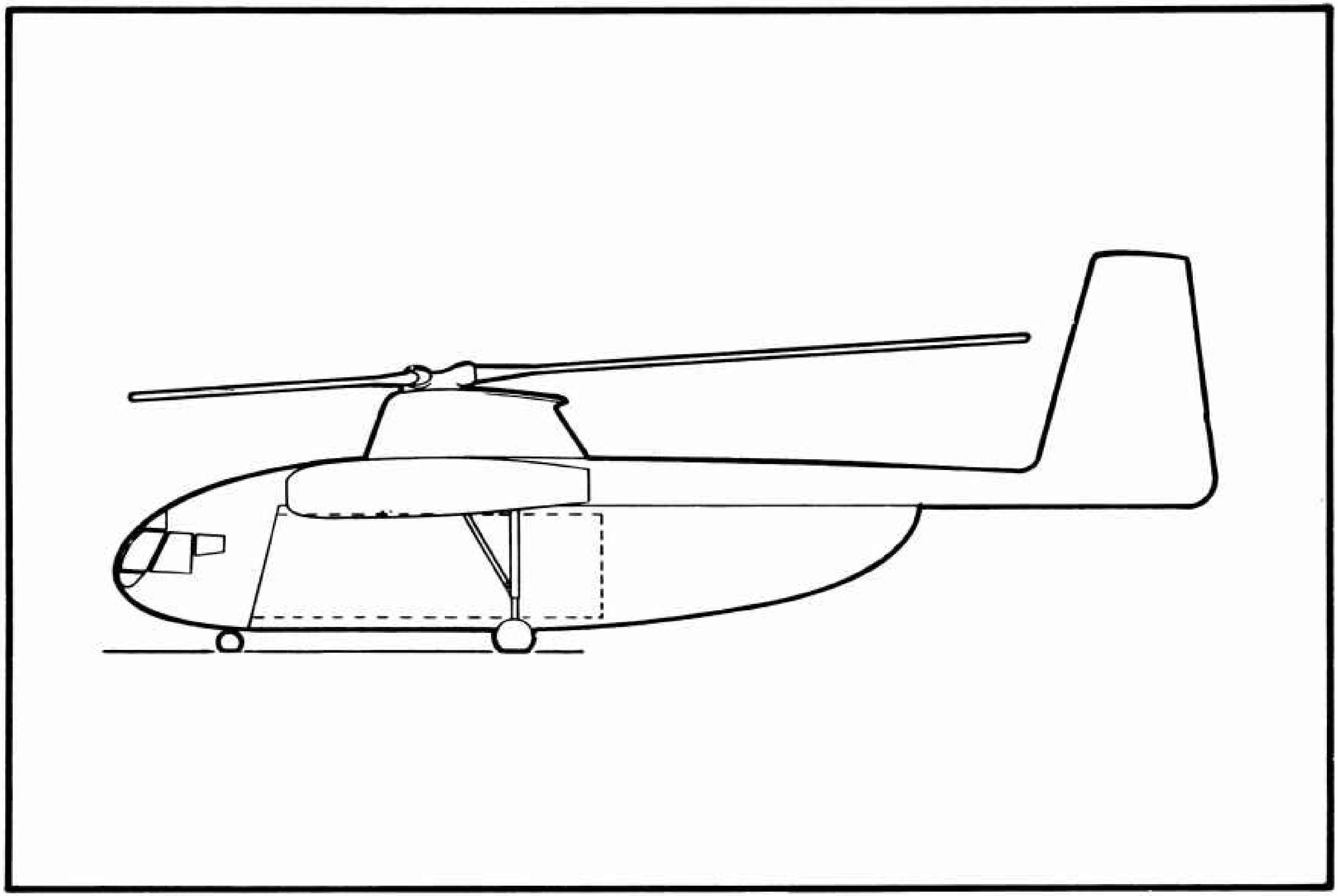


Figure 31. Configuration 1 Helicopter.

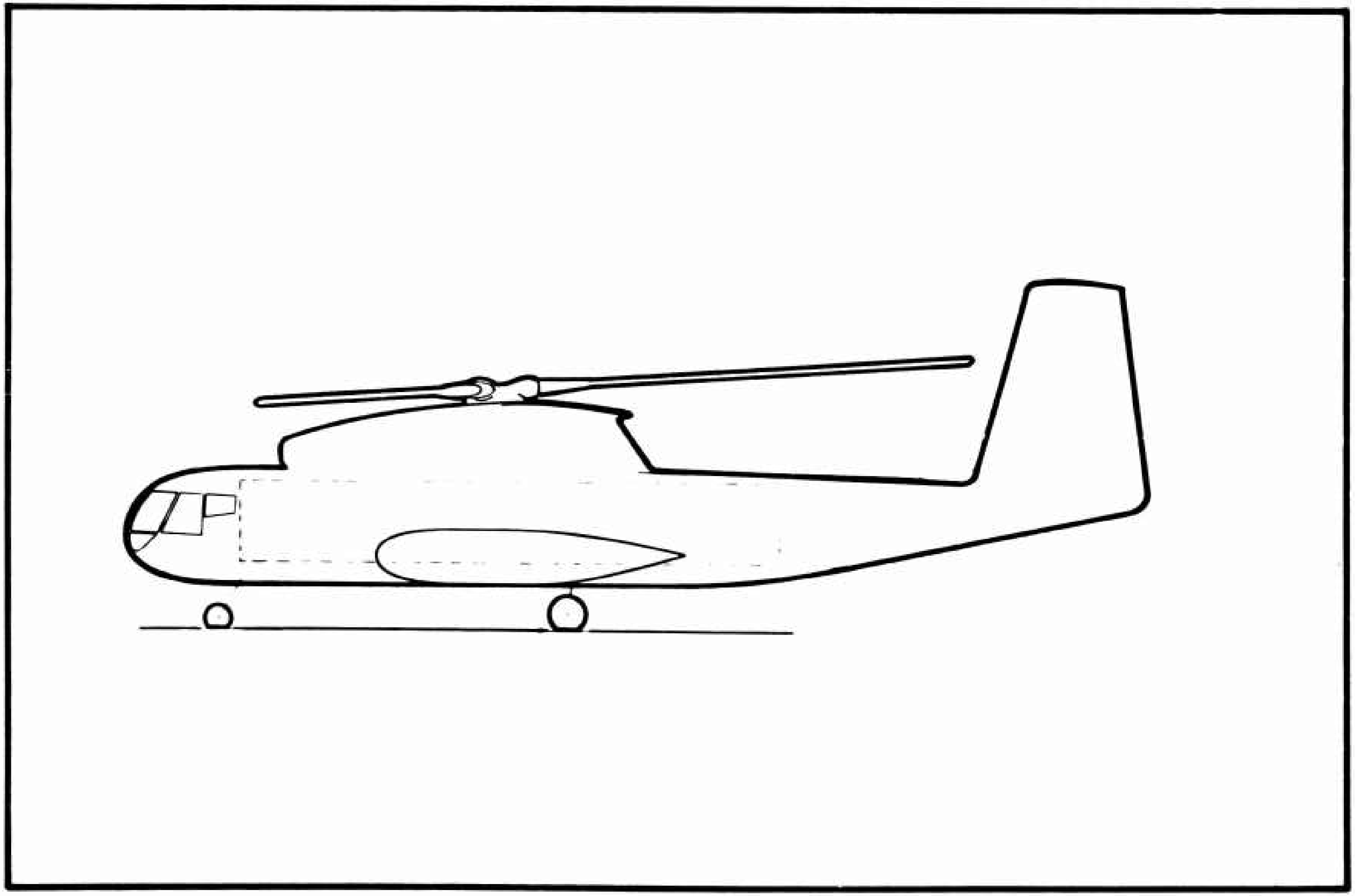


Figure 32. Configuration 2 Helicopter.

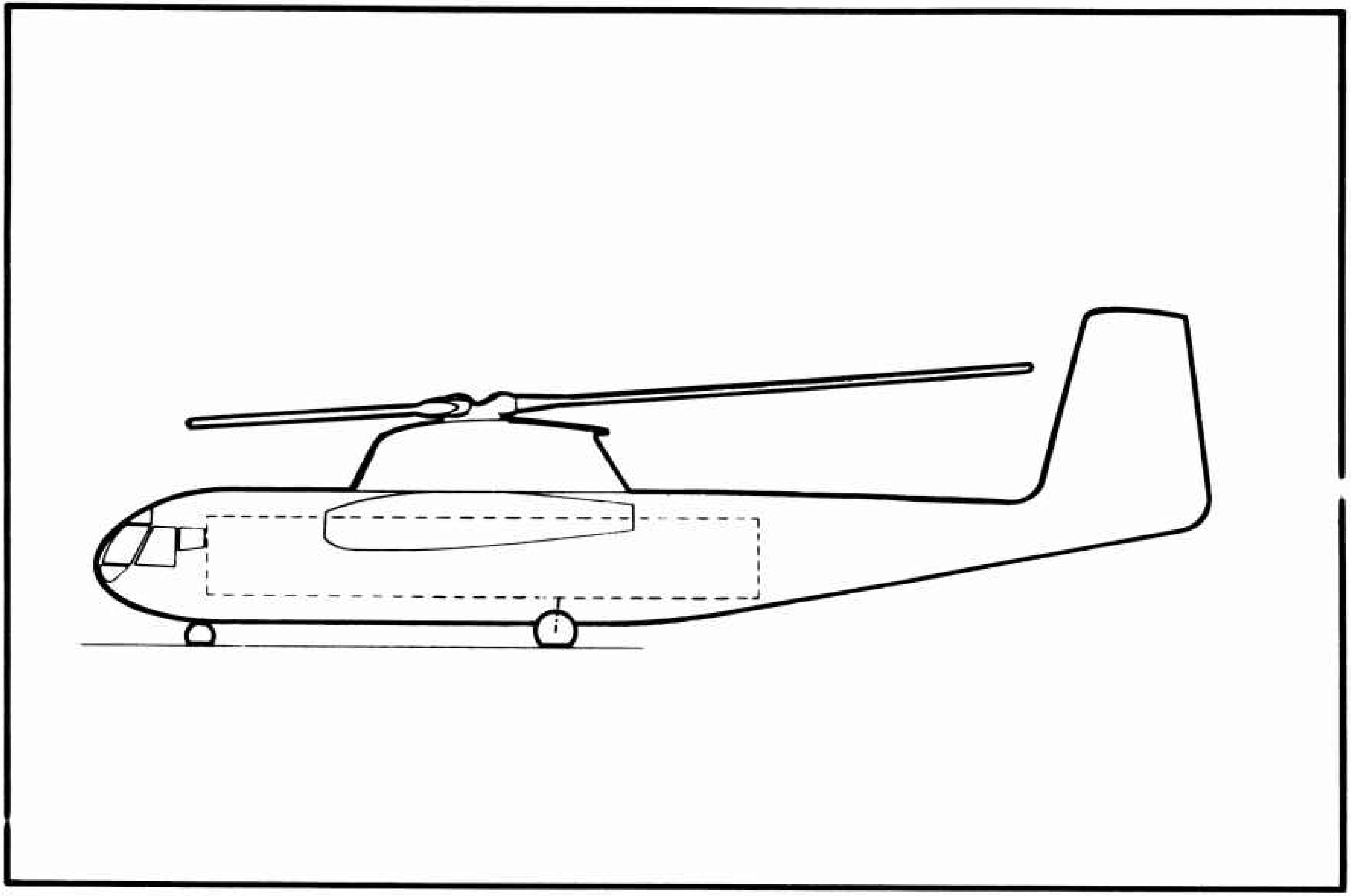


Figure 33. Configuration 3 Helicopter.

### CONFIGURATION 4 (Figure 34)

This configuration is identical with configuration 1 except that it assumes that both the transport and heavy-lift mission payloads are carried externally (no pod).

### CONFIGURATION 5 (Figure 35)

This configuration utilizes the same rotor and fuselage as configuration 3 except that wings and fans have been added for operation as a compound helicopter.

## HUB CONFIGURATIONS STUDIED

Four combinations of hub shaft and mast were studied and are described in the following paragraphs. All configurations use blade retention straps to provide flapping and feathering freedom. Weights of these configurations are shown in Table XII.

### TILTING HUB WITH INTERNAL CONTROLS (Figure 36)

This configuration, similar to that used on the XV-9A Hot Cycle research aircraft, employs two retention straps per blade, displaced chordwise to provide chordwise rigidity. The retention straps connect the blades to the hub assembly, which in turn is gimbally attached to the rotor shaft. In this hub configuration, the shaft passes through the center of the gas duct, and the flight control push-pull tubes pass from the swashplate, mounted below the hub, up through the center of the shaft to walking beams that transmit the motion to the blade pitch links. Hub restraint for improved controllability is provided in the form of air springs. Because of the necessity of gimbally mounting the tilting hub and the routing of gas ducts between the retention straps, the required hub fairing envelope is larger than that for the articulated hub.

### TILTING HUB WITH EXTERNAL CONTROLS (Figure 37)

The tilting hub with external controls is very similar to the tilting hub with internal controls except that the rotor controls,shaft and mast are installed outside the ducts. The swashplate is guided on the outside of the shaft. A heavy "spider" structure is necessary to transmit rotor loads from the gimbal to the external shaft, which results in a heavier hub with a slightly smaller fairing envelope than that of the tilting hub with internal shaft.

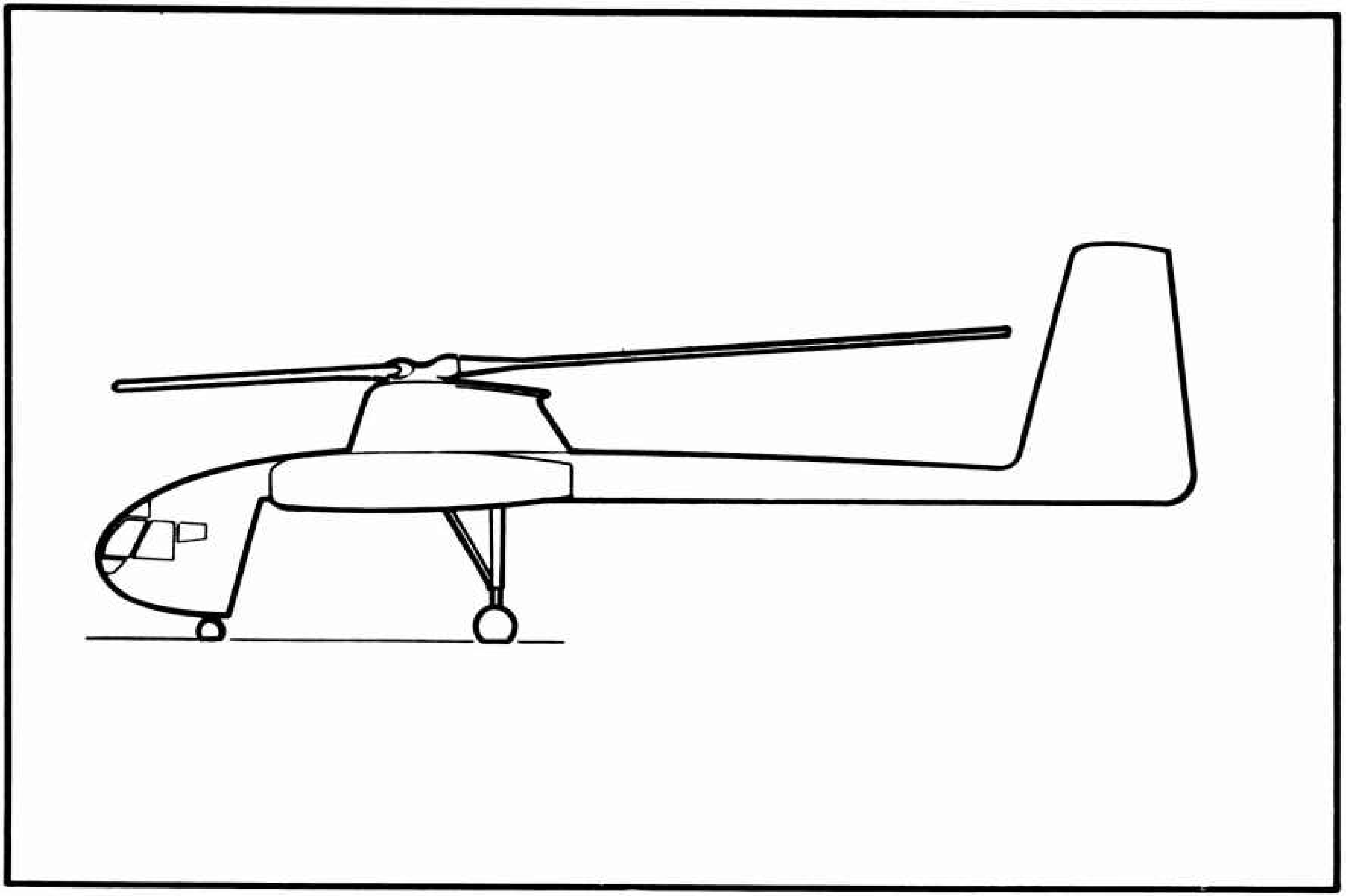


Figure 34, Configuration 4 Helicopter.

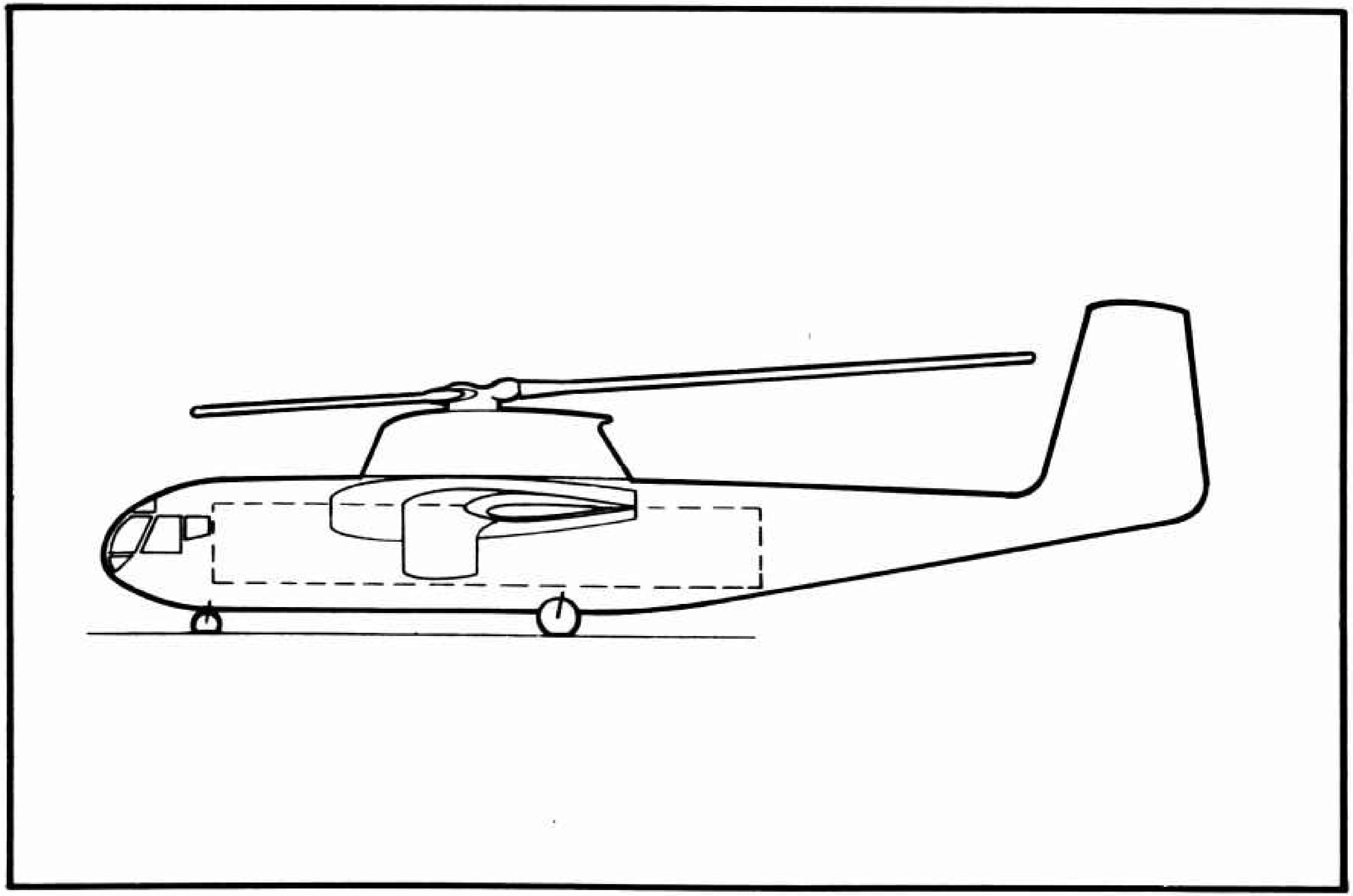


Figure 35. Configuration 5 Compound Helicopter.

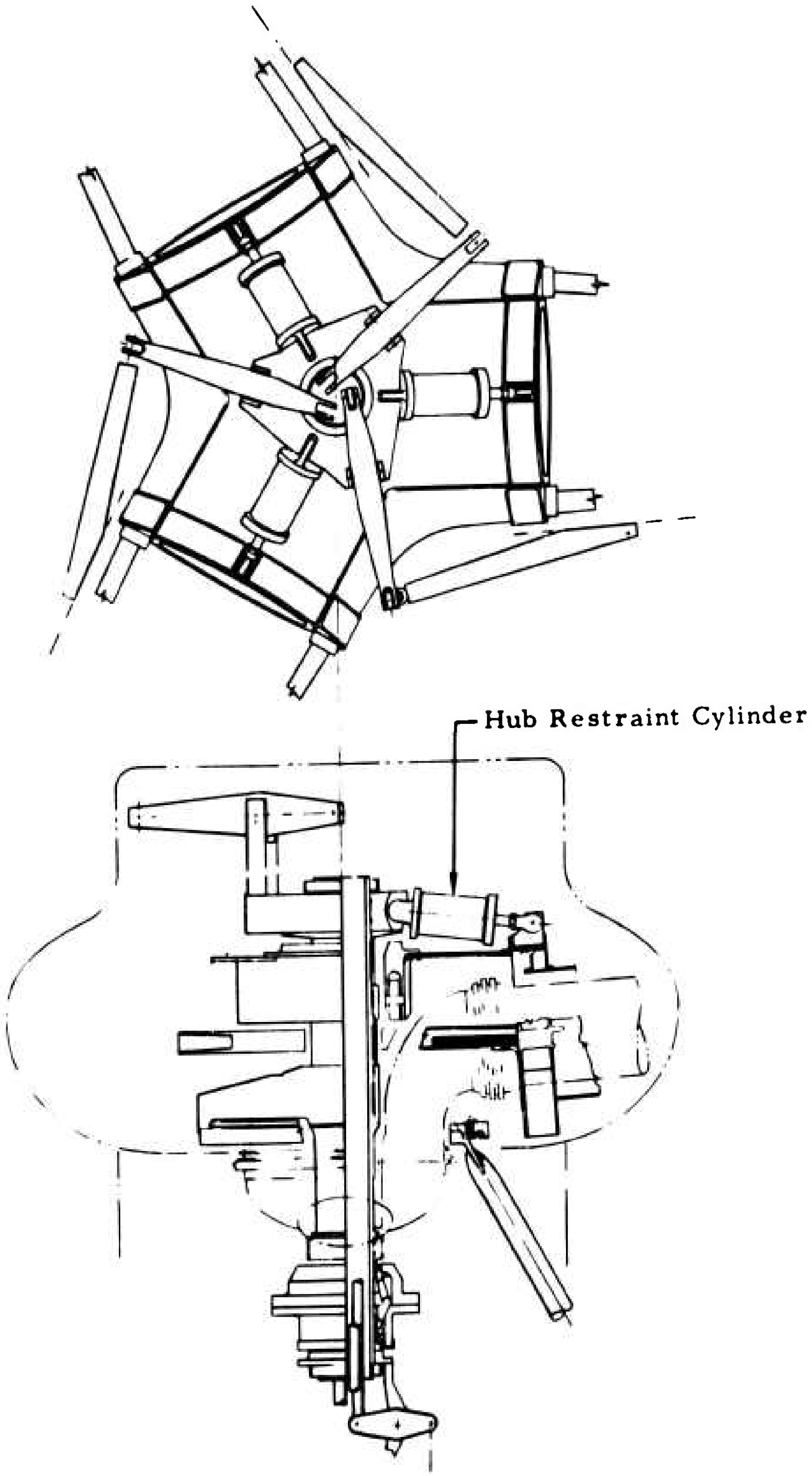


Figure 36. Tilting Hub With Internal Controls.

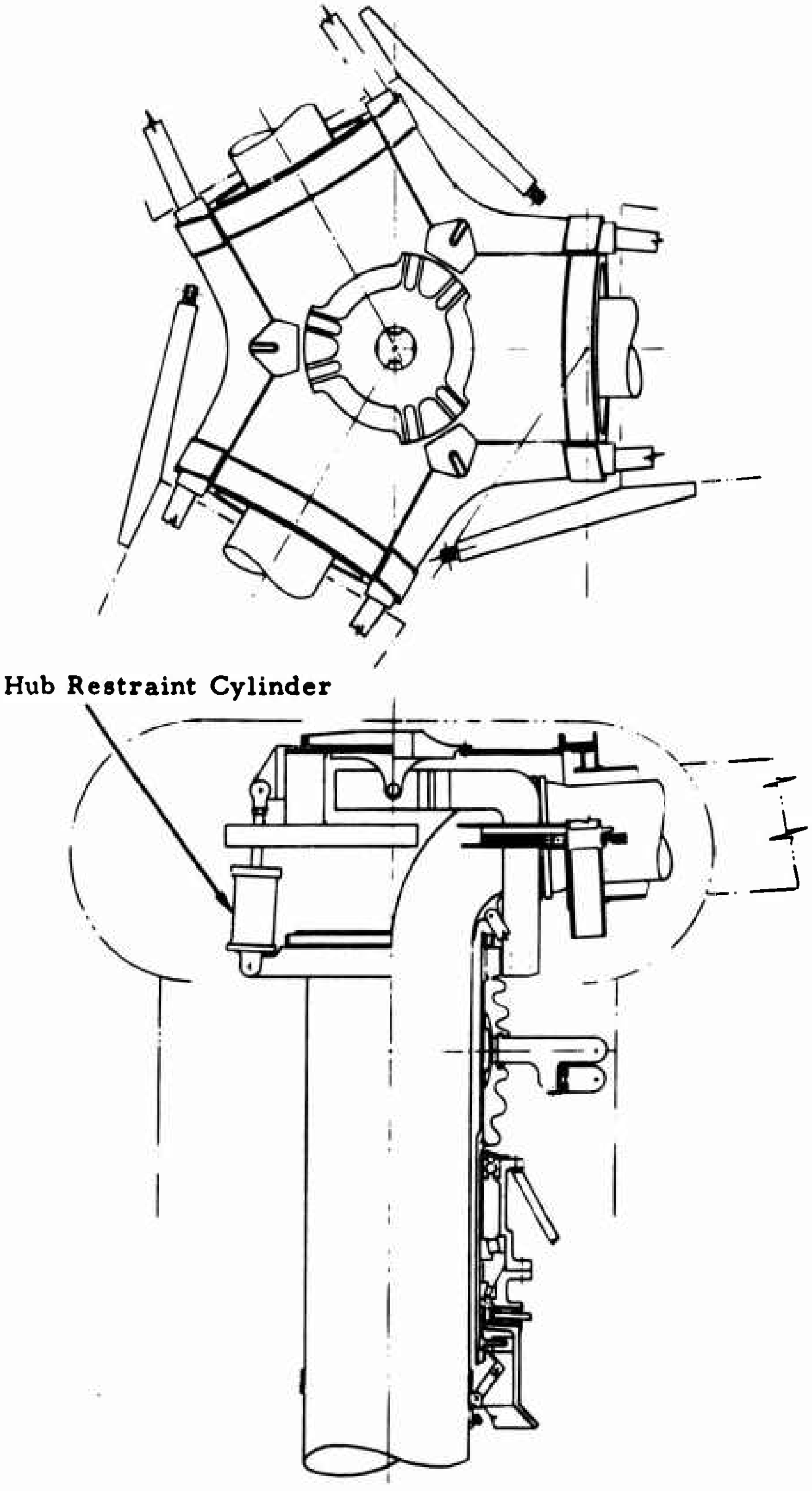


Figure 37. Tilting Hub With External Controls.

## TABLE XII
## WEIGHT OF VARIOUS HUB CONFIGURATIONS

| | | Tilting Hub | | Articulated Hub | |
|---|---|---|---|---|---|
| | | Internal Controls | External Controls | External Controls | Internal Controls |
| | XV-9A* | 395-0930** | 395-0931** | 395-0932** | 395-0933* |
| HUB ASSEMBLY | | | | | |
| Structure | 502 | 1,296 | 1,296 | 308 | 160 |
| Gimbal | 125 | 436 | 1,042 | - | - |
| Bearings, housings, and supports | 148 | 361 | 1,044 | 630 | 480 |
| Hardware | 10 | - | - | - | - |
| Hub restraint | - | 89 | 122 | - | - |
| Total | (785) | (2,182) | (3,504) | (938) | (640) |
| MAST ASSEMBLY | | | | | |
| Mast | 82 | 400 | 460 | 280 | 540 |
| Spoke | 33 | 183 | - | - | 244 |
| Spacers and retainers | 35 | 181 | 234 | 230 | 175 |
| Total | (150) | (764) | (694) | (510) | (959) |
| FAIRINGS | | | | | |
| Hub fairing | 8 | 45 | 40 | 30 | 40 |
| Total | (8) | (45) | (40) | (30) | (40) |
| TOTAL COMPONENT WEIGHT | | | | | |
| (Excluding blades retention, etc) | 964 | 3,016 | 4,263 | 1,503 | 1,664 |

*Based on actual weights of XV-9A components.

**All weights were estimated from drawings 395-0930 through 395-0933; weights were then used as input to the computer program (47-ft rotor radius).

## ARTICULATED HUB WITH INTERNAL CONTROLS (Figure 38)

This hub configuration also uses retention straps, as previously noted; however, on the articulated hubs the straps converge as they travel outboard to the lead-lag hinge point, resulting in lower control loads. To permit this convergence, the gas ducts are routed outside the blade retention straps. Flight control push-pull tubes pass from the swashplate, mounted below the hub, up through the center of the shaft to the walking beams that transmit the motion to the pitch arms. Since load paths are

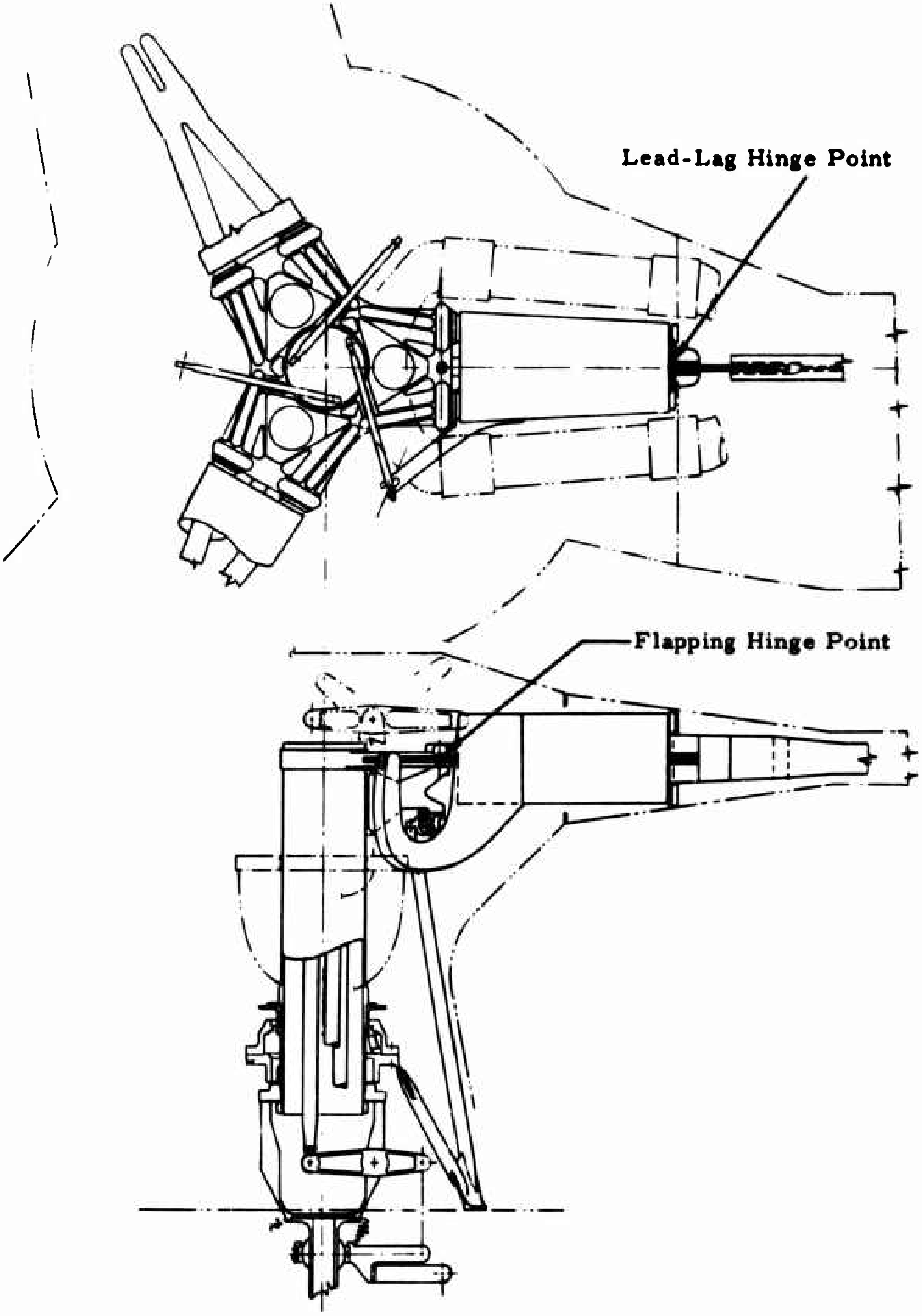


**Figure 38. Articulated Hub With Internal Controls.**

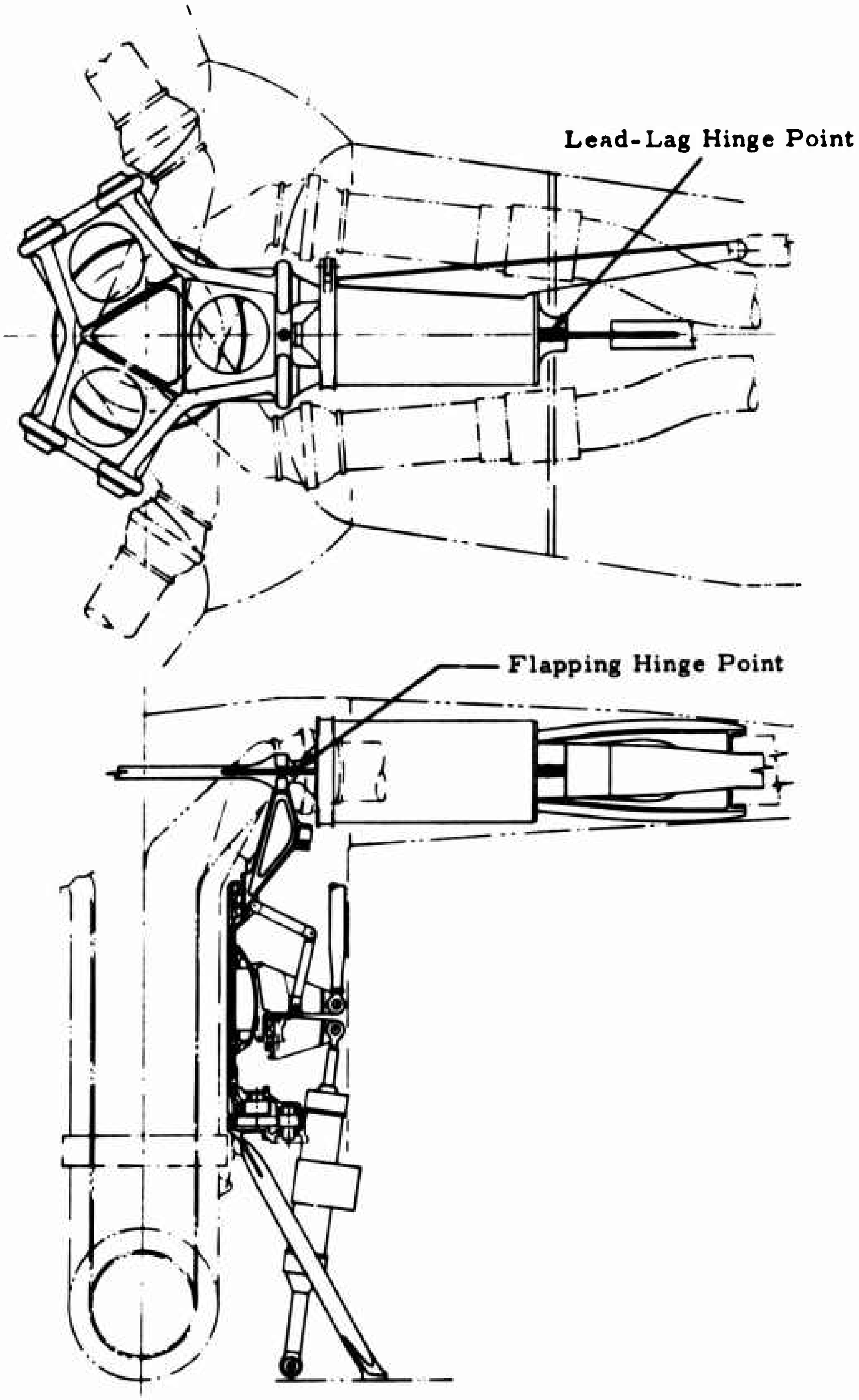


Figure 39. Articulated Hub With External Controls.

direct in the hub structure and hub tilting is not necessary, the weight of this configuration is much lower and the fairing envelope much smaller than for the tilting hubs.

### ARTICULATED HUB WITH EXTERNAL CONTROLS (Figure 39)

This configuration is almost identical with the articulated hub with internal controls. The difference is primarily in the relocation of the swashplate assembly, shaft, and mast to the outside of the gas ducts. This configuration is the lightest, as a result of the simpler ducts and flight controls, and has the smallest fairing envelope of all the hubs considered.

## BLADE CONFIGURATIONS STUDIED

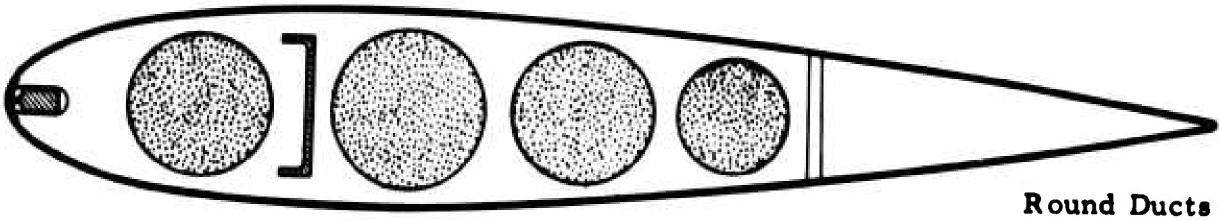
Several blade configurations, their design dictated primarily by gas duct style and shape, were studied and are shown in Figure 40. Common to all these configurations is the structural concept used on the XV-9A of constructing the blades in segments held together with flexures and spars and using nonstructural trailing-edge segments. This type of construction permits carrying all flapwise bending and centrifugal loads in the spars with the segments and flexures providing the load path for blade torsion, chordwise shear, and air loads. This configuration also allows for thermal expansion, minimizing loads in hot parts. In addition to duct shape, the other blade variables introduced into the parametric study were chord, radius, thickness ratio, number of blades, and number and location of spars. The detailed blade weight breakdown used in the computer study is shown in Table XIII. All blades studied use an NACA 0018 section on the inboard 0. 75 radius and an NACA 0018, 0016, or 0014 on the outboard 0. 25 radius.

### ELLIPTICAL DUCTS

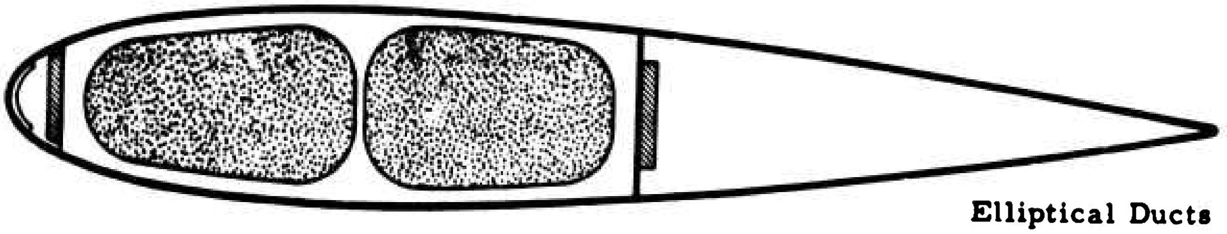
The first blade configuration studied was one dictated by the elliptical duct as used on the XV-9A. This type of duct is an integral structural part of each segment and produces the highest ratio of duct area to airfoil cross section area. To minimize thermal stress problems, a thin duct liner was used to reduce the duct wall temperature.

### ROUND DUCTS

The second configuration studied was one utilizing multiple round ducts that are not an integral part of the structure. Gas-tight bellows are installed at spanwise intervals as necessary to provide for duct expansion and to eliminate duct bending stresses. This type of construction minimizes the



Round Ducts



Elliptical Ducts

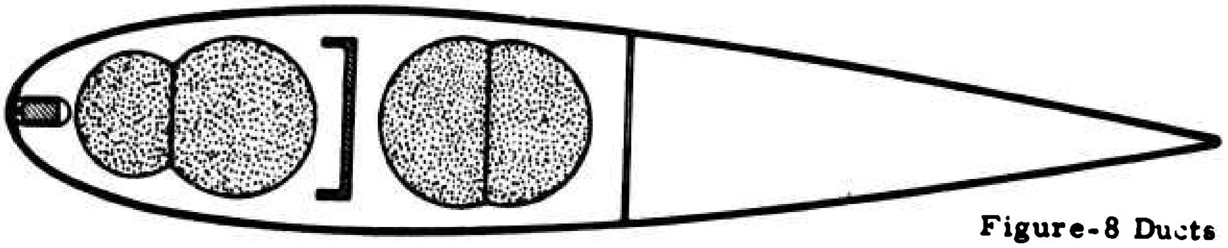


Figure-8 Ducts

Figure 40. Blade Duct Configurations.

## TABLE XIII
## WEIGHT FOR VARIOUS BLADE SECTIONS*

| | Pounds Per Inch | | | | |
|---|---|---|---|---|---|
| | 395-0902 1 Spar at 28% | 395-0903 2 Spar | 395-0904 XV-9A Type | 395-0905 1 Spar at 30% | 395-0907 1 Spar at 25% Stepped Airfoil |
| Trailing edge structure | 0.1402 | 0.1286 | 0.1295 | 0.1402 | 0.1523 |
| Main segment skin and corrugations | 0.3215 | 0.2281 | 0.1573 | 0.2758 | 0.2999 |
| Ribs and caps | 0.2399 | 0.2189 | 0.4518 | 0.2323 | 0.2227 |
| Closure channel | 0.0288 | 0.0474 | 0.1434 | 0.0260 | 0.0306 |
| Rib stiffeners | 0.0396 | 0.0294 | - | 0.0345 | 0.0196 |
| Flexure | 0.1496 | 0.1364 | 0.1664 | 0.1343 | 0.1210 |
| Duct (forward) | 0.1565 | 0.1541 | 0.1716 | 0.1526 | 0.1401 |
| Duct (aft) | 0.1565 | 0.1541 | 0.1664 | 0.1526 | 0.1401 |
| 28 percent or 30 percent channel | 0.0660 | - | - | 0.0570 | 0.0600 |
| Leading edge | - | 0.0450 | 0.0386 | - | - |
| Total nonbending material | 1.2986 | 1.2420 | 1.4250 | 1.2053 | 1.1863 |
| Front spar (at tip) | 0.214 | 0.143 | 0.143 | 0.214 | 0.214 |
| Rear spar (at tip) | - | 0.086 | 0.086 | - | - |
| Balance (at 23%) | 0.5804 | 0.6954 | 0.6241 | 0.6891 | 0.4264 |
| Total blade weight at tip** | 2.0930 | 2.1664 | 2.2781 | 2.1084 | 1.8276 |

*Based on full-scale layouts optimized for skin gages and materials.
**Excluding cascade.

NOTE: All sections use the figure-8 duct except 395-0904, which uses the elliptical duct.

thermal stress problems and makes it possible to design a structure that lends itself to ease of inspection and repair. However, this configuration was abandoned because of the poor ratio of duct area to airfoil area and the difficulty in pairing and sizing the ducts to make possible separate engine ducts. It also required more balance weight in the leading edge to provide for an acceptable chordwise center of gravity location.

### FIGURE-8 DUCTS

The third configuration considered is identical with the round duct type in principle except that the duct shape has been changed to a figure-8 formed by two intersecting circles. A web connecting the intersection points of the perimeters completes the duct. The figure-8 duct results in an adequate ratio of duct area to airfoil area, as well as two equal-area ducts necessary for separate engine operation. External insulation on the ducts minimizes the thermal stress effects.

## ELEMENTS OF THE PARAMETRIC STUDY

### WEIGHT EQUATIONS

The weight equations used in the parametric study are based on HTC-AD data compiled from analytical evaluations and statistical studies of numerous production and proposed research aircraft. The weight effects peculiar only to the Hot Cycle rotor system are also reflected in the group weight equations, where applicable.

With few exceptions, these equations were developed from the most basic parameters describing helicopter vehicles; namely, gross weight, rotor radius, and ultimate load factor. The end result produced expressions of simple form that checked reasonably with other estimating methods.

A detailed discussion and substantiation of the equations used may be found in the Weight section of this report.

### WEIGHT EQUATIONS FOR ROTOR BLADE

The Hot Cycle rotor employs a unique type of blade construction with rotor diameter and blade chord larger than those of most existing rotors. As a result of this unique design, a method of rotor weight prediction more realistic and reliable than extrapolation of statistical data on existing rotors is necessary.

To meet this requirement, an analytical method using the results of detail layouts and stress analyses of the actual blade structural configurations has been developed. The blade structure is divided into two categories; namely, (1) basic bending structure (spars) carrying flapwise loads, chordwise bending loads, and centrifugal force; and (2) nonbending structure (comprising ducts, ribs, heat shielding, flexures, skins, leading and trailing edge fairings, and miscellaneous hardware) carrying local airloads, duct gas pressures, thermal gradients, chordwise shear, and blade torsional loads.

The analytical method starts with the carefully designed and analyzed nonbending material weight and develops the spar sizes needed to support the total blade weight in ground flapping and ground wind conditions, to resist the flapwise and chordwise fatigue moments, to prevent proximity of flapwise and chordwise natural frequencies to integer multiples of rotor speed, and to provide chordwise balance as indicated by aeroelasticity considerations.

The method provides for a three-step spanwise variation in exterior skin thickness in order to efficiently fulfill the torsional strength and stiffness requirements along the blade length. In addition, a reduced chord length section may be incorporated on the inboard portion of the blade to minimize torsional divergence problems on the retreating blade. Spars are assumed to have a bilinear taper, thus permitting design of the tip station (station t), the 75-percent radius station (station 2), and the 20-percent radius station (station $\ell$ ) to meet the strength requirements precisely. The spanwise weight distribution established by the foregoing considerations is presented in Appendix III. It is apparent that enough flexibility in choice of parameters is available to provide a very close approximation of actual blade weight distribution.

## ROTOR DESIGN LOADS

The most critical item in an analytical weight prediction is the accurate determination of the design loads. Therefore, as much data as possible has been obtained from flight test data on similar blades flown on the XV-9A Hot Cycle research aircraft. For the nonbending material, thermal gradients, which are a major source of structural stress, were based on flight measurements modified to account for the slightly different hot gas conditions associated with the GE1/J1 engines. Torsional loads have been scaled from flight measurements, using a scale factor proportional to gross weight times blade chord. Spar fatigue loads, both flapwise and chordwise, for the tilting-hub rotor have been scaled from XV-9A flight data, using a scale factor proportional to gross weight times radius. For the fully articulated rotor, flight test data from the OH-6A and the CH-34A helicopters have been scaled to be proportional to gross weight times radius. Dynamic effects are accounted for by using a dynamic amplification factor similar to that for a single-degree-of-freedom system, based on first mode natural frequency compared with nearest applied frequency (one per rev chordwise, three per rev flapwise). Natural frequency based on rotating mode shapes has been calculated by computing charts similar to those shown in Yntema (Reference 16). The charts were computed using a Myklestad method and assuming linear mass and stiffness distributions and a distributed tip weight. As Reference 16 uses nonrotating mode shapes and tip weight lumped at the tip of the blade, the frequencies computed herein are more accurate. An additional effect of flapwise stiffness is accounted for by assuming the bending radius of curvature to be constant for a given centrifugal force. Vibratory flapwise bending stress, as a result, is proportional to spar depth. Maximum cruise speed loads have been increased by 20 percent to obtain design endurance limit fatigue loads. Fatigue stress allowables are based on full-scale tests of the XV-9A blades.

## ANALYTICAL PROCEDURE

In outline, the analytical method proceeds as follows (equations referenced may be found in Appendix III):

1. For a given rotor radius and chord, calculate center of gravity and nonbending material weight from equations (53) and (54), which were based on data obtained from detail layouts.
2. From equation (55), compute main spar weight at tip ($W_{r_t}$) required to support the centrifugal force generated by the tip weight and the cascades plus the gas pressure on the cascades.
3. Compute the front spar weight required for chordwise balance (equation 56).
4. Compute nonbending material weight and its chordwise location at station 2 (75-percent radius) from equations (57) and (58).
5. Compute station 2 design fatigue moment from equation (59).
6. The roots of equation (60) give the front spar weight needed at station 2 to meet chordwise balance requirements in conjunction with the flapwise and chordwise fatigue stress requirements in both the front and main spar. Substitute the front spar weight from equation (60) into equation (61) to obtain the total section weight at station 2.
7. Compute nonbending material weight and center of gravity at station 1 (50-percent radius) from the appropriate ones of equations (62) and (63).
8. The dead weight bending moment at station 1, excepting the part contributed by the unknown spar weight at station 1, is given by equation (64).
9. Compute from equation (65) the front spar weight required to balance (1) the nonbending material and (2) the main spar weight necessary to support the ground flapping and ground wind conditions at station 1.

10. Compute the design fatigue moments at station 1 from equation (66).

11. Compute the front spar weight required to take the flapwise fatigue loads and the cf and to provide chordwise balance at station 1 from equation (67). If the spar weight is determined by flapwise fatigue loads, then the depth of the spar is lowered until the blade is designed by ground flapping. In addition, when the front spar is critical for chordwise fatigue stress instead of for chordwise balance, the balance equations are bypassed and the main spar is designed by strength requirements. This results in a favorable chordwise balance further forward than the design requirements specify.

12. Using the largest value of front spar weight at station 1, compute the total section weight at station 1 from equation (68) and extend to total root section weight by equation (69).

13. Total blade weight required to meet all static and fatigue criteria and to meet the prescribed chordwise balance condition is given by equation (70).

14. Blade stiffness and inertial properties are given by equations (71), (72), (73), (74), (75), and (76).

15. Compute the flapwise and chordwise natural frequencies.

16. Compute the dynamic amplification factors from equation (59a) (includes 10-percent damping).

17. Recompute the fatigue moment from equations (59b) and (66a), and repeat steps 6 and 8 through 16. Continue to convergence.

For the compound configuration:

18. Check for bending instability at $\Psi$ = 180 degrees. Add tip weight if instability is found. Check for retreating blade torsional divergence. If necessary, increase the blade skin gages for greater torsional stiffness. Check advancing blade flutter, including three modes of vibration -- flapping, first bending, and first torsional. If a disturbance does not damp to one-half amplitude in two cycles or less, move the chordwise balance forward by the addition of tip weight.

19. Repeat steps 2 through 18 until stability is obtained.

## ROTOR HORSEPOWER AND CRUISE FAN PERFORMANCE

In order to calculate the rotor horsepower available, it is necessary to determine the conditions at the blade tip. These items depend on the duct

Mach number, duct friction coefficient, and hydraulic diameter of the ducts. The duct Mach number is a function of the engine exit conditions and the duct area. The available duct area and the duct wetted perimeter are determined from blade design drawings. These characteristics are then made into general equations in order to design helicopters of various sizes. Equations are prepared by investigation of blades of various chords, thickness ratios, and rear spar locations. Blade root duct area and hydraulic diameter equations for the figure-8 duct design are as follows:

$$A_D = 200\,\bar{c}^{2}\left(\frac{\frac{t}{c}}{0.18}\right)\left(\frac{\frac{X_r}{c}}{0.528}\right)^{1.20} \tag{29}$$

$$D_h = 9.05\,\bar{c}\left(\frac{\frac{t}{c}}{0.18}\right)^{0.2}\left(\frac{\frac{X_r}{c}}{0.528}\right)^{0.93} \tag{30}$$

For the outboard portion of the rotor blade ($t/c = 0.14$):

$$A_D = 205\,\bar{c}^{2}\left(\frac{\frac{t}{c}}{0.18}\right)^{1.5}\left(\frac{\frac{X_r}{c}}{0.528}\right)^{1.20} \tag{31}$$

$$D_h = 9.20\,\bar{c}\left(\frac{\frac{t}{c}}{0.18}\right)^{0.31}\left(\frac{\frac{X_r}{c}}{0.528}\right)^{0.93} \tag{32}$$

The corresponding equations for the elliptical (XV-9A) duct area follow.

$$A_D = 106.5\,\bar{c}^{2.13}\left(\frac{\frac{X_r}{c}}{0.524}\right)^{0.85}\left(\frac{\frac{t}{c}}{0.18}\right)^{0.95} \tag{33}$$

$$D_h = 7.717\,\bar{c}^{0.94}\left(\frac{\frac{X_r}{c}}{0.524}\right)^{0.07}\left(\frac{\frac{t}{c}}{0.18}\right)^{0.87} \tag{34}$$

With these equations, use of the proper dimensions will lead to duct area and hydraulic diameter.

The next step is to determine blade duct inlet Mach number. This is found from the following relationships, using consistent values of flow, temperature, and pressure ($W_8$, $T_8$, and $P_8$) taken from engine characteristics. These are determined from an IBM deck for engines such as the GE1/J1 or GE T64/S4B.

$$\left(\frac{W\sqrt{T}}{A_D}\right)_9 = \frac{N_{eng}\, W_8\, \sqrt{T_8}}{bA_D\, \frac{P_9}{P_8}\, P_8} = \frac{M_9\sqrt{\frac{g\gamma}{R}}}{\left(1 + \frac{\gamma - 1}{2}\, M_9{}^2\right)^{\frac{\gamma + 1}{2(\gamma - 1)}}} \qquad (35)$$

$$\frac{P_9}{P_8} = 1 - 0.04 = 0.96 \text{ based on XV-9A tests}$$

This equation must be iterated to find the blade duct inlet Mach number, which is typically 0.30 to 0.40.

The next step is to find the variation of duct Mach number down the blade. The blade is broken into a number of equal stations (say 20) and is further divided into two thickness ratios at an arbitrary station. The basic relationship of the Mach number change is taken from Reference 17, and assuming constant area over the duct length being checked, the following results:

$$\Delta M = \left[2f\, \frac{\gamma R}{D_h}\left(\frac{\Delta r}{R}\right)\right]\left[\frac{M^3\left(1 + \frac{\gamma - 1}{2}\, M^2\right)}{1 - M^2}\right] - \left[\frac{V_T{}^2}{2gRT_8}\left(\frac{\Delta r}{R}\right)^2\right](2n + 1)\left[\frac{M\left(1 + \frac{\gamma - 1}{2}\, M^2\right)^2}{1 - M^2}\right] \qquad (36)$$

The first term involves the friction coefficient, f, which is conservatively assumed to be 0.004, as 0.003 was measured during XV-9A tests. The second term is related to centrifugal force.

The total change in Mach number is accumulated from duct inlet to the arbitrary station where the area change takes place. At this location, the total pressure is determined from the following relationship:

$$\frac{P_{T_{9.7}}}{P_{T_9}} = \frac{M_9}{M_{9.7}} \left( \frac{1 + \frac{\gamma - 1}{2} M_{9.7}^2}{1 + \frac{\gamma - 1}{2} M_9^2} \right)^{\frac{\gamma + 1}{2(\gamma - 1)}} \tag{37}$$

and the new flow function is found from

$$\left( \frac{W\sqrt{T}}{A_D P_T} \right)_{9.7} = \left( \frac{W\sqrt{T}}{A_D P_T} \right)_9 \times \frac{1}{\left( \frac{P_{T_{9.7}}}{P_{T_{9.0}}} \right)} \tag{38}$$

A new Mach number on the other side of the area change is found from the change in duct area

$$\left( \frac{W\sqrt{T}}{A_D P_T} \right)_{9.71} = \left( \frac{W\sqrt{T}}{A_D P_T} \right)_{9.7} \times \left( \frac{A_{D_{9.7}}}{A_{D_{9.71}}} \right) = \frac{M_{9.71} \sqrt{\frac{g\gamma}{R}}}{\left(1 + \frac{\gamma - 1}{2} M_{9.71}^2\right)^{\frac{\gamma + 1}{2(\gamma - 1)}}} \tag{39}$$

which is iterated to find $M_{9.71}$.

A new value of hydraulic diameter, $D_h$, using equation (32), is also calculated, and then the process of accumulating duct Mach number changes to the blade tip is performed.

At the blade tip, the pressure ratio is finally determined as:

$$\frac{P_{T_{10}}}{P_{T_9}} = \frac{M_9}{M_{10}} \frac{A_{D_{9.0}}}{A_{D_{9.7}}} \left( \frac{1 + \frac{\gamma - 1}{2} M_{10}^2}{1 + \frac{\gamma - 1}{2} M_9^2} \right)^{\frac{\gamma + 1}{2(\gamma - 1)}} \tag{40}$$

With the tip pressure ratio known, it is now possible to determine available energy per degree from the relationship

$$\text{available energy per degree} = \left(\frac{AE}{T}\right) = C_p\left[1 - \left(\frac{1}{\frac{P_{T_{10}}}{P_{T_0}}}\right)^{\frac{\gamma - 1}{\gamma}}\right] \quad (41)$$

and

$$\text{jet velocity} = V_j = 224\ C_v\ \sqrt{\frac{AE}{T}\ T_8} \quad (42)$$

Velocity coefficient, $C_V$, is assumed to be 0.98. The XV-9A value was measured at 0.94. It is assumed from available data that with a development program the higher value can be achieved.

Based on measurements of the temperature in the XV-9A blade, the tip temperature is taken as being equal to the engine discharge temperature. (The temperature drop through the duct is approximately equal to the temperature rise due to centrifugal pumping.)

Finally, to get rotor horsepower, the following calculation is performed:

$$\text{rhp} = \frac{W_g\left(V_j - V_T\right)V_T N_{eng}}{g \times 550} \quad (43)$$

and specific fuel consumption is given by

$$\text{SFC} = \frac{N_{eng}\ W_f}{\text{rhp}}$$

where again fuel flow, $W_f$, is taken from engine data consistent with the other gas conditions.

This value of specific fuel consumption is determined as a function of rotor horsepower over the range of powers.

In addition to the determination of power with all engines, an alternate case was prepared with one engine out. This case checks the potential improvement of specific fuel consumption with reduced duct losses (assuming a variable area nozzle). Another possible improvement of specific fuel consumption might be checked by using four engines instead of two for the one-engine-out case. The increase in power available with three engines instead of one engine remaining operational permits cruise at a higher altitude and can possibly lead to greater range.

Cruise fan performance was derived from General Electric Report R64FPD155a. Since this report is classified, the data are not included.

## INTEGRATION OF ELEMENTS OF THE STUDY WITH MISSION REQUIREMENTS

The integration of the elements of the study and the mission requirements utilized an IBM computer and was based on the program outlined below.

## PERFORMANCE COMPUTATION METHOD

The method of computing power required for helicopter performance for both hovering and forward flight is presented in Reference 3. The compound helicopter flight was computed by standard methods with the addition of rotor thrust and drag. Rotor aerodynamic data from References 18 (Figures 19 and 28), 19 (Figure 4), and 20 (Figures 3 and 4) indicate that at advance ratios of more than 1. 0 the lift coefficient/solidity ratio is a constant equal to 0. 1667 and the drag coefficient/solidity ratio is a constant equal to 0. 03888. The thrust of the autorotating rotor is then $T = 0.16667\ A_b \times q$, and the drag is equal to $D = 0.03888\ A_b \times q$.

### Pure Helicopter

The design gross weight (transport mission weight) is computed as the maximum gross weight for hovering out of ground effect at 6, 000 feet, 95°F day, with takeoff power. This includes a download factor on the fuselage.

The blade weight and component weights are computed to determine the empty weight and the minimum flying weight.

The heavy-lift weight is defined as the weight for hovering out of ground effect, sea level standard day, or the weight for a design load factor of 2, whichever is lower.

Available rotor horsepower is reduced 2 percent for yaw control requirements.

The payloads for both the 100-nautical-mile transport mission and the 20-nautical-mile heavy-lift mission are determined as follows:

1. Warmup and takeoff fuel for 2 minutes at normal rated power is computed.
2. Power for hover at takeoff weight less warmup and takeoff fuel is determined and fuel for the start hover time is obtained.
3. The fuel flow for cruise at the takeoff weight less hover, warmup, and takeoff fuel is computed. Using this fuel flow, the average weight for the outbound leg is estimated by subtracting the fuel required for one-half the radius.
4. Using the average weight, the cruise fuel flow is recomputed and the fuel for the outbound leg is determined.
5. The hovering power and fuel flow for the landing weight after the outbound le, determine the hover fuel at midpoint.
6. An estimate of the reserve fuel is made, assuming that the fuel for the return leg is the same as that for the outbound leg. This is added to the minimum flight weight to give the mission landing weight.
7. The fuel flow for cruise at the landing weight is used to compute an average weight for the inbound leg. In addition, a more accurate estimate of the reserve fuel and landing weight can be made.
8. Using the average weight, the fuel for the inbound leg is determined.
9. The payload is the takeoff weight less the minimum flight weight and mission fuel, including reserve.

The configurations with internal loading are assumed to require twice the takeoff and warmup fuel, as they would have to shut down to unload at the destination. It is assumed that with external loading, the ship does not land at the destination.

The cruise speed of the outbound leg is 95 knots for the heavy-lift mission and 110 knots for the transport mission, or speed for best range, whichever is larger, unless the retreating tip drag coefficient is greater than 0.06. This value is assumed to be the stall limit of the ship and will be the maximum speed with the required weight and parasite area. The return leg is at 130 knots, or speed for best range, if it is greater, unless the stall limit is reached.

In the program, the ferry range is computed using the following method. The curve of specific range ($R_{sp}$ = nmi/lb of fuel) is assumed to have the following form:

$$R_{sp} = \frac{C_1}{W} + C_2 \left( W - \frac{W_{TO} - W_L}{2} \right) \quad (44)$$

W = weight at given range

where:

$W_{TO}$ = takeoff weight

$W_L$ = landing weight

Integrating this,

$$\text{range} = C_1 \left( \ell n \frac{W_{TO}}{W_L} \right) \quad (45)$$

The range is assumed to be a climb cruise; thus, to determine the constants in the equation, the specific range at takeoff weight at sea level and the landing weight at best cruise altitude up to 20,000 feet are used to give two equations in two unknowns, $C_1$ and $C_2$. $C_1$ can then be used in the range equation. To make an allowance for climb fuel, an energy equation is used:

$$W_{f_t} = (SFC) \left( \frac{HW_s}{325\eta} \right) \quad (46)$$

where H = altitude in nautical miles

$W_s$ = weight at start of climb

Then one-half of the climb fuel is subtracted from the takeoff weight and one-half is added to the landing weight.

As the ferry mission can be performed with a running takeoff, the takeoff weight is determined by allowable load factor or the maximum weight at which the ship will cruise at 60 knots or greater, as limited by retreating blade stall. In this condition, the ship is never allowed to exceed maximum continuous power. If the ship will not cruise at a speed of 60 knots, the takeoff weight is reduced by 5,000-pound increments until 60 knots can be achieved. As the cruise with takeoff weight is usually stall-limited, hovering tip speed is used, as this greater tip speed results in a higher stall speed.

The specific range at landing weight is determined as follows. The specific range at 20,000-foot altitude with two engines is computed. If this is stall-limited, the specific range at 15,000 feet is computed, and so on, until speed for best range can be achieved. The same procedure is followed using one engine, and then the largest value of specific range is used in the range equation.

A more detailed computation of ferry range was made for the best rotor for each configuration.

The takeoff weight is determined as the lowest of the following: 2-g load factor or cruise at 60 knots, as limited by retreating tip stall or normal continuous power. The energy equation was used for climb fuel, assuming a final altitude of 20,000 feet and average flight weight. One-half of this fuel was subtracted from the takeoff weight and one-half was added to the minimum flight weight. The minimum flight weight also includes a reserve of 10 percent of the total cruise fuel.

For each weight, the specific range is optimized for hovering or cruise tip speed, altitude, and use of one or two engines. The ferry range was then determined by integrating the specific range using Simpson's rule.

## Compound Helicopter

The design weight and heavy-lift weight are determined in the same manner as for the pure helicopter, including the download on the fuselage and wing folded at 60-percent span.

The blade weight computation includes a check (and standard beef-up, if necessary) of bending stability, torsional divergence, and flutter at the high advance ratio conditions appropriate to compound helicopter flight.

The payload is computed for 200-, 300-, and 500-nautical-mile missions at sea level and altitude for best cruise. These missions are computed in the same manner as the helicopter mission.

The heavy-lift and overload weights are checked for ability to perform a transition. If this cannot be achieved, the weights are reduced. The criterion for transition is an overlap of 20 knots between helicopter and autogyro flight and an overlap of 20 knots between autogyro and airplane flight.

In helicopter flight, it is assumed that the flaps are used on the wings to compensate for the download that would result from the nose-down angle of the fuselage, resulting in zero lift. The wing profile drag coefficient is assumed to be raised to 0.03. This is the value with flaps deflected enough to compensate for a negative angle of 5 degrees. As the wing has flaps, a maximum $C_L$ of 2 is assumed for airplane flight.

The method of computing the ferry mission is the same as that used in the helicopter routine.

## RESULTS OF THE PARAMETRIC STUDY

The results of the parametric study show the ability of all configurations to meet the mission requirements with a rotor of modest size and remarkably high payload-to-empty-weight ratios. For configurations 2 and 3, the rotor size was determined by the requirement for the ship to have the ability to make a safe autorotational landing at design gross weight. The transport mission is critical for configuration 1, and the ferry mission is critical for configuration 4. It is well to note that the optimum rotor was arrived at by a process of elimination. Thus the tables showing the effect of the parametric variables are examples taken for rotors where comparable data were available and are not necessarily the selected optimum rotors.

## EFFECT OF PARAMETRIC VARIABLES ON MISSION PERFORMANCE

### Effect of Restrained Versus Articulated Hub

It became obvious early in the program that the articulated hub would provide a substantial saving in weight and hub envelope size. This reduction in weight and drag for a rotor of a given size resulted in an increase in range in the order of 25 percent, an increase in transport payload of 70 percent, and an increase in heavy-lift payload of approximately 20 percent over the restrained tilting hub. The tilting hub, employing rigid chordwise blade restraint with its attendant higher blade stresses, also results in an empty weight approximately 33 percent greater than that for an articulated rotor of the same size. These differences are shown in Table XIV. The major share of the noted weight difference is in the rotor itself, the restrained tilting rotor with in-plane rigidity being almost twice the weight of the same size articulated rotor.

### Effect of Internal Rotor Controls Versus External Rotor Controls

A small improvement in performance is gained by using an external controls instead of the internal controls on the articulated hub, as shown in Table XV. This is because of its smaller envelope and lighter weight.

### Effect of Blade Duct Shape

As shown in Table XVI, the figure-8 duct blade proved superior to the elliptical duct blade. Though the elliptical duct area was greater, results

## TABLE XIV

## EFFECT OF TILTING HUB WITH RESTRAINT VERSUS ARTICULATED HUB

| | Tilting Hub | Articulated Hub |
|---|---|---|
| Mission | | |
| Ferry | 1,374 nmi | 1,731 nmi |
| Transport | 7.02 tons | 11.96 tons |
| Heavy-lift | 22.57 tons | 27.30 tons |
| Empty weight | 37,269 lb* | 27,926 lb* |
| Payload/empty weight | | |
| Transport | 0.3378 | 0.7426 |
| Heavy-lift | 1.2112 | 1.9552 |
| Computer run number (configuration 1, 55-foot radius, 55-inch chord, $V_t$ hover = 750 fps, $V_t$ cruise = 700 fps) | 1-5 | 1-3 |

*Empty weight for heavy-lift mission only (no pod).

## TABLE XV

## EFFECT OF INTERNAL ROTOR CONTROLS VERSUS EXTERNAL ROTOR CONTROLS

| | Articulated Hub | |
|---|---|---|
| | Internal Shaft | External Shaft |
| Mission | | |
| Ferry | 1,685 nmi | 1,715 nmi |
| Transport | 11.96 tons | 12.08 tons |
| Heavy-lift | 26.58 tons | 26.66 tons |
| Empty weight (transport mission) | 24,331 lb | 24,189 lb |
| Payload/empty weight | | |
| Transport | 0.8355 | 0.8477 |
| Heavy-lift | 2.1849 | 2.2043 |
| Computer run number (configuration 1, 50-foot radius, 55-inch chord, $V_t$ hover = 750 fps, $V_t$ cruise = 700 fps) | 1-8 | 1-4 |

## TABLE XVI
## EFFECT OF BLADE DUCT SHAPE

| | Elliptical Duct | Figure-8 Duct |
|---|---|---|
| Mission | | |
| Ferry | 1,731 nmi | 1,791 nmi |
| Transport | 11.96 tons | 12.20 tons |
| Heavy-lift | 27.3 tons | 27.47 tons |
| Empty weight (transport mission) | 27,926 lb* | 26,303 lb* |
| Payload/empty weight | | |
| Transport | 0.7426 | 0.7974 |
| Heavy-lift | 1.9552 | 2.0887 |
| Computer run number (configuration 1, 55-foot radius, 55-inch chord, $V_t$ hover = 750 fps, $V_t$ cruise = 700 fps) | 1-3 | 1-4 |

*Empty weight for transport mission only (includes pod).

of the study show that this benefit was more than offset by the lighter construction of the figure-8 blade and that its duct area was adequate. Because of this tradeoff and the difficulty of transferring the centrifugal load from the two spars to the lead-lag hinge, the elliptical duct configuration was abandoned early in the study.

### Tip Speed

Tip speeds of 750, 725, and 700 feet per second in hover were used with cruise tip speeds of 725, 700, and 675 feet per second. It was determined that, in general, a high tip speed will give better performance for heavy-lift operations and extended hovering times and a lower tip speed is favored for performance at cruise with a lesser payload. Thus, a constant tip speed for all missions can be considered only as a poor compromise, as indicated by the figures in Table XVII, where the results for the best constant tip speed ship are listed in the last column.

Fortunately, the Hot Cycle principle allows for a quick and easy adaption of tip speeds in a rather wide range for best mission performance. In opposition to a gear-driven helicopter, no penalties will result as to

## TABLE XVII
## EFFECT OF ROTOR-BLADE TIP SPEED

| | Tip Speed (fps) | | | | |
|---|---|---|---|---|---|
| | Hover = 750<br>Cruise = 700 | Hover = 750<br>Cruise = 675 | Hover = 725<br>Cruise = 700 | Hover = 725<br>Cruise = 675 | Hover = 725<br>Cruise = 725 |
| Mission | | | | | |
| Ferry | 2,038 nmi | 2,065 nmi | 1,917 nmi | 1,949 nmi | 1,868 nmi |
| Transport | 13.97 tons | 13.98 tons | 13.82 tons | 13.83 tons | 13.77 tons |
| Heavy-lift | 25.39 tons | 25.40 tons | 25.13 tons | 25.14 tons | 25.12 tons |
| Empty weight | 17,832 lb | 17,832 lb | 17,637 lb | 17,637 lb | 17,637 lb |
| Payload/empty weight | | | | | |
| Transport | 1.5665 | 1.5680 | 1.5670 | 1.5685 | 1.5612 |
| Heavy-lift | 2.0848 | 2.8489 | 2.8494 | 2.8503 | 2.8482 |
| Computer run number (configuration 2, 40-foot radius, 60-inch chord) | 2-13 | 2-15 | 2-13 | 2-13 | - |

gearbox life, fuel consumption, and engine performance, if not operated at the design point. For example, increasing rotor tip speed for improved hover and heavy-lift capability also raises the Hot Cycle propulsion efficiency. This improvement cannot be found in the shaft-driven helicopter.

### Effect of Blade Chord Length

Blade chord lengths were varied from a minimum of 45 inches to a maximum of 65 inches. The optimum chord for best performance is nominally 60 inches, depending upon spar location and blade radius. Changing one parameter quite often requires the change of an additional one to approach the optimum rotor. This interaction of spar position, chord length, and blade radius is discussed further under Effect of Spar Position. The effect of varying chord length is shown in Figures 41 and 42.*

### Effect of Blade Radius

The effect of varying the blade radius for each rotor configuration is shown in Figures 43 and 44. In general, the performance drops off with any decrease in radius from the minimum selected to meet the mission.*

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*The discontinuities of the ferry range curves in Figures 41, 43, and 44 are the result of changes of cruise altitude and/or number of engines operated.

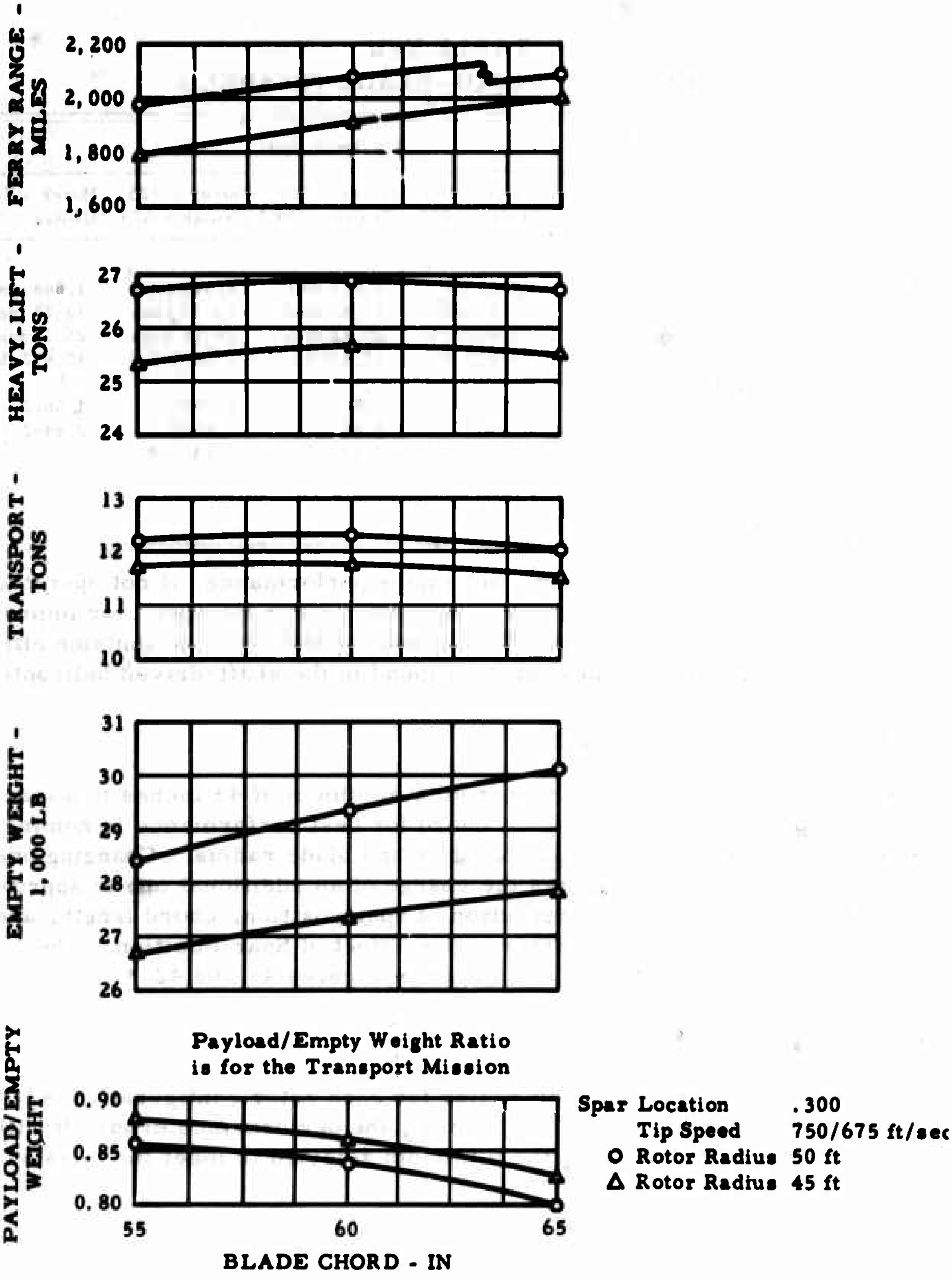


Figure 41. Influence of Blade Chord - Configuration 1.

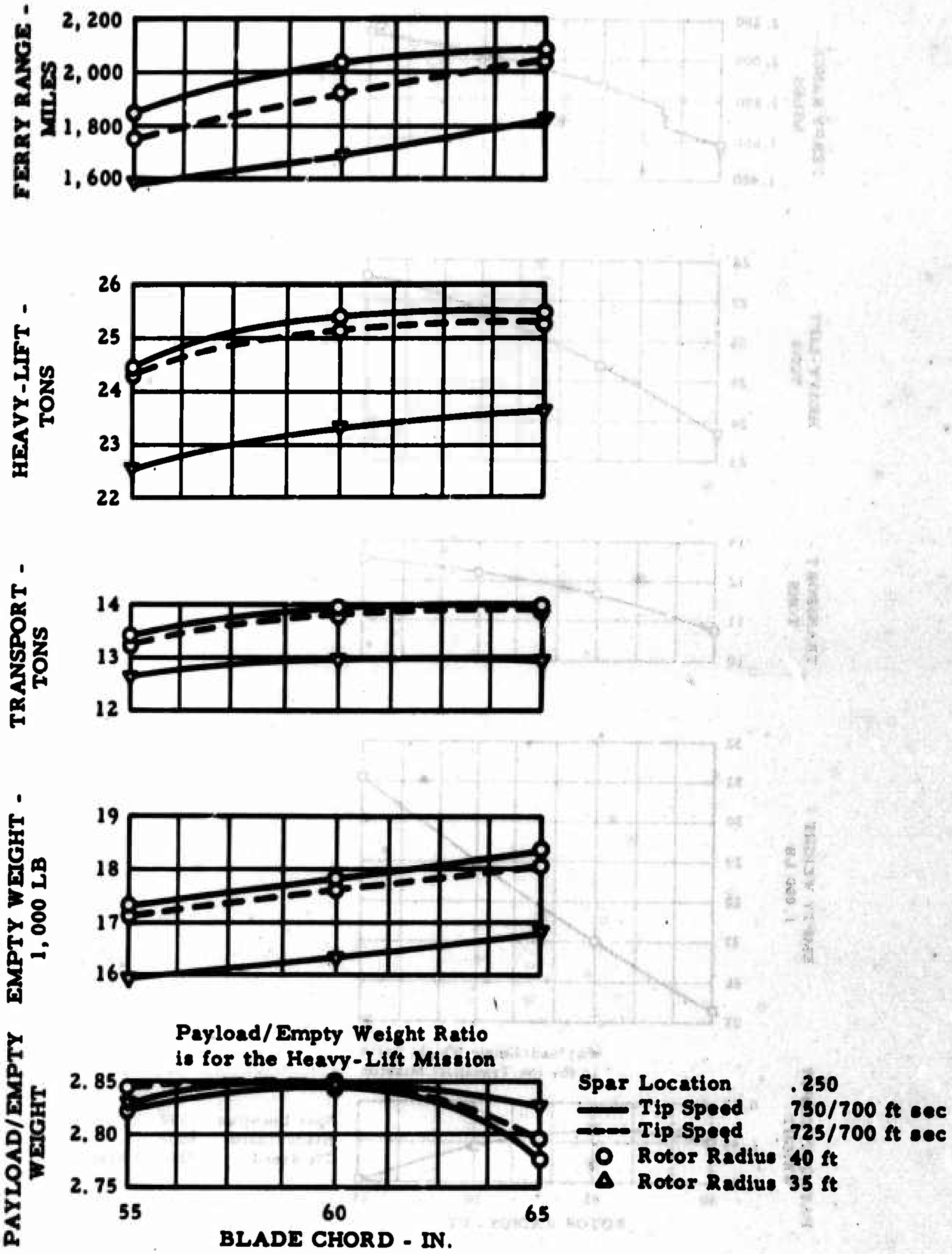


Figure 42. Influence of Blade Chord - Configuration 2.

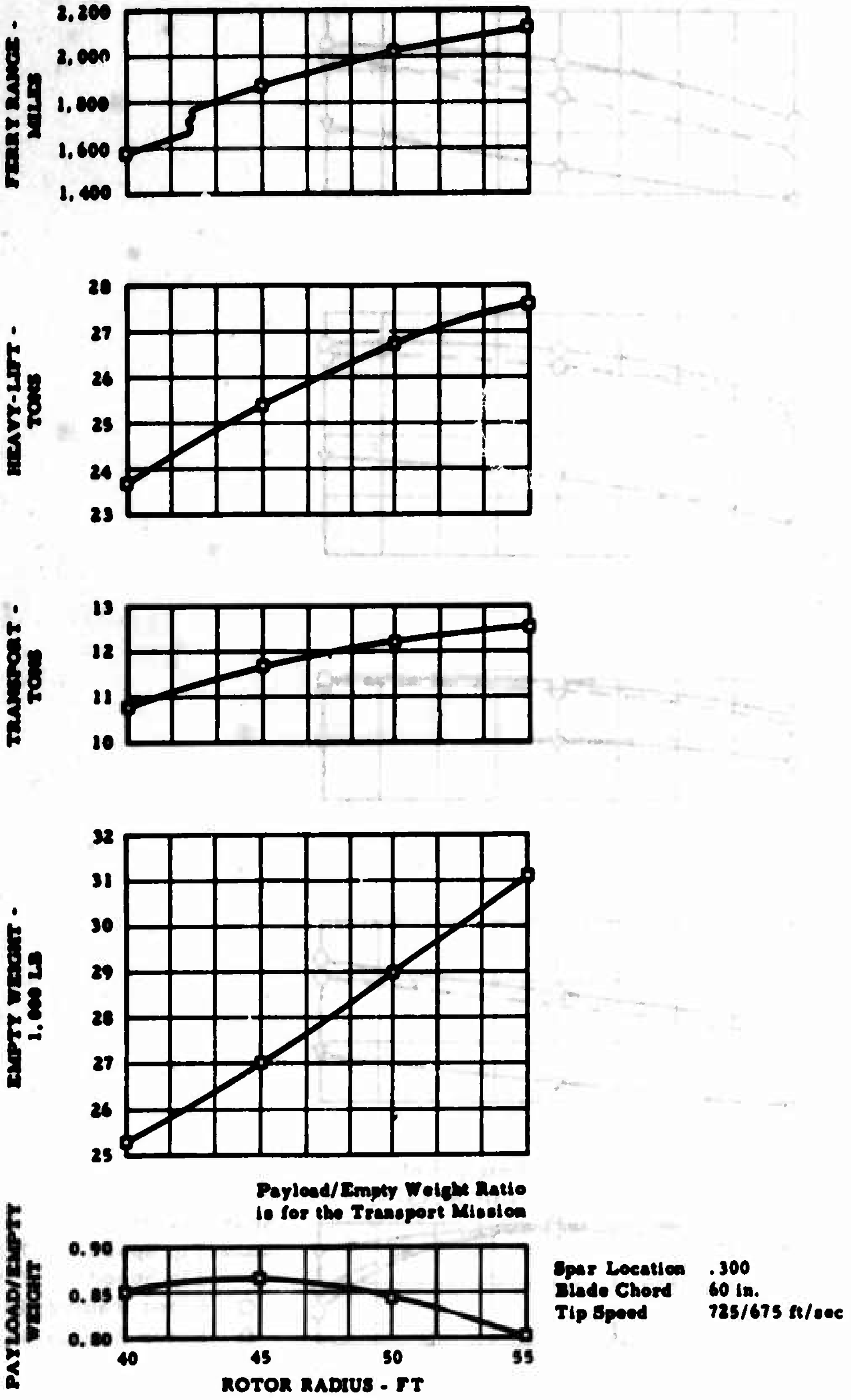


Figure 43. Influence of Rotor Radius - Configuration 1.

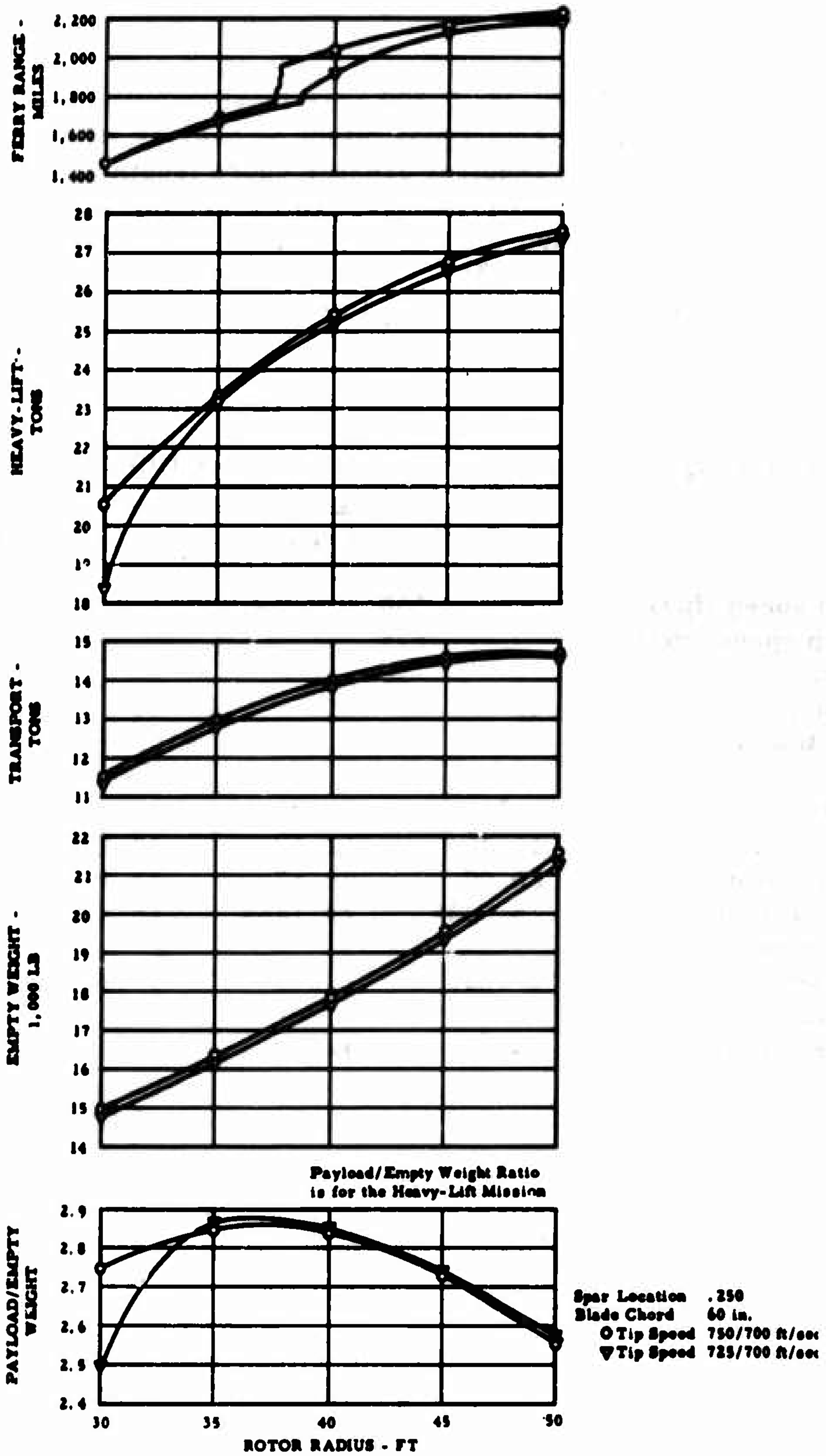


Figure 44. Influence of Rotor Radius and Tip Speed - Configuration 2.

## Effect of Spar Position

For the figure-8 duct blades, the 0. 300 spar location will permit greater duct area, whose benefit is sometimes offset by heavier structure. The 0. 250 spar location usually requires greater chord length to maintain duct area, which also leads to heavier blades. However, for smaller rotor radii, the 0. 250 spar location will result in an overall better payload/ empty weight ratio and range. This is indicated in Table XVIII, which shows the best rotors of configurations 1 and 2 for the 50- and 35-foot radii and both spar locations.

TABLE XVIII

EFFECT OF SPAR LOCATION ON FIGURE-8 DUCT BLADES

| | Configuration 1 | | Configuration 2 | |
|---|---|---|---|---|
| Hover tip speed (fps) | 750 | | 750 | |
| Cruise tip speed (fps) | 675 | | 700 | |
| Radius (ft) | 50 | | 35 | |
| Chord (in. ) | 65 | 55 | 60 | 55 |
| Spar location | 0. 250 | 0. 300 | 0. 250 | 0. 300 |
| Mission | | | | |
| Ferry (nmi) | 2, 015 | 1, 965 | 1, 688 | 1, 517 |
| Transport (tons) | 11. 46 | 12. 17 | 12. 17 | 13. 06 |
| Heavy-lift (tons) | 26. 00 | 26. 67 | 23. 30 | 22. 87 |
| Empty weight (lb) | 25, 523* | 24, 146* | 16, 349 | 16, 266 |
| Payload/empty weight | | | | |
| Transport | 0. 7687 | 0. 8559 | 1. 5863 | 1. 6058 |
| Heavy-lift | 2. 0377 | 2. 2088 | 2. 8503 | 2. 8121 |
| Computer run number | 1-22 | 1-29 | 2-13 (Sheet 1 of 2) | 2-14 (Sheet 1 of 2) |

*Empty weight for heavy-lift mission.

## Effect of Fixed Versus Retracted Landing Gear

All configurations were programmed both with a fixed and with a retracted landing gear. The lighter fixed gear proved to be more efficient. A typical example is shown in Table XIX.

## TABLE XIX
## EFFECT OF FIXED VERSUS RETRACTED LANDING GEAR

| | Landing Gear | |
|---|---|---|
| | Retracted | Fixed |
| Mission | | |
| Ferry (nmi) | 1,762 | 1,772 |
| Transport (tons) | 13.81 | 13.99 |
| Heavy-lift (tons) | 25.91 | 26.11 |
| Empty weight (lb) | 19,308 | 18,881 |
| Payload/empty weight | | |
| Transport | 1.4306 | 1.4818 |
| Heavy-lift | 2.6839 | 2.7657 |
| Computer run number (configuration 2, 45-foot radius, 55-inch chord, $V_t$ hover = 750 fps, $V_t$ cruise = 700 fps) | 2-2 | 2-3 |

### Effect of Thickness Ratio

Blade airfoil thickness ratios of the following combinations were programmed, and the results are shown in Table XX, which shows the thinner blade section to give superior performance.

| Inboard 75 Percent Span | Outboard 25 Percent Span |
|---|---|
| 18% | 14% |
| 18% | 16% |
| 18% | 18% |

### Effect of Engine Installation

Two General Electric GE1/J1 engines were considered as the primary power source. An alternate engine arrangement utilizing four General Electric T64/S4B gas generators was also surveyed. See Table XXI.

### Effect of Four Blades

A check was made on the effect of using four blades instead of three. Though the four-bladed configuration showed promise of having adequate performance, it was abandoned as a result of the added difficulties of routing the gas through the hub and into four blades because of space limitation in the hub (reference run 2-7).

## TABLE XX
## EFFECT OF BLADE THICKNESS

| | Airfoil Thickness (% of Chord) | | |
|---|---|---|---|
| Inboard 0.75 R | 18 | 18 | 18 |
| Outboard 0.25 R | 18 | 16 | 14 |
| Mission | | | |
| Ferry (nmi) | 1,339 | 1,376 | 1,661 |
| Transport payload (tons) | 12.25 | 12.61 | 12.80 |
| Heavy-lift payload (tons) | 20.43 | 22.82 | 23.13 |
| Empty weight (lb) | 16,178 | 16,185 | 16,174 |
| Payload/empty weight | | | |
| Transport | 1.5141 | 1.5580 | 1.5832 |
| Heavy-lift | 2.5259 | 2.8196 | 2.8602 |
| Computer run number | 2-22 | 2-12 | 2-13 |

## TABLE XXI
## EFFECT OF ENGINE INSTALLATION

| | Engine Installation | |
|---|---|---|
| | GE1/J-1 (2) | T-64/S4B (4) |
| Mission | | |
| Ferry (nmi) | 2,038 | 1,782 |
| Transport (tons) | 13.97 | 10.16 |
| Heavy-lift (tons) | 25.39 | 21.26 |
| Empty weight (lb) | 17,832 | 18,297 |
| Payload/empty weight | | |
| Transport | 1.5665 | 1.0103 |
| Heavy-lift | 2.8480 | 2.1154 |
| Computer run number (configuration 2, 40-foot radius, 60-inch chord, $V_t$ hover = 750 fps, $V_t$ cruise = 700 fps) | 2-13 | 2-11 |

## Effect of Drag

To evaluate the effect of increased drag on the mission performance, two cases were run on configuration 2 -- one doubling the estimated drag of the helicopter and the other doubling the estimated drag of the external payloads. The results, presented in Table XXII, show the increased fuselage drag to have little effect on any mission except ferry range and the increased payload drag to have a small effect on transport mission payload.

TABLE XXII
EFFECT OF DRAG ON PERFORMANCE

| | Estimated Fuselage Payload Drag | Double Estimated Payload Drag | Double Estimated Fuselage Drag |
|---|---|---|---|
| Mission | | | |
| Ferry (nmi) | 1,587 | 1,587 | 1,443 |
| Transport (tons) | 13.54 | 12.87 | 13.15 |
| Heavy-lift (tons) | 24.55 | 24.31 | 24.48 |
| Empty weight (lb) | 17,364 | 17,364 | 17,364 |
| Payload/empty weight | | | |
| Transport | 1.5591 | 1.4826 | 1.5145 |
| Heavy-lift | 2.8277 | 2.8000 | 2.8200 |
| Computer run number (configuration 2, 40-foot radius, 55-inch chord, $V_t$ hover = 750 fps, $V_t$ cruise = 700 fps) | 2-3 | 2-5 | 2-6 |

## SPECIFIC RANGE

Figures 45 and 46 show the influence of flight speed on the specific range. It can be seen that at the speeds specified by the requirements, the curves show optimum values for the specific range.

## FUEL REQUIRED FOR TRANSPORT AND HEAVY-LIFT MISSIONS

To show the transportation performance achieved by a certain amount of fuel consumed, the payloads in ton-miles per pound of fuel were calculated for the various configurations and missions. These figures, as opposed to fuel flow per hour or miles per pound of fuel, are of major importance for estimating actual costs and logistics of helicopter operations, and are shown in Tables XXIII and XXIV for the transport and heavy-lift missions. Results for an operational helicopter (CH-47A) have been included in Table XXIV for comparison.

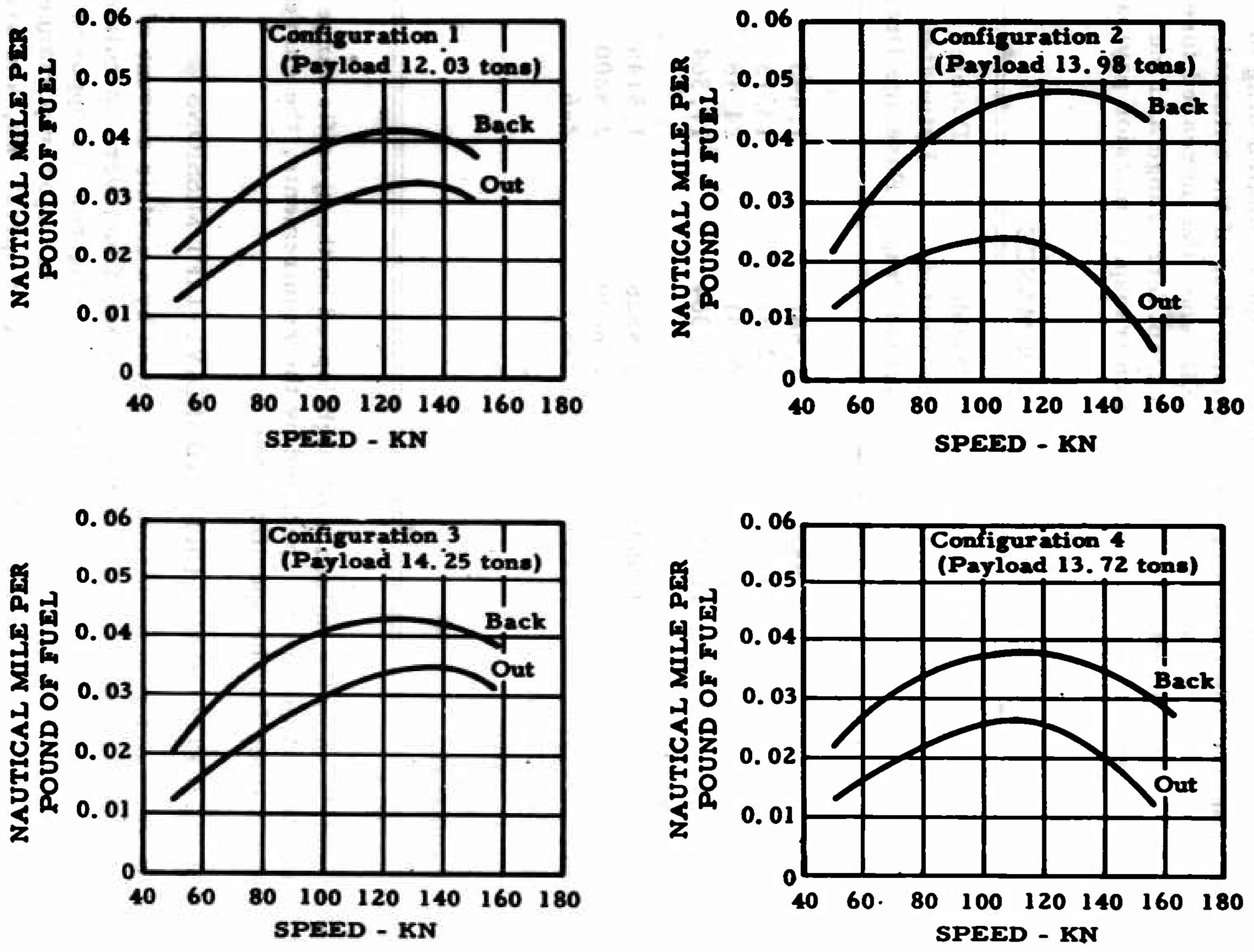


Figure 45. Effect of Speed on Specific Range Based on Average Gross Weight Out and Average Gross Weight Back - Transport Mission.

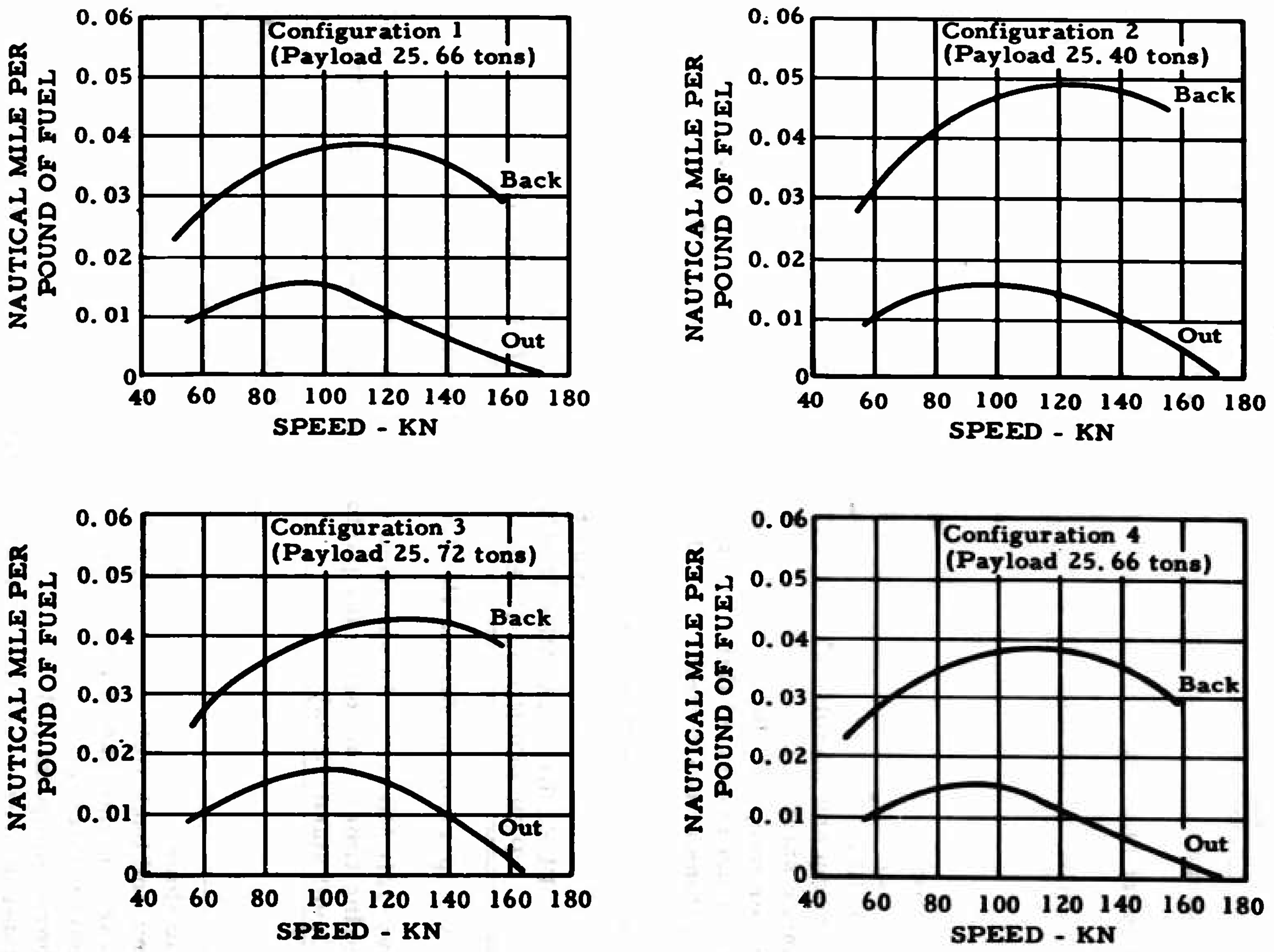


Figure 46. Effect of Speed on Specific Range Based on Average Gross Weight Out and Average Gross Weight Back - Heavy-Lift Mission.

## TABLE XXIII
## FUEL REQUIREMENTS AND PAYLOAD TON-MILES PER POUND OF FUEL - TRANSPORT MISSION

**Comparison of Various Hot Cycle Helicopter Configurations**

Transport Mission: As specified in requirements, but no warmup and hovering times considered for payload ton-miles/pound fuel numbers. Hover OGE, 6,000 feet, 95°F.

| | Configuration | | | |
|---|---|---|---|---|
| | 1 | 2 | 3 | 4 |
| Payload (tons) | 12.03 | 13.98 | 14.08 | 13.72 |
| Fuel out (lb) | 3,050 | 4,142 | 2,897 | 3,840 |
| Fuel back (lb) | 2,429 | 2,079 | 2,350 | 2,714 |
| Fuel warmup and takeoff (lb) | 500 | 250 | 500 | 250 |
| Fuel hover start (lb) | 308 | 307 | 308 | 308 |
| Fuel hover midpoint (lb) | 186 | 181 | 187 | 183 |
| Fuel total (no reserves) (lb) | 6,473 | 6,959 | 6,242 | 7,295 |
| Payload (ton-miles/lb fuel) | 0.220 | 0.223 | 0.268 | 0.210 |

## TABLE XXIV
## FUEL REQUIREMENTS AND PAYLOAD TON-MILES PER POUND OF FUEL - HEAVY-LIFT MISSION

**Comparison of Various Hot Cycle Helicopter Configurations**

Heavy-Lift Mission: As specified in requirements, but no warmup and hovering times considered for payload ton-miles/pound fuel numbers. Sea level standard day.

| | Configuration | | | | CH-47A |
|---|---|---|---|---|---|
| | 1 | 2 | 3 | 4 | (Ref 9) |
| Payload (tons) | 25.66 | 25.40 | 25.55 | 25.66 | 7.01 |
| Fuel out and back (lb) | 1,825 | 1,650 | 1,669 | 1,825 | 1,170 |
| Fuel warmup and takeoff (lb) | 250 | 250 | 250 | 250 | - |
| Fuel hover start (lb) | 776 | 775 | 775 | 776 | - |
| Fuel hover midpoint (lb) | 1,406 | 1,422 | 1,426 | 1,406 | - |
| Fuel total (no reserves) (lb) | 4,257 | 4,097 | 4,120 | 4,257 | 1,170 |
| Payload (ton-miles/lb fuel) | 0.281 | 0.308 | 0.306 | 0.281 | 0.120 |

## RESULTS OF AUTOROTATION REQUIREMENT STUDY

A spot check was made on several of the configurations studied to estimate the autorotational performance. While autorotation was found to be noncritical on the optimum rotors for configurations 1 and 4, the rotor diameters of configurations 2 and 3 were in fact determined by the autorotational rather than the mission requirements.

Using a method outlined in Appendix III of Reference 2, an index number K was calculated that represents a kinetic energy ratio.

$$K = \frac{\text{usable rotor kinetic energy}}{\text{helicopter sink-rate kinetic energy}}$$

To permit a quick check of the autorotational capability, the following assumptions were made.

1. Safe autorotation is required with all engines failed and at a gross weight associated with the internal payload shown below:

| Configuration | Internal Payload (tons) |
|---|---|
| 1 | 12 (in pod) |
| 2 | 7 |
| 3 | 12 |

2. An external payload would be jettisoned in case of emergency.

Table XXV shows the results of this check for configurations 1, 2, and 3 with various rotor radii and gross weights. The comparable index number for some operational helicopters was approximated and added for comparison. It can be seen that the index numbers of the selected rotor radii of 45, 40, and 45 feet for configurations 1, 2, and 3, respectively, fall within the range of the operational helicopter index numbers.

## TABLE XXV
## AUTOROTATION - ROTOR KINETIC ENERGY INDEX

| | Helicopters in Operation | | | | | | Hot Cycle Heavy-Lift | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | 269A 2,900 rpm | Hughes OH-6A | Bell UH-1B | Sikorsky H-19 | Sikorsky H-34 | Sikorsky S-64 | Configuration 1 R = 45 ft | Configuration 1 R = 50 ft | Configuration 1 R = 55 ft | Configuration 2 5-ton payload R = 35 ft | Configuration 2 5-ton payload R = 40 ft | Configuration 2 7-ton payload R = 35 ft | Configuration 2 7-ton payload R = 40 ft | Configuration 3 R = 40 ft | Configuration 3 R = 45 ft |
| Gross weight (lb) | 1,670 | 2,400 | 6,600 | 7,100 | 11,900 | 38,000 | 50,600 | 61,060 | 64,744 | 33,940 | 35,617 | 40,000 | 40,000 | 54,100 | 58,509 |
| I - rotor (slug ft$^2$) | 131 | 215 | 1,920 | 2,060 | 4,490 | 21,000 | 59,793 | 86,975 | 124,251 | 22,290 | 35,270 | 22,290 | 35,270 | 35,270 | 59,793 |
| b - number of blades | 3 | 4 | 2 | 3 | 4 | 6 | 3 | 3 | 3 | 3 | 3 | 3 | 3 | 3 | 3 |
| c - chord (ft) | 0.562 | 0.562 | 1.75 | 1.366 | 1.366 | 1.971 | 5.0 | 5.0 | 5.0 | 5.0 | 5.0 | 5.0 | 5.0 | 5.0 | 5.0 |
| R - radius (ft) | 12.5 | 13.167 | 22.0 | 26.5 | 28.0 | 36.0 | 45.0 | 50.0 | 55.0 | 35.0 | 40.0 | 35.0 | 40.0 | 40.0 | 45.0 |
| $A_b$ - blade area (sq ft) | 21.1 | 29.63 | 77.0 | 108.6 | 153 | 426 | 675 | 750 | 825 | 525 | 600 | 525 | 600 | 600 | 675 |
| A - disc area (sq ft) | 491 | 544.6 | 1,520.5 | 2,206 | 2,463 | 4,071.5 | 6,362 | 7,854 | 9,503 | 3,848.5 | 5,027 | 3,848.5 | 5,027 | 5,027 | 6,362 |
| DL - disc loading (lb/sq ft) | 3.40 | 4.41 | 4.34 | 3.22 | 4.83 | 9.33 | 9.20 | 11.07 | 6.81 | 8.82 | 7.09 | 10.40 | 7.96 | 10.76 | 9.20 |
| $\Omega$ - angular velocity (rad/sec) | 50.6 | 49.2 | 32.9 | 22.0 | 23.2 | 19.3 | 15.6/16.7 | 14.0/15.0 | 12.7/13.6 | 20.0/21.4 | 17.5/18.8 | 20.0/21.4 | 17.5/18.8 | 17.5/18.8 | 15.6/16.7 |
| $V_T$ - tip speed (fps) | 632 | 640 | 724 | 583 | 650 | 696 | 700/750 | 700/750 | 700/750 | 700/750 | 700/750 | 700/750 | 700/750 | 700/750 | 700/750 |
| $A_\pi$ - parasite area (sq ft) | 14.0 | 5.0 | 22.8 | 40.0 | 36.5 | 120.0 | 34.5 | 34.5 | 34.5 | 23.1 | 23.1 | 23.1 | 23.1 | 25.9 | 25.9 |
| V for minimum sink rate (kn) | 41.4 | 62.6 | 54.0 | 45.0 | 50.0 | 60.0 | 103.3 | 100.7 | 90.2 | 98.5 | 96.4 | 106.9 | 100.0 | 113.1 | 110.9 |
| $K_{max}$ - kinetic energy index | 3.20 | 3.90 | 4.81 | 3.95 | 3.26 | 1.81 | 3.49/3.72 | 4.40/4.56 | 5.47/5.49 | 3.55/3.45 | 4.46/4.17 | 2.[illegible]/3.00 | 4.06/3.93 | 2.16/2.41 | 3.73/3.96 |

## COMPOUND HELICOPTER STUDY

The study of the compound helicopter was undertaken on a limited basis to identify the compromises in weight, size, complexity, and performance required to attain a substantial increase in cruise speed. For this study, the configuration 3 helicopter (conventional fuselage) was compounded and redesignated configuration 5. Compounding was accomplished by the addition of wings and ducted fans for thrust. The resulting increased structural and system requirements were also incorporated into the basic helicopter configuration.

The missions selected to be studied for the compound were transport missions of 200-, 300-, and 500-nautical-mile radii and ferry range. These missions were considered to be run at both sea level and optimum altitude. Three takeoff conditions were studied: hover at sea level standard conditions, hover at 6,000 feet and 95°F, and STOL operation.

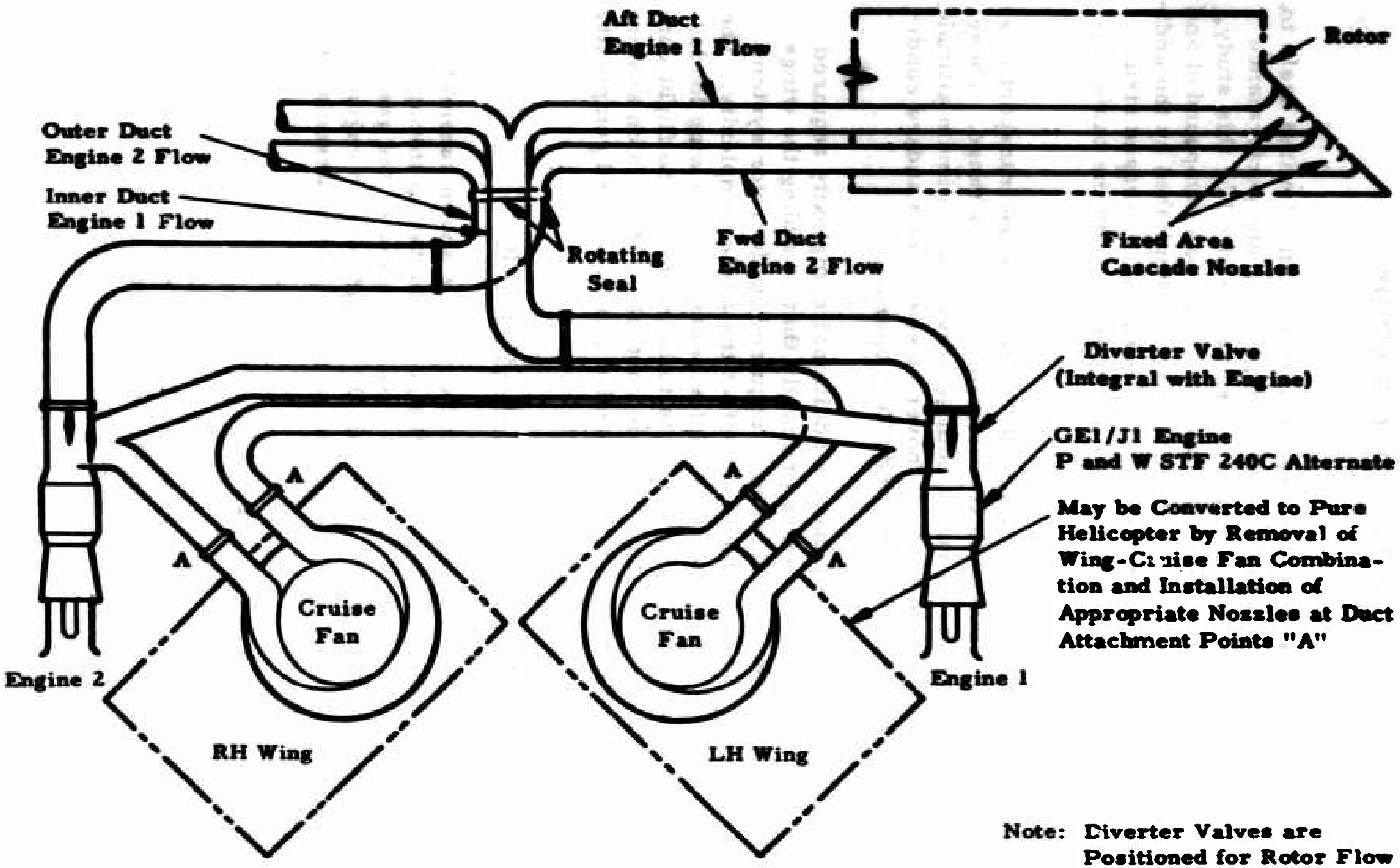
In this study, it was determined that the additional complexity required by the compound may be identified as primarily that of adding the wings and thrust fans. Some weight increase is required in the rotor system for dynamic reasons, but this should not affect the blade complexity. The fuselage, of course, is slightly more complex because of the wing loads and increased tail loads and additional ducts to be routed. The flight controls are modified by the addition of aileron controls. Propulsion system controls, valves, and ducting must be expanded in number and complexity for the compound version.

## PROPULSION SYSTEM FOR COMPOUND HELICOPTER

The propulsion system for the compound helicopter is shown schematically in Figure 47. For helicopter operation, the gas is ducted up through the rotor in the normal manner. For operation as a compound, the gas is diverted to drive the ducted fans. One-half the output of each engine is routed to each of the ducted fans, to minimize the problems associated with single-engine operation.

## WEIGHTS FOR COMPOUND HELICOPTER

The consideration of compound helicopter operation required modification of some group weight constants and equations and addition of new expressions reflecting this conversion, as discussed in the Weight section of this report.



Note: Diverter Valves are Positioned for Rotor Flow

Figure 47. Propulsion System Schematic - Two-Engine Compound Helicopter Installation.

A comparison of the empty weights of the compound version, the pure helicopter, and the same helicopters with provisions to be converted to a compound is shown in Table XXVI for a typical case. This shows the compound helicopter to have an empty weight of 7,543 pounds more than the pure helicopter and 2,621 pounds more than the helicopter having provisions for compounding.

TABLE XXVI

EMPTY WEIGHT SUMMARY - HELICOPTER, COMPOUND HELICOPTER, AND HELICOPTER HAVING PROVISIONS FOR COMPOUNDING

| | Configuration 3 Helicopter | Configuration 5 Helicopter | Configuration 5 Compound |
|---|---|---|---|
| Fixed provisions for compounding | | | |
| Rotor | - | 905 | 905 |
| Structure, controls, ducting, etc | - | 1,716 | 1,716 |
| Removable provisions for compounding | | | |
| Wings, fans, etc | - | - | 4,922 |
| Empty weight | 21,080 | 23,701 | 28,623 |

## PERFORMANCE OF COMPOUND

Several aspect ratios and wing spans as installed on the configuration 5 compound were included in the parametric study. Table XXVII shows the effect of varying aspect ratio and wing span at both maximum continuous power and power for best range for a 200-, 300-, and 500-nautical-mile mission.

The ferry weight and empty weight are also noted. Figure 48 plots the payload versus mission radius of the compound for various hovering capabilities using an aspect ratio of 10 and a wing span of 65 feet. Also shown in this figure is the estimated curve for the pure helicopter (configuration 3) performing the optimum altitude mission with a 6,000-foot 95°F and with a sea-level standard-day hovering capability. The ferry range for the compound is 2,886 nautical miles, compared with 2,040 nautical miles for the configuration 3 helicopter.

## TABLE XXVII
## SUMMARY - PAYLOAD AND FERRY RANGE
### (At Altitude for Best Range)

| Wing Span (ft) | Aspect Ratio | Payload (Tons) for Mission Radii of 200, 300, and 500 nmi | | | | | | | | | Ferry Range (nmi) | Empty Weight (lb) |
|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | Hover - 6,000 ft 95° F | | | Hover SL Std | | | STOL | | | | |
| | | 200 | 300 | 500 | 200 | 300 | 500 | 200 | 300 | 500 | | |
| **Maximum Continuous Power** | | | | | | | | | | | | |
| 65 | 8 | 8.14 | 6.42 | 3.00 | 16.74 | 14.30 | 9.46 | 17.01 | 14.55 | 9.73 | 2,226 | 28,315 |
| 75 | 8 | 7.97 | 6.38 | 3.21 | 16.63 | 14.65 | 10.51 | 17.66 | 15.35 | 11.47 | 2,359 | 28,708 |
| 85 | 8 | 7.62 | 6.02 | 2.88 | 16.44 | 14.33 | 10.15 | 19.59 | 17.25 | 13.34 | 2,523 | 29,062 |
| 55 | 10 | 8.09 | 6.06 | 2.67 | 16.30 | 13.29 | 8.11 | 16.13 | 13.13 | 7.95 | 2,135 | 28,257 |
| 65 | 10 | 8.15 | 6.42 | 3.05 | 16.81 | 14.38 | 9.58 | 17.03 | 14.59 | 9.79 | 2,886 | 28,669 |
| 75 | 10 | 7.99 | 6.40 | 3.28 | 16.71 | 14.77 | 10.63 | 17.21 | 15.23 | 11.11 | 2,358 | 29,079 |
| 85 | 10 | 7.65 | 6.08 | 2.96 | 16.55 | 14.45 | 10.32 | 18.01 | 15.89 | 11.72 | 2,756 | 29,485 |
| 55 | 12 | 8.04 | 6.04 | 2.64 | 16.30 | 13.29 | 8.12 | 16.09 | 13.07 | 7.90 | 2,146 | 28,515 |
| 65 | 12 | 8.10 | 6.40 | 3.02 | 16.81 | 14.38 | 10.23 | 17.01 | 14.57 | 9.81 | 2,255 | 28,976 |
| 75 | 12 | 7.95 | 6.37 | 3.27 | 16.71 | 14.78 | 10.67 | 16.70 | 14.77 | 10.65 | 2,830 | 29,435 |
| 85 | 12 | 7.60 | 6.03 | 2.91 | 16.72 | 14.78 | 10.95 | 17.03 | 15.08 | 11.28 | 2,339 | 29,892 |
| **Power for Best Range** | | | | | | | | | | | | |
| 55 | 6 | 8.14 | 6.24 | 2.92 | 16.29 | 13.42 | 8.48 | 16.48 | 13.61 | 8.64 | 2,252 | 27,708 |
| 65 | 6 | 8.04 | 6.43 | 3.27 | 16.61 | 14.23 | 9.70 | 17.59 | 15.28 | 10.68 | 2,416 | 28,011 |
| 75 | 6 | 7.84 | 6.20 | 3.06 | 16.47 | 14.24 | 10.42 | 19.74 | 17.42 | 12.74 | 2,594 | 28,310 |
| 55 | 8 | 8.22 | 6.34 | 3.05 | 16.46 | 13.63 | 8.73 | 16.56 | 13.72 | 8.81 | 2,286 | 27.991 |
| 65 | 8 | 8.21 | 6.57 | 3.46 | 16.38 | 14.48 | 10.03 | 17.34 | 14.97 | 10.48 | 2,436 | 28,315 |
| 75 | 8 | 8.05 | 6.53 | 3.63 | 16.75 | 14.57 | 10.82 | 18.00 | 15.79 | 11.95 | 2,582 | 28,708 |
| 55 | 10 | 8.22 | 6.35 | 3.08 | 16.51 | 13.70 | 8.83 | 16.57 | 13.75 | 8.88 | 2,302 | 28,257 |
| 65 | 10 | 8.22 | 6.59 | 3.52 | 16.90 | 14.57 | 10.17 | 16.90 | 14.58 | 10.17 | 2,431 | 28,669 |
| 55 | 12 | 8.18 | 6.32 | 3.06 | 16.51 | 13.71 | 8.86 | 16.53 | 13.73 | 8.88 | 2,310 | 28,515 |
| 65 | 12 | 8.18 | 6.56 | 3.51 | 16.90 | 14.59 | 10.21 | 16.85 | 14.54 | 10.16 | 2,444 | 28,976 |

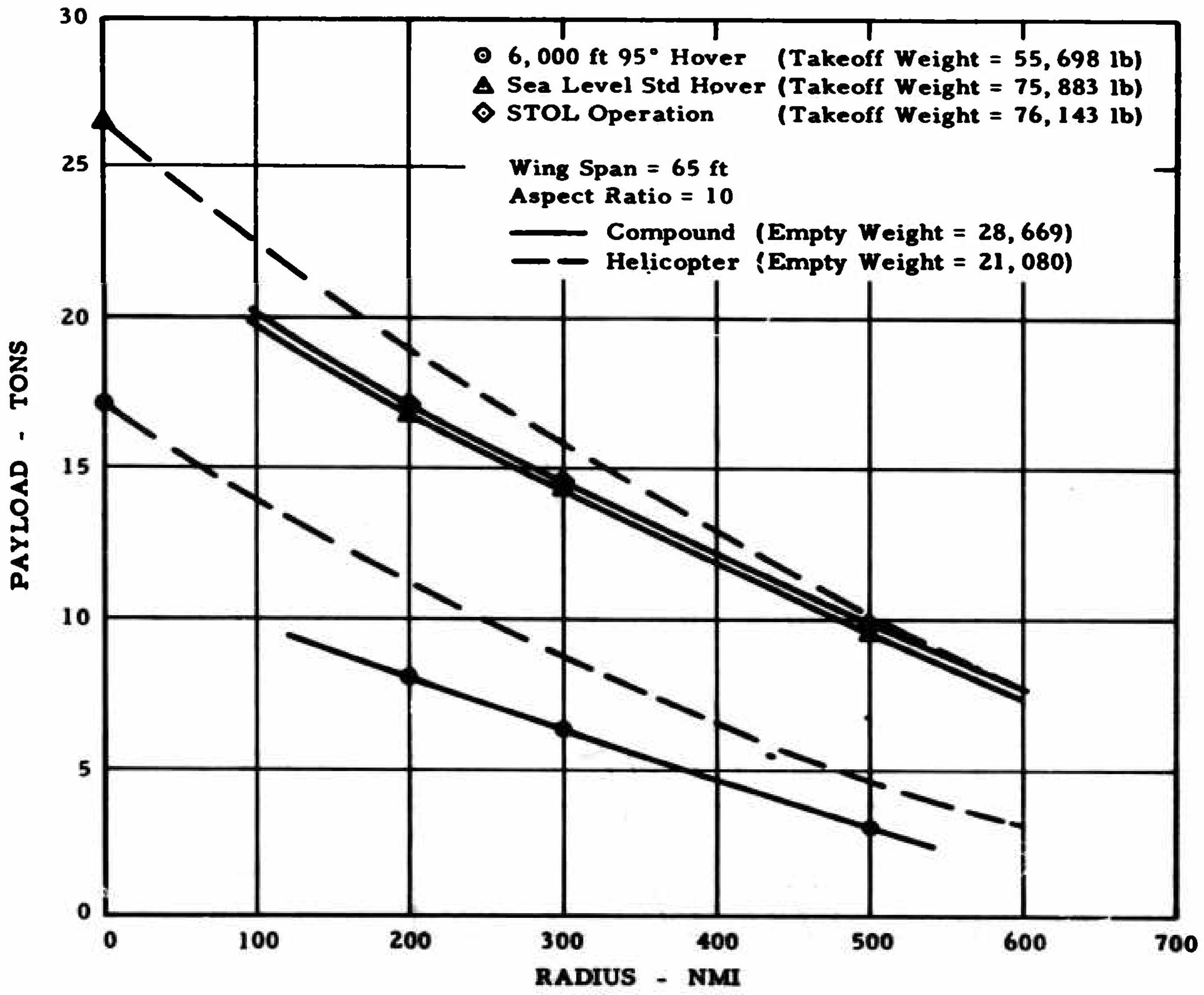


Figure 48. Payload Versus Mission Radius for Compound Helicopter.

The average cruise speed for the transport mission is approximately 225 knots at power for best range, 255 knots for maximum continuous power as a compound, and approximately 110 knots as a helicopter. A productivity parameter that takes this speed difference into account may be expressed as follows, and is shown plotted against range in Figure 49.

$$\text{Productivity} = \frac{\text{payload (lb) x speed (kn)}}{\text{weight empty (lb)}}$$

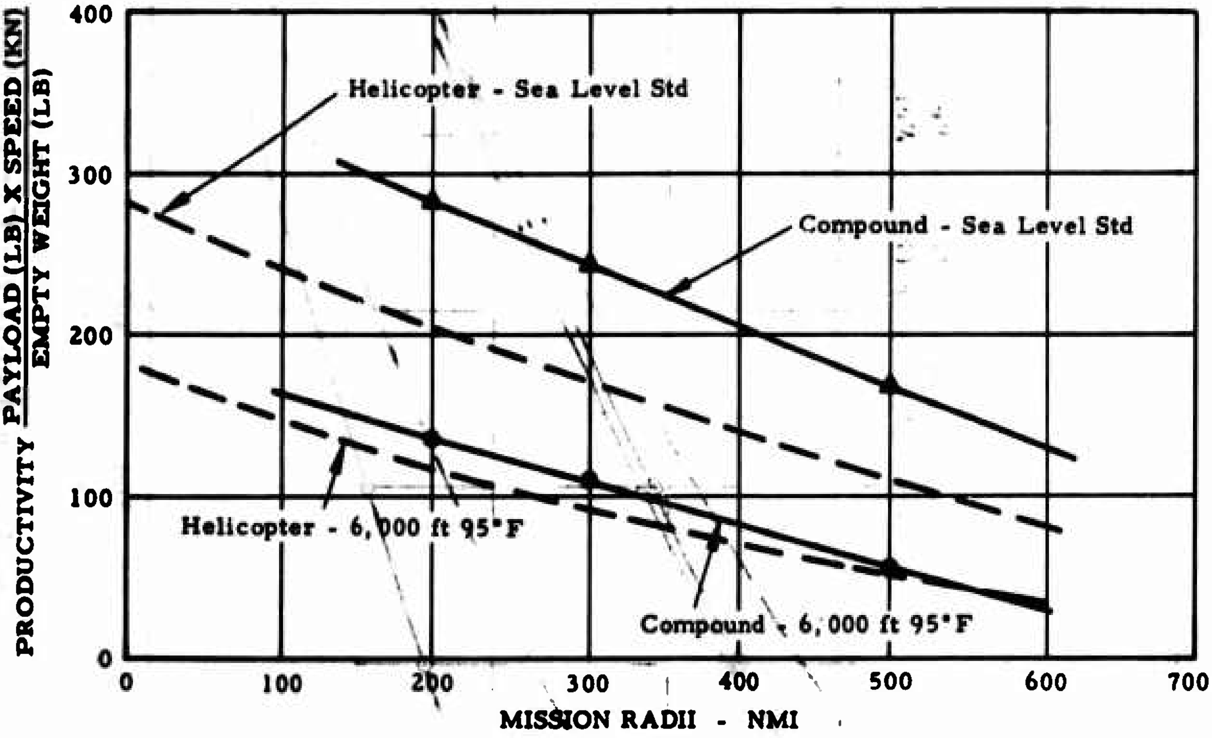


Figure 49. Productivity - Compound Helicopter.

# FULLY COUPLED BLADE RESPONSE AND DYNAMIC STABILITY ANALYSIS USING SADSAM IV

## INTRODUCTION

SADSAM IV is a digital computer program that was developed by the MacNeal-Schwendler Corporation under contract to the Hughes Tool Company. The development of this program has been summarized and previously submitted in Reference 1.

The program can be applied to the full range of helicopter dynamic problems, including fuselage vibration analysis and all types of rotor dynamic analysis. A nonlinear representation of blade air loads, including lift and moment hysteresis, is incorporated in the program to provide capability for fully coupled blade loads analysis in forward flight.

Problem formulation is generalized to permit application to any structural configuration. The structure is described by means of lumped elements. Problem size is limited to maximize computing efficiency by ensuring that most mathematical operations are accomplished using only high-speed core storage.

## PROGRAM CAPABILITY

### PROBLEM TYPES

The program is designed to treat structural dynamics problems in which the structure is described by lumped linear elements (springs, masses, dampers, and leverage devices). The user of the program specifies the manner in which the elements are connected. The program is, therefore, applicable to any structural configuration, including, for example, bridges, buildings, fixed-wing aircraft, and helicopter rotors.

In addition, a stripwise formulation of subsonic aerodynamic theory is incorporated into the program for the specific purpose of simulating, when required, the air loads on a rotor blade in hovering or in forward flight.

The following types of mathematical analysis can be performed with the program:

1. Determine vibration modes of an undamped, linear, conservative structure.

2. Determine the complex eigenvalues (or roots of the stability equation) of a damped, linear, unconservative system. The primary application of this provision is flutter analysis.
3. Determine the response of a damped or undamped, conservative or unconservative, linear system to sinusoidal excitation at a sequence of discrete frequencies.
4. Determine the response of a damped or undamped, conservative or unconservative, linear or nonlinear system to transient excitation with prescribed time history.

## PROBLEM SIZE

The maximum number of degrees of freedom is 50. The maximum number of elements in each class (springs, masses, dampers, and leverage devices) is 99. These limitations are translated below into the maximum number of spanwise stations for various idealizations of a rotor blade.

| | Rotor Blade Idealization | Number of Stations |
|---|---|---|
| 1. | Flapwise bending only | 49 |
| 2. | Flapwise bending and twist | 25 |
| 3. | Flapwise and chordwise bending | 24 |
| 4. | Flapwise bending, chordwise bending, and twist (fully coupled) | 16 |
| 5. | Same as (4) but including chordwise shear flexibility (thereby making chordwise bending slope an independent degree of freedom) | 12 |

## MATHEMATICAL METHODS

### REDUCTION OF PROBLEM TO MATRIX FORM

The first step performed by the computer in the solution of any problem is to reduce the problem to the following matrix form:

$$\left[Mp^2 + Bp + K\right] \{x\} = \{F\} \tag{47}$$

where

$p = \frac{d}{dt}$

$[M]$ = mass matrix

$[B]$ = damping matrix

$[K]$ = stiffness matrix

$\{x\}$ = vector of independent displacements

$\{F\}$ = vector of applied forces including, for transient analyses only, nonlinear functions of the independent displacements

Because of the presence of leverage devices that impose constraints on components of displacement, the total number of "node" points to which elements are connected exceeds the number of independent displacements. The computer program senses this fact, selects an independent set of displacements, and refers all mass, stiffness, and damping properties to that set. The method used is substantially the same as that described in Reference 21.

## DETERMINATION OF FREQUENCY RESPONSE

Frequency response is obtained by replacing p by $i\omega$, where $\omega$ is a specified real number in equation (47) and solving for $\{x\}$ in terms of a given $\{F\}$. $\{F\}$ may have components. The user has the option of specifying a level of structural damping by substituting $(1+ig) \cdot [K]$ for $[K]$. An efficient method of triangular resolution, Reference 22, is employed in solving for $\{x\}$.

## EXTRACTION OF EIGENVALUES AND EIGENVECTORS

Eigenvalues and eigenvectors for both damped and undamped systems are obtained by a special algorithm developed by the MacNeal-Schwendler Corporation. The basis of the algorithm is that if $\{F\}$ in equation (47) is a specified vector and if p is approximately equal to an eigenvalue $(p = r_k + \epsilon)$, then all components of $\{x\}$ will be large. In fact, in the neighborhood of the $k^{th}$ eigenvalue, any particular component of $\{x\}$ is approximated by

$$x_j = \frac{A_{jk}}{p - r_k} + C_{jk} \qquad (48)$$

where $r_k$, $A_{jk}$, and $C_{jk}$ are constants.

The algorithm essentially consists of the following steps:

1. Evaluate $x_j$ for three trial values of p by solving equation (47) with $p = p_1, p_2, p_3$.
2. Solve for $A_{jk}$, $C_{jk}$, and $r_k$ using equation (48).
3. Replace one of the $p_i$'s by the value $r_k$ estimated from step (2). Replace the $p_i$ that is farthest from $r_k$.
4. Repeat steps (1), (2), and (3) to convergence. In trial applications, convergence to six significant figures is obtained in approximately six iterations.

The algorithm is provided with means for sweeping previously found eigenvalues and for testing convergence. All roots within a frequency band specified by the user will be found.

## EVALUATION OF TRANSIENT RESPONSE

Transient response is evaluated by direct numerical integration of the equations of motion rather than by modal decomposition. The integration algorithm has been carefully selected to avoid numerical instability while maintaining accuracy. It therefore permits the use of relatively large time steps. The algorithm is described in Reference 23.

## TREATMENT OF ROTOR BLADE AERODYNAMICS

Strip theory is used; that is, the aerodynamic forces at a given station are calculated using the translations and rotations at that station. A linear, incompressible formulation is used for flutter analysis. The linear formulation, which is essentially identical with that presented in Reference 24, is summarized by the following equations for the forces and moments acting on a strip. The formulation includes mechanical Coriolis effects.

$$P_Z = \Delta r \left\{ \pi\rho(\Omega r)^2 c \left[ \alpha_{\frac{3}{4}} + \frac{c}{4\Omega r} (\dot{\theta} + \Omega\beta) \right] \right\} \tag{49}$$

$$P_x = \gamma_0 P_Z - \Delta r \, m\Omega \sin a \cdot \dot{Z} \tag{50}$$

$$M = \left( x_{ref} - \frac{c}{4} \right) P_Z - \left[ \frac{\pi\rho}{8} \Omega r c^3 (\dot{\theta} + \Omega\beta) \right] \Delta r \tag{51}$$

$$\alpha_{\frac{3}{4}} + \frac{c}{4\Omega r} (\dot{\theta} + \Omega\beta) = \dot{\theta} - \frac{1}{\Omega r} \left[ \dot{Z} + \gamma_o \dot{x} + (c - x_{ref}) (\dot{\theta} + \Omega\beta) \right] \tag{52}$$

where

$\dot{x}$ = in-plane component of velocity, tangent to cone of rotation, positive aft

$\dot{Z}$ = vertical component of velocity, normal to cone of rotation, positive up, on reference axis

$\theta$ = pitch angle about reference axis, positive leading edge up

$\beta$ = spanwise slope, positive tip up

$P_Z$ = vertical force, positive up

$P_x$ = in-plane force, positive aft

$M$ = moment about reference axis, positive leading edge up

$\alpha_{\frac{3}{4}}$ = angle of attack at three-fourths chord

$r$ = distance from axis of rotation

$\Delta r$ = spanwise width of strip

$\rho$ = air density

$\Omega$ = rotor speed (rad/sec)

$c$ = chord

$\gamma_0$ = inflow angle

$m$ = mass per unit length

$a$ = steady coning angle

$x_{ref}$ = distance from leading edge to reference axis

It will be noted that:

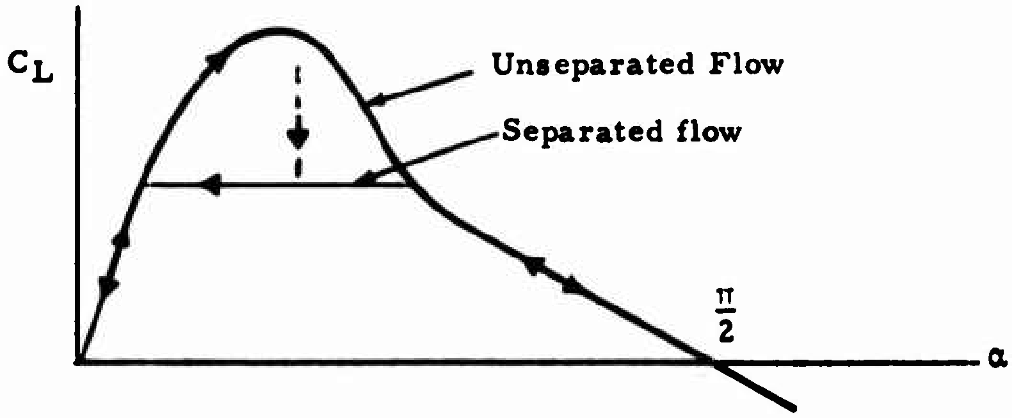
1. $C_{l\alpha}$ is assumed to be equal to $2\pi$.
2. The center of pressure is at the one-fourth chord.
3. Lift deficiency (Theodorsen's function) is ignored.
4. $\Omega\beta$ is added to $\theta$ to obtain true pitching velocity relative to the airstream.

5. The second term in equation (50) is all that remains of the Coriolis effect after cancellation with some small aerodynamics terms.
6. The inflow angle, $\gamma_0$, is assumed to be small.

In the nonlinear aerodynamic formulation, additional account is taken of the following:

1. The velocity of the air relative to the blade element can have any magnitude and any direction.
2. Lift, drag, and moment coefficients are nonlinear functions of Mach number and angle of attack.
3. The lift coefficient can be higher than the steady-state value in the stalled region because of rapid change of angle of attack (lift hysteresis).

Lift, drag, and moment coefficients are obtained from reported experiments on specific airfoil sections and are presented to the computer as tabular data. Lift hysteresis is accounted for in the manner shown in the sketch below.



For increasing angle of attack, the lift coefficient follows the upper curve, which is obtained either from experiments on oscillating airfoils or by reasonable extrapolation of the lift curve in the unstalled region. For decreasing angle of attack, or for any subsequent reversal of $\alpha$ in the stalled region, the lift coefficient follows the lower curve, which approximates the steady-state lift coefficient. The transition from the upper curve to the lower curve is abrupt. A separate pair of curves is used for each Mach number.

Moment hysteresis is accounted for by assuming that the center of pressure remains near the one-fourth chord point, as modified by Mach number effect, for unseparated flow, and that it shifts abruptly to the midchord when the flow separates. It is easily shown, incidentally, that energy is transferred from the airstream to the pitch degree of freedom during a hysteresis cycle.

## PROBLEM PREPARATION

The first step that must be performed for a new problem is to formulate a lumped mathematical model of the structure using springs, masses, dampers, and leverage devices as elements. The model formulation is most conveniently accomplished by using electric circuit modeling techniques that have been developed for passive analog computers. A complete account of such techniques is given in Reference 25. A detailed application to the dynamic analysis of rotor blades is described in Reference 24.

Once the model is formulated, the elements and the nodes (degrees of freedom) to which they are connected are numbered. Cards are then punched that record the nodes to which each element is connected and the numerical values of the elements. The constants that describe the aerodynamic force coefficients and the nodes on which they act are recorded on separate cards.

Problem input is completed by listing the constants that describe the applied forces and the points at which they act, and by listing the tasks to be performed (steady-state response at specified frequencies, vibration modes in a specified frequency range, and/or transient response in a specified time range).

The model formulation step need not be repeated for a problem that is topologically similar to a previous problem, because the cards that describe interconnection data can be saved. This feature is a great convenience for rotor blade analysis where geometrical configurations tend to be quite similar. Model reformulation will be required only if it is desired to change the basic assumptions (for example, eliminate twist as a degree of freedom), to change the number of spanwise stations, or to change the hub configuration. It should be noted that model reformulation is relatively easy with the program, because it can be accomplished by changes in input data rather than by changes in program instructions.

## CORRELATION OF COMPUTED LOADS AND FLIGHT TEST DATA

To verify the ability of the fully coupled rotor dynamic analysis to predict blade loads, a comparison was made of loads measured in high-speed cruise flight on the OH-6A helicopter and loads computed using SADSAM IV with an OH-6A blade model. The OH-6A was selected because it has the same hub configuration chosen for the heavy-lift helicopter; namely, a fully articulated hub with offset flapping hinges and lead-lag hinges with dampers.

For this analysis, a lift curve slope of 5.73/radian was used with a maximum $C_L$ of 1.6. The theoretical drag coefficient of the NACA polar was also used. As the load computation was made at 103 knots, the Mach number effects on lift and drag coefficients were neglected.

The satisfactory correlation of flapwise and chordwise bending moments as computed by this program and as measured in flight on the OH-6A helicopter is shown in Figures 50 and 51. It can be seen in Figure 50 that the flapwise moment at the critical root section is matched almost exactly, and in Figure 51 that the root chordwise moment is also in exceptionally good agreement. On the outboard part of the blade, the computed flapwise moment is somewhat higher than the measured moment, perhaps as a result of the simplified aerodynamic representation used in this example case; the correlation is considered satisfactory even in this region. The computed root torsional moment is ±124 in.-lb compared with ±115 in.-lb measured in flight, which is also acceptable verification of torsional loads.

## HEAVY-LIFT HELICOPTER BLADE AND HUB CONNECTION DIAGRAM

Having shown the ability of SADSAM IV to predict blade loads accurately, the same method of analysis was applied to the heavy-lift helicopter blade. Details of the blade and hub structure are given under the Structures section in this report. The hub is included in the problem because the root end fixity condition (hinged, cantilever, damper, and so forth) strongly influences blade loads at the root.

The blade was broken down into ten structural cells, each of the general arrangement shown in Appendix V. The blade and hub structural analog, called a connection diagram, is presented in Appendix IV. This connection diagram shows the masses, stiffness values, and lengths in the form of condensers, inductors, and transformers, as described in Appendix IV. Specific values of the elements are given in the diagram on page 347 (Appendix IV).

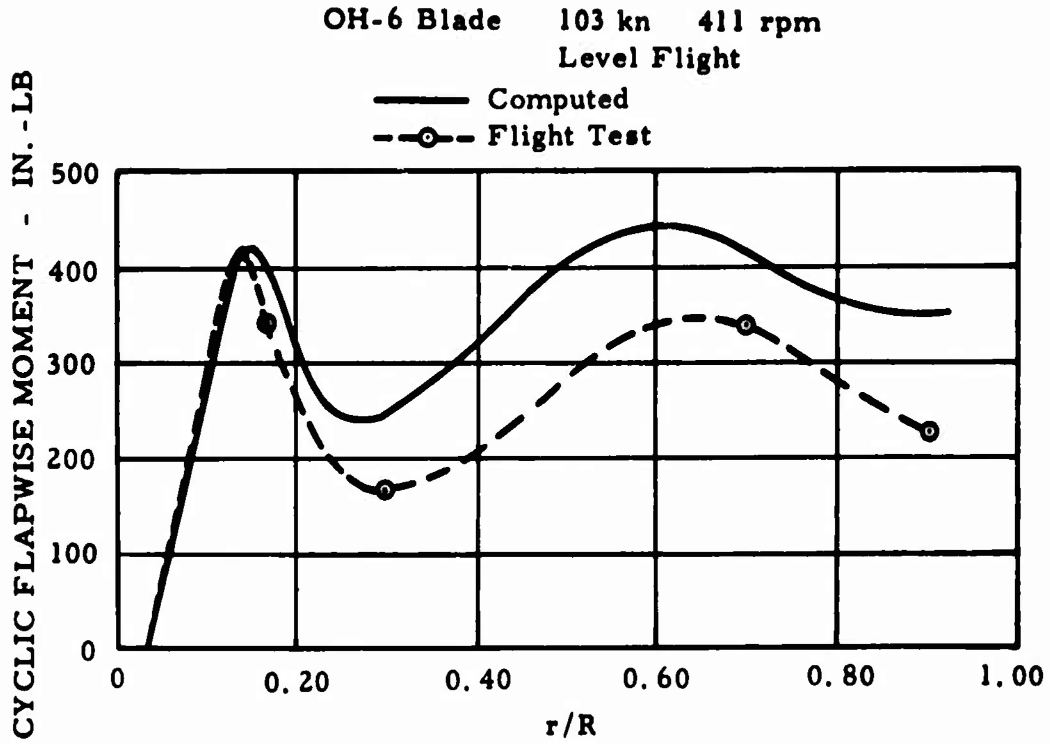


Figure 50. Flapwise Moment Distribution - Comparison of Theory and Flight Test.

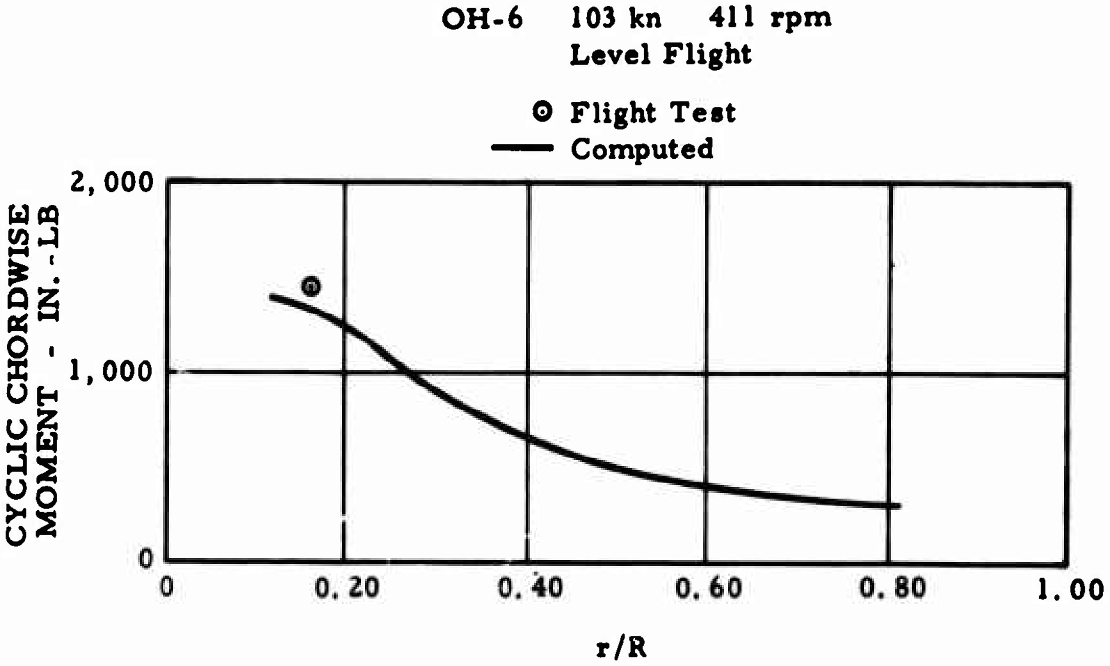


Figure 51. Chordwise Moment Distribution - Comparison of Theory and Flight Test.

## RESULTS OF FULLY COUPLED HEAVY-LIFT HELICOPTER STATIC AND DYNAMIC LOADS ANALYSIS

### BLADE AND CONTROL SYSTEM LOADS

The computed blade loads for static and fatigue design purposes are shown in Figures 52 through 55. Some minor modifications to these loads were made in lieu of minor adjustments to the computer program. Typical samples of the unmodified computer output may be found in Appendix VI. The cyclic flapwise bending moments were multiplied by the ratio of actual moment in the OH-6 to computed moment for the OH-6, and the result is shown in Figure 52.

Since the lead-lag damper has a constant moment independent of amplitude (for the small amplitudes encountered here), the cyclic chordwise momert at the lead-lag hinge must match the known damper moment. Figure 53 shows the result of making this adjustment.

Two sources of conservatism are present in the cyclic flapwise loads. First, chordwise-flapwise coupling causes increased flapwise moment because of the excessive chordwise moment. Second, the thrust developed in the computed condition is 60,000 pounds, compared with 52,744 pounds for the actual design condition. Cyclic flapwise moment is therefore too high by approximately the ratio of the thrusts (60,000/52,744).

Figures 54 and 55 show the maximum loads developed in a pullup to the maximum attainable load factor for this flight condition (tip speed = 675 fps, forward speed = 110 knots, gross weight = 66,000 pounds).

### DYNAMIC AND EROELASTIC INSTABILITY OR FLUTTER

Examination of the transient and steady-state blade load versus time plots indicates convergence to a steady-state or decreasing load level for all structural elements. This indicates the existence of positive real parts of the eigenvalues for all modes at the flight condition studied (namely, 110-kn forward speed, 675-ft/sec tip speed, sea level standard atmosphere, 60,000-lb gross weight). Freedom from flutter is therefore substantiated at this condition.

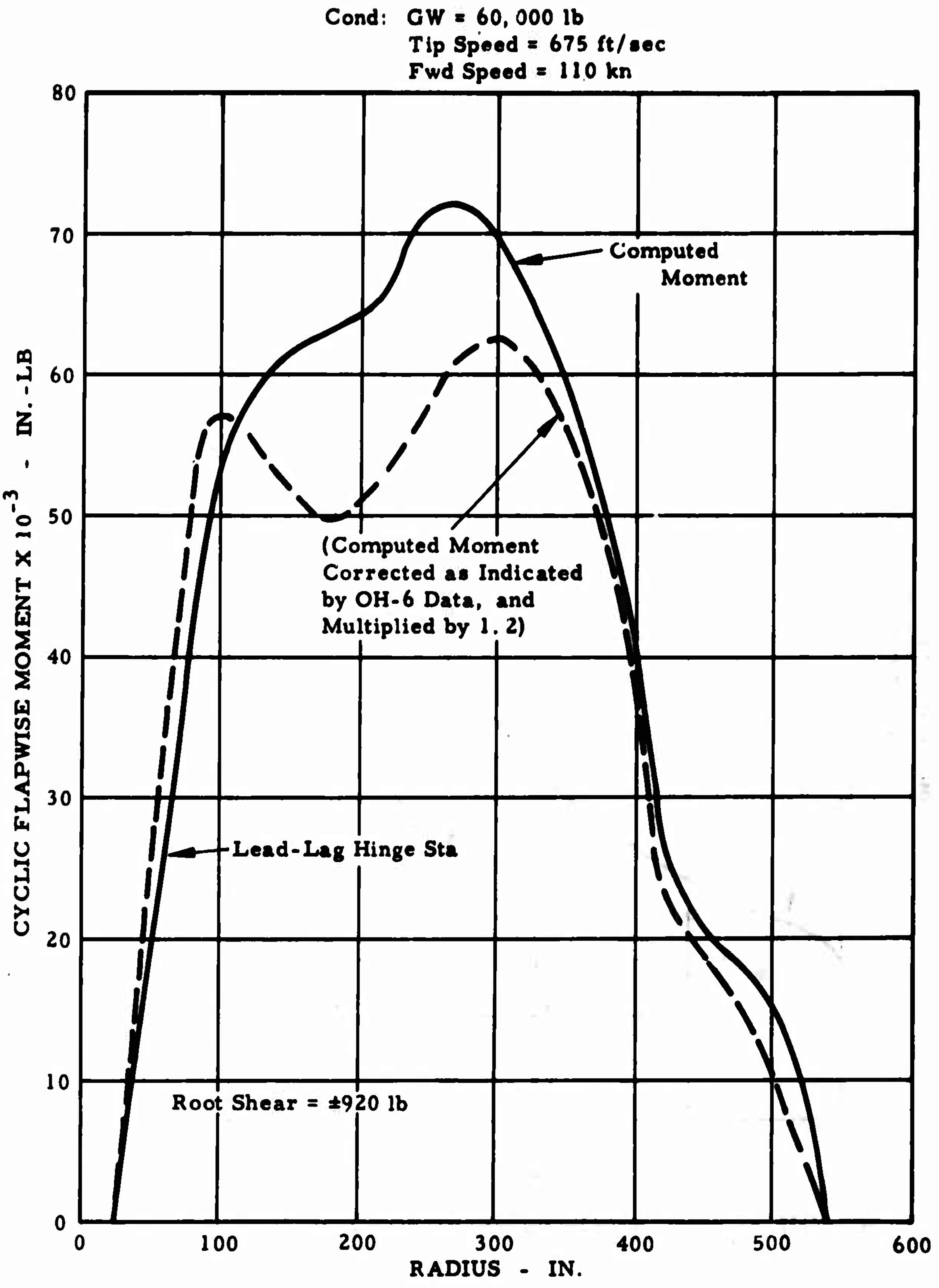


Figure 52. Cyclic Flapwise Moment Distribution.

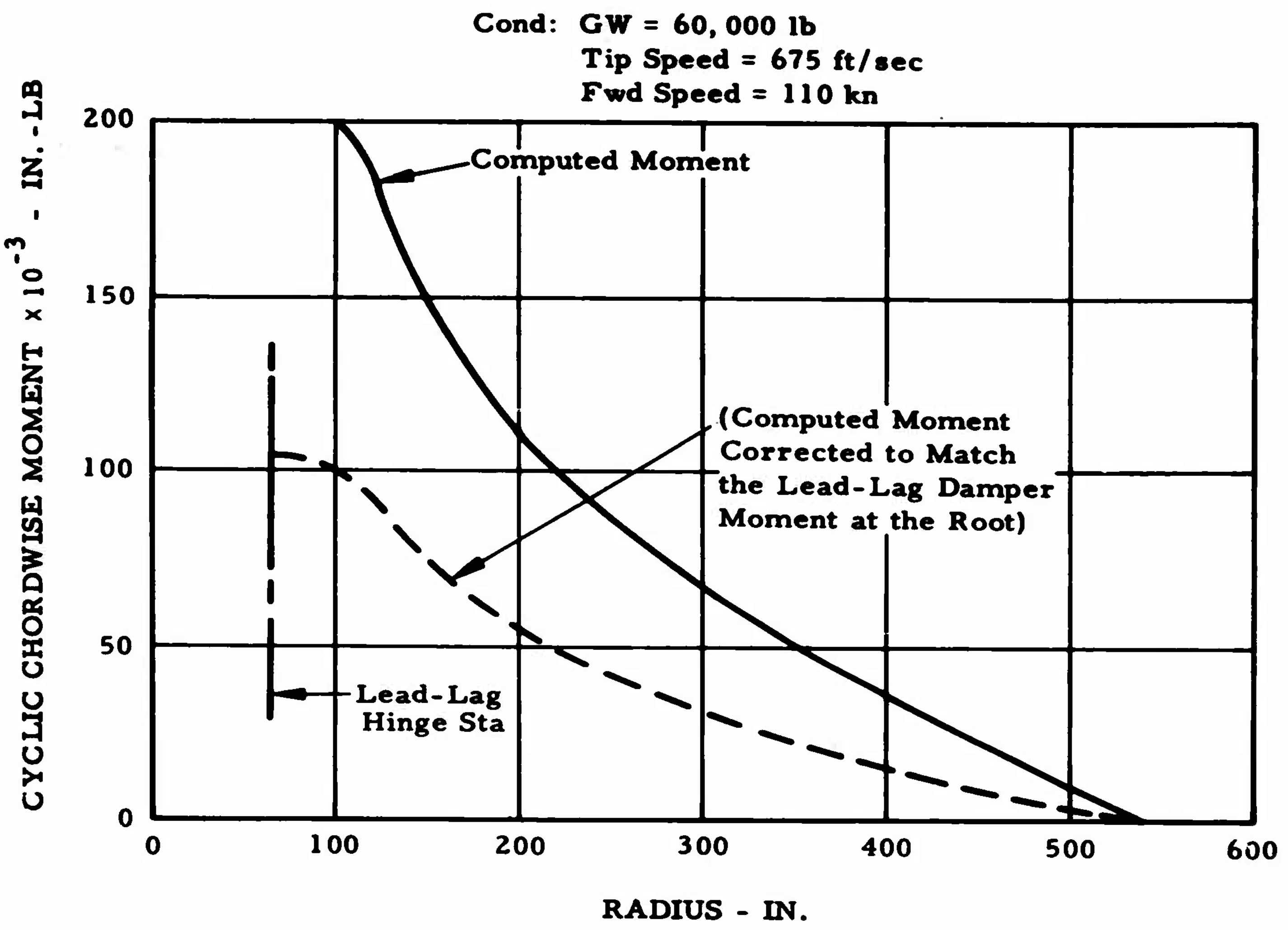


Figure 53. Cyclic Chordwise Moment Distribution.

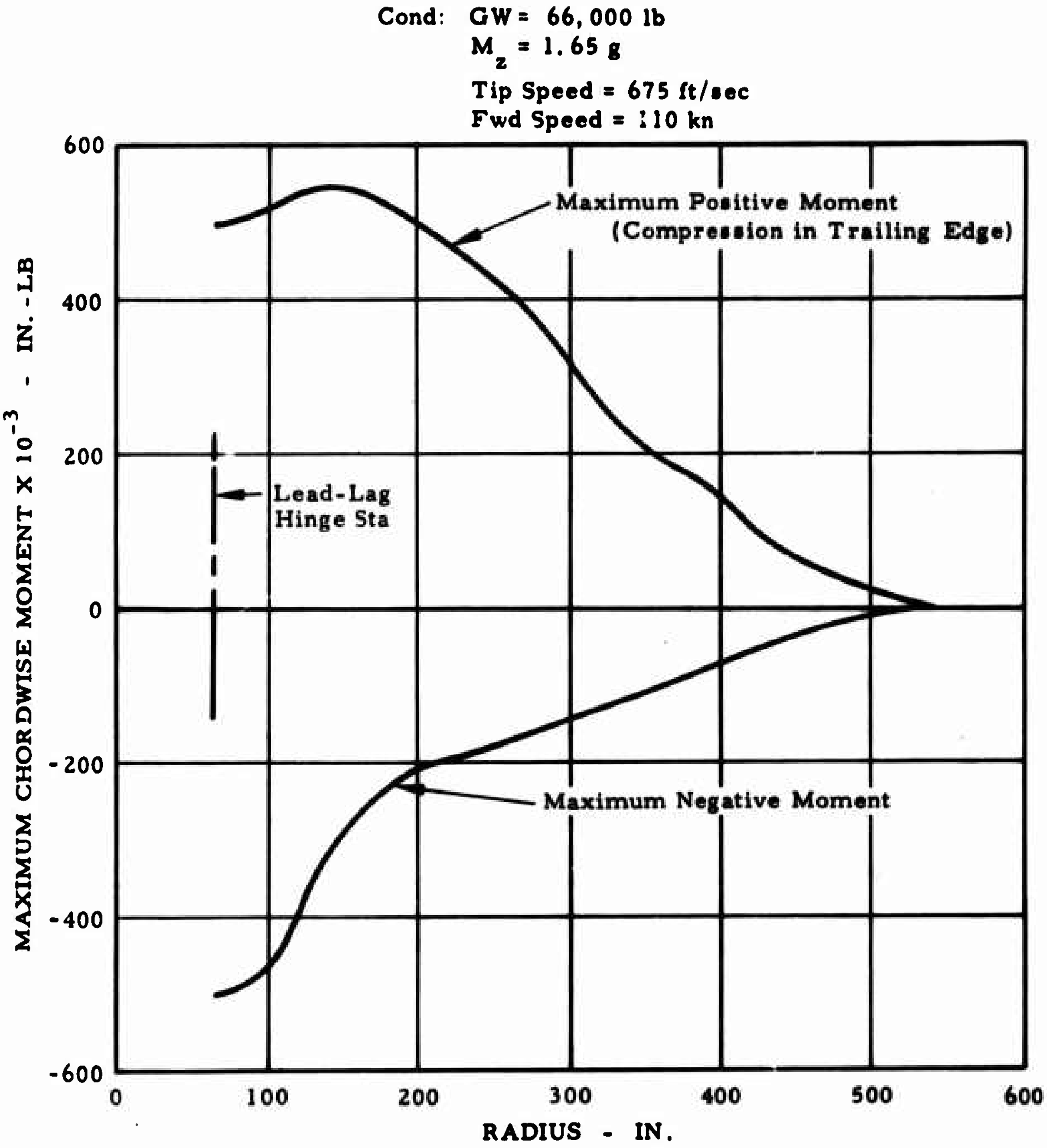


Figure 54. Maximum Chordwise Moment Distribution.

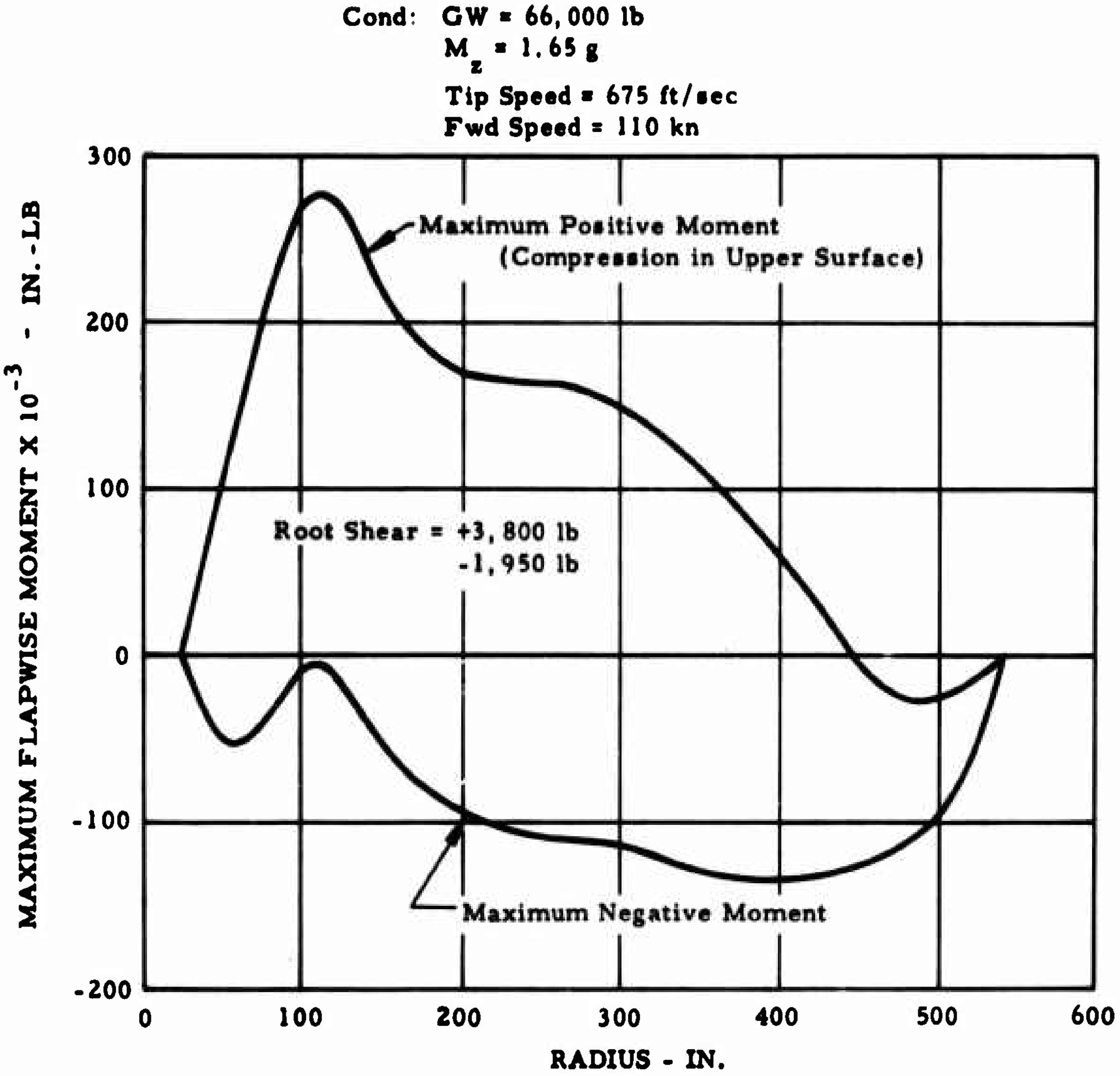


Figure 55. Maximum Flapwise Moment Distribution.

# HUB MOCKUP

## DESCRIPTION

A full-size mockup of the proposed Hot Cycle heavy-lift helicopter rotor hub has been designed and fabricated. The parts that have been simulated are the mast, swashplate and lift link, ducting from rotating seal to blade constant section, hub structure, torque tube, retention straps, torque flexures, lead-lag damper, and the blade transition area. The various parts have been fabricated from wood for the most part, with some plastic and sheet metal material used as required. The mockup has been built around a vertical standpipe, mounted to a platform, that supports the components in their proper relationship. The mockup has been designed and built in such a manner that components may be moved through their design motions in flapping, feathering, and lead-lag. The completed mockup is shown in Figures 56 through 64.

## PURPOSE

The purpose of constructing the mockup was to check the following so that the necessary changes could be incorporated in the design:

1. Structural, control, and duct clearances through the full range and combinations of control movements and blade-hub motions.
2. Structure and controls for simplifications of load paths and fabrication.
3. Ducts for simplification of fabrication and routing for minimum duct losses.
4. Action of the retention straps, droop stops, and lead-lag hinges, stops, and dampers, if applicable.

## RESULTS

As a result of constructing the mockup, the following items were accomplished.

1. The centrifugal force load path from the retention straps through the hub structure was simplified by a redesign of the lower plate.
2. Interferences between hub structure and gas ducts were determined, and the design was corrected.
3. A redesign of the torque tube was determined that permitted a more favorable routing of the gas ducts. By narrowing the

outboard end of the torque tube, it was possible to move the ducts in tighter to the feathering centerline, which results in less motion at the articulating duct seals.

4. The lead-lag damper was relocated to shorten the load path.
5. Interferences between the damper support and ducts were determined and corrected.

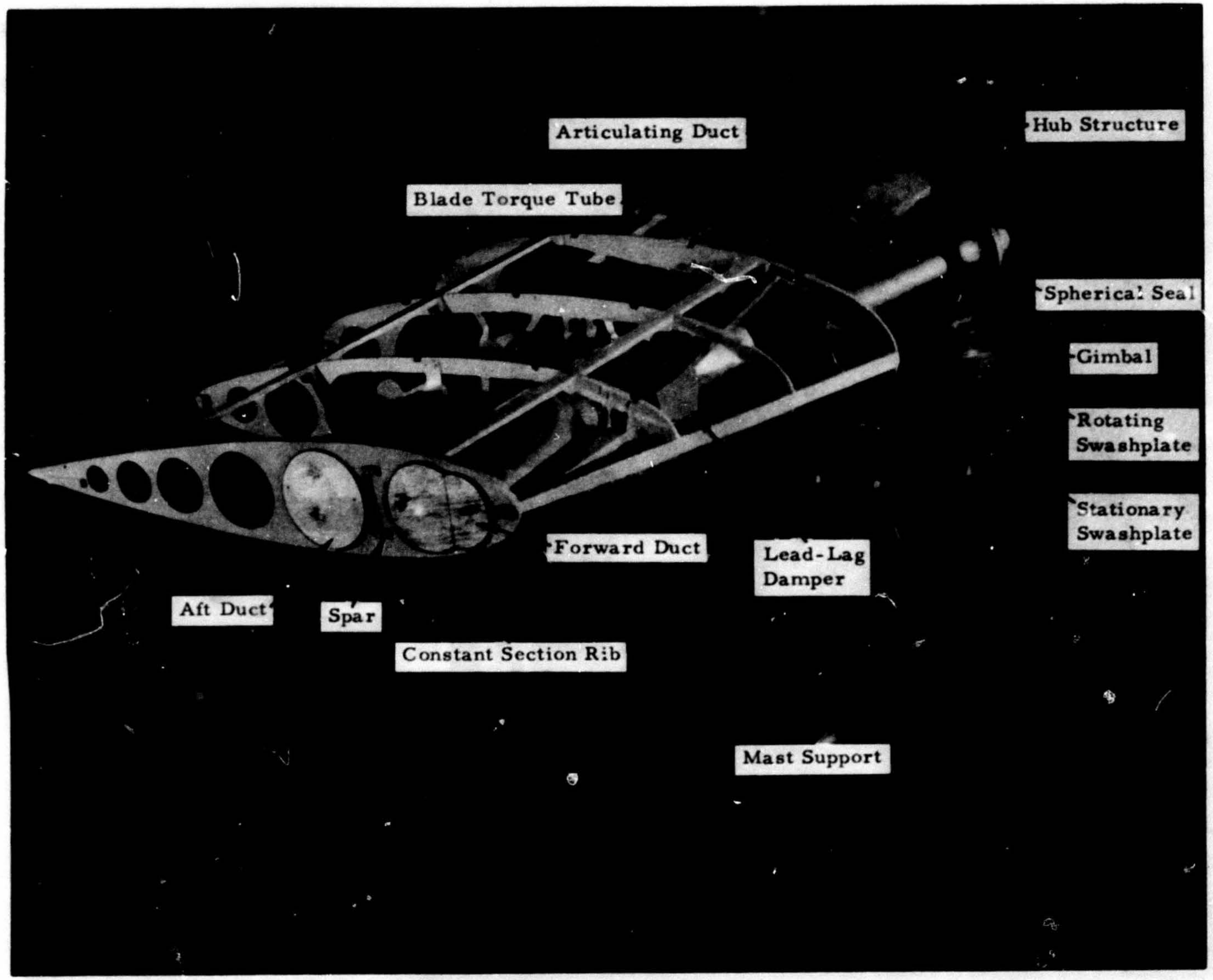


Figure 56. Plan View of Blade Transition Area.

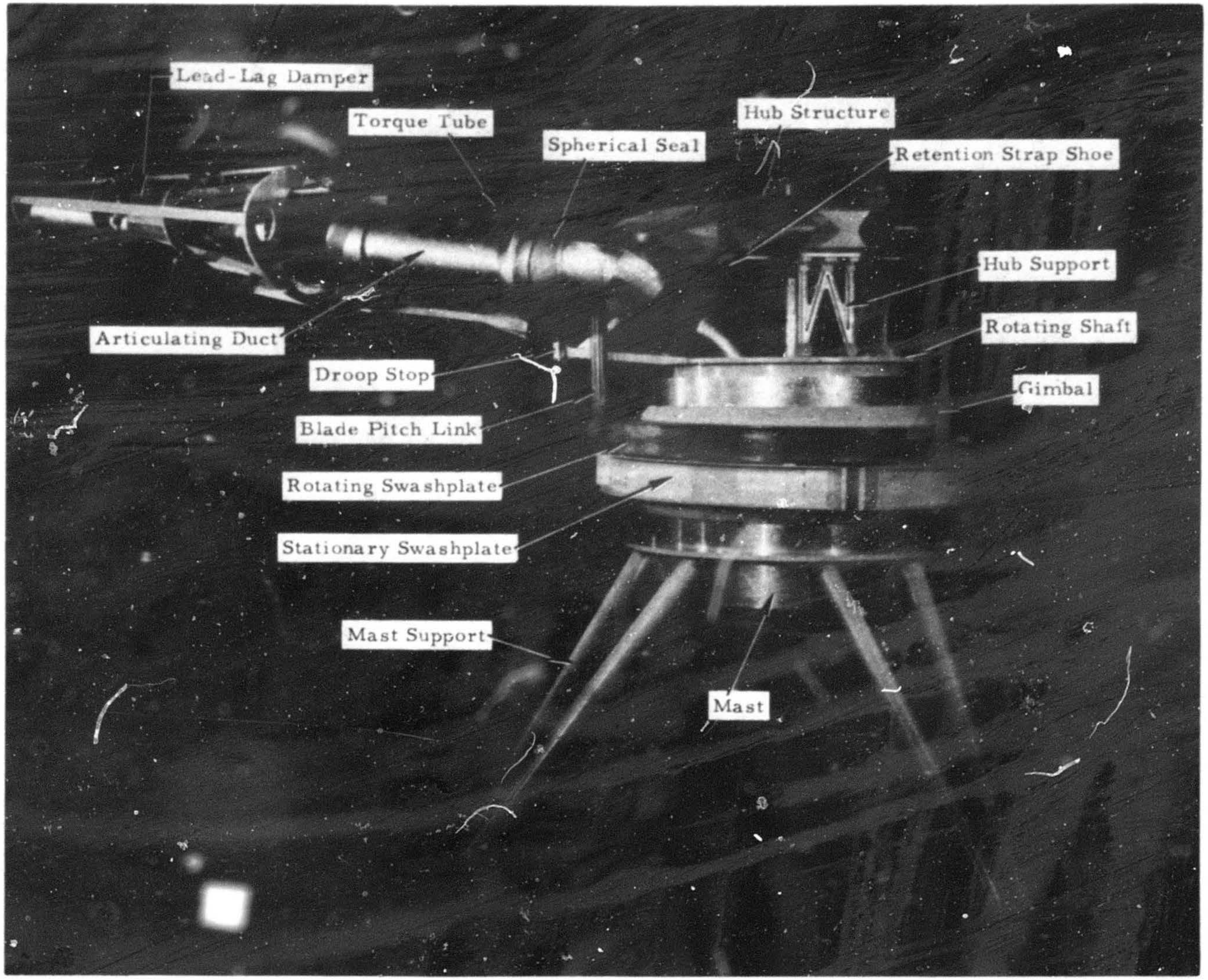


Figure 57. Blade in Cruise Coning Position.

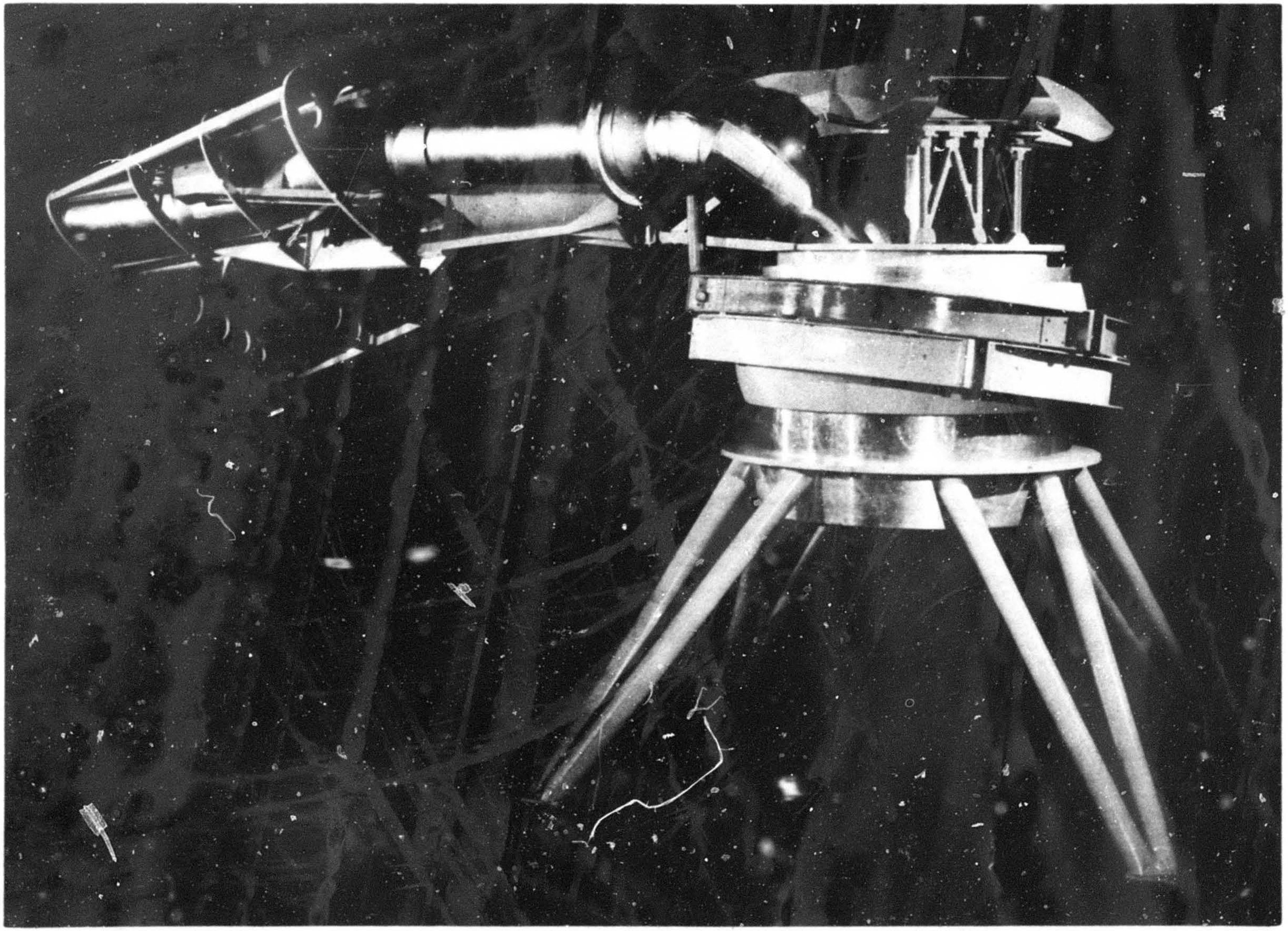


Figure 58. Blade on Droop Stop (Maximum Positive Feathering).

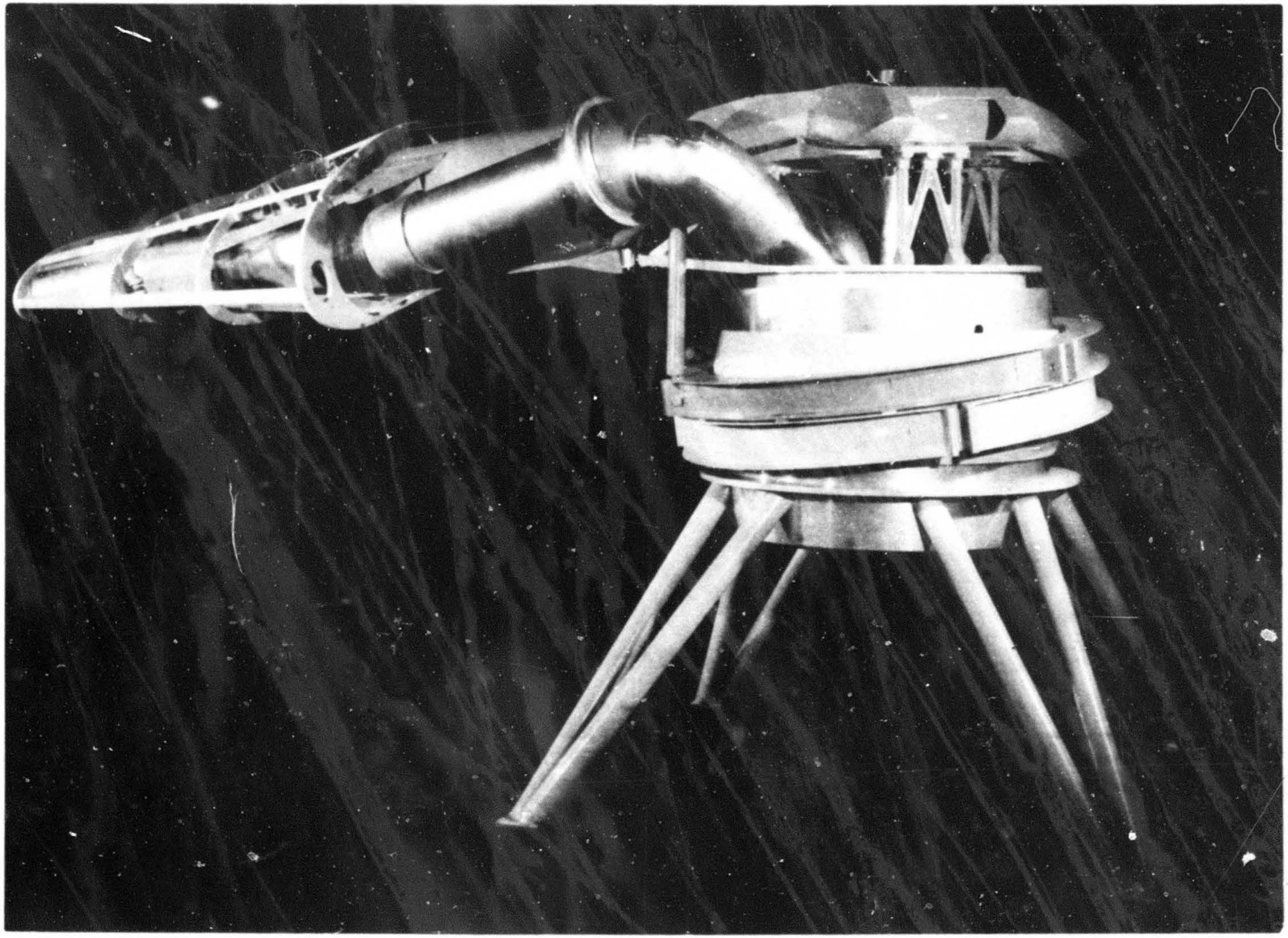


Figure 59. Blade on Droop Stop (Maximum Negative Feathering).

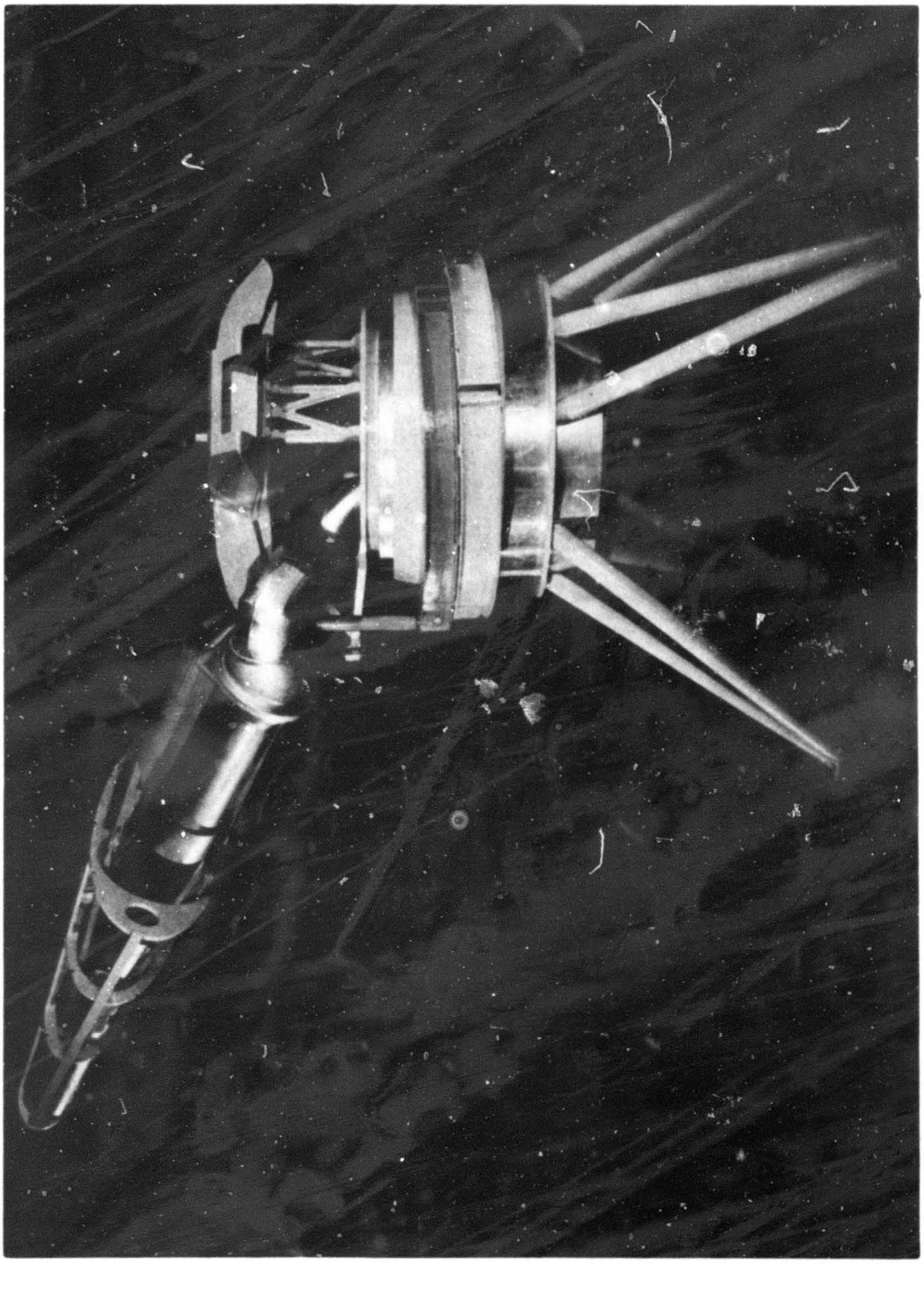


Figure 60. Blade in Maximum Up Flapping Condition.

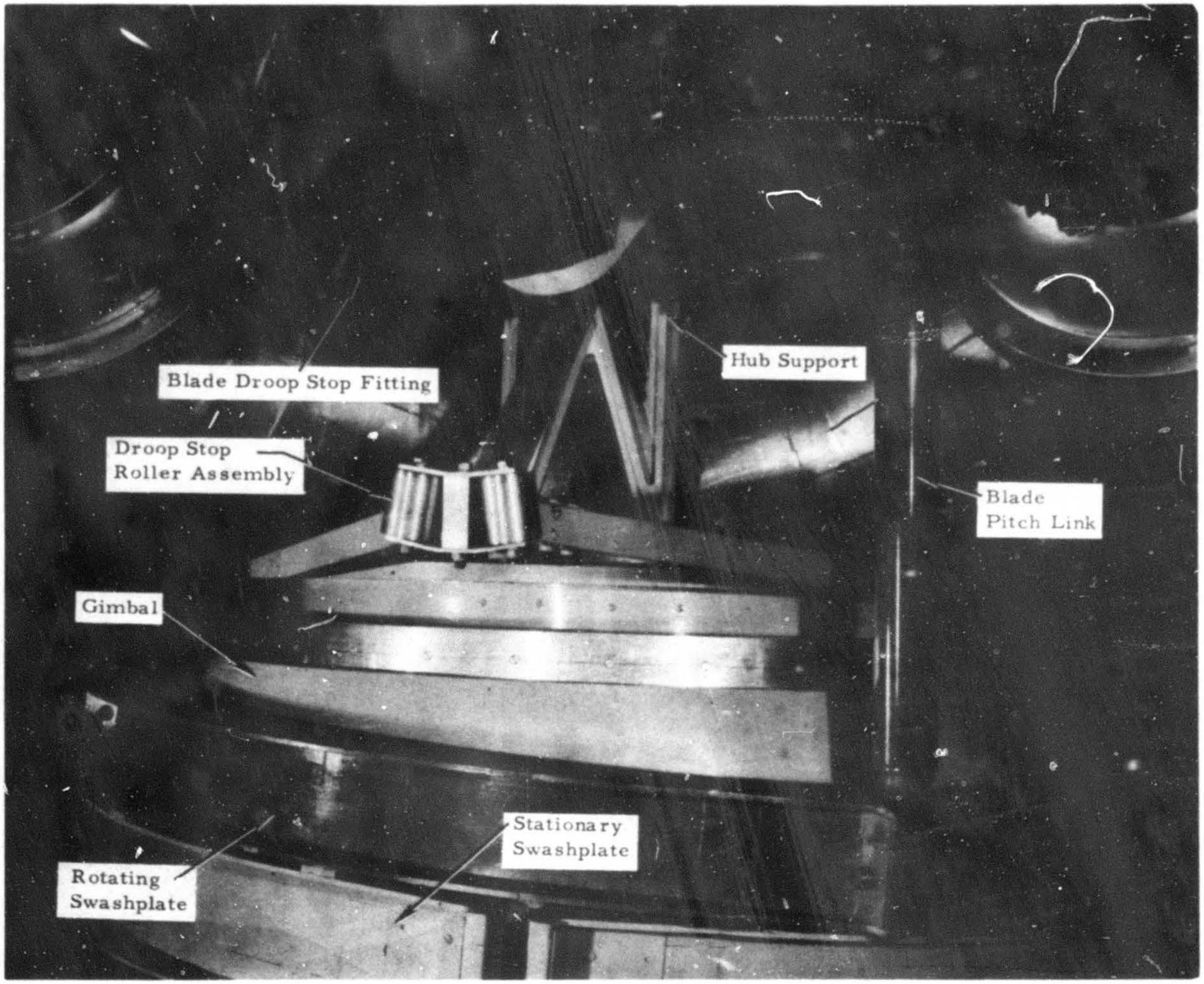


Figure 61. Droop Stop (Blade in Cruise Coning Position).

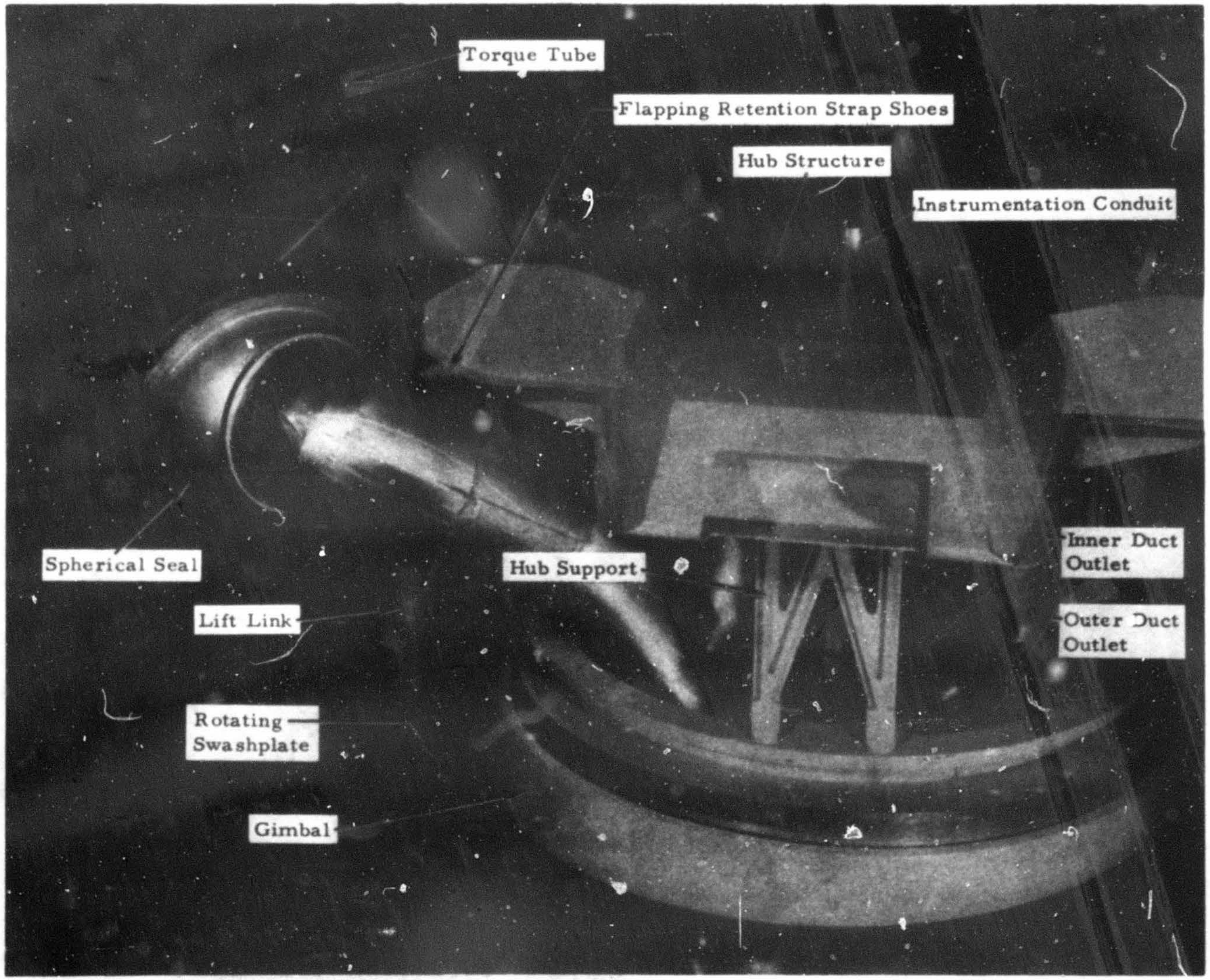


Figure 62. Hub and Duct Configuration.

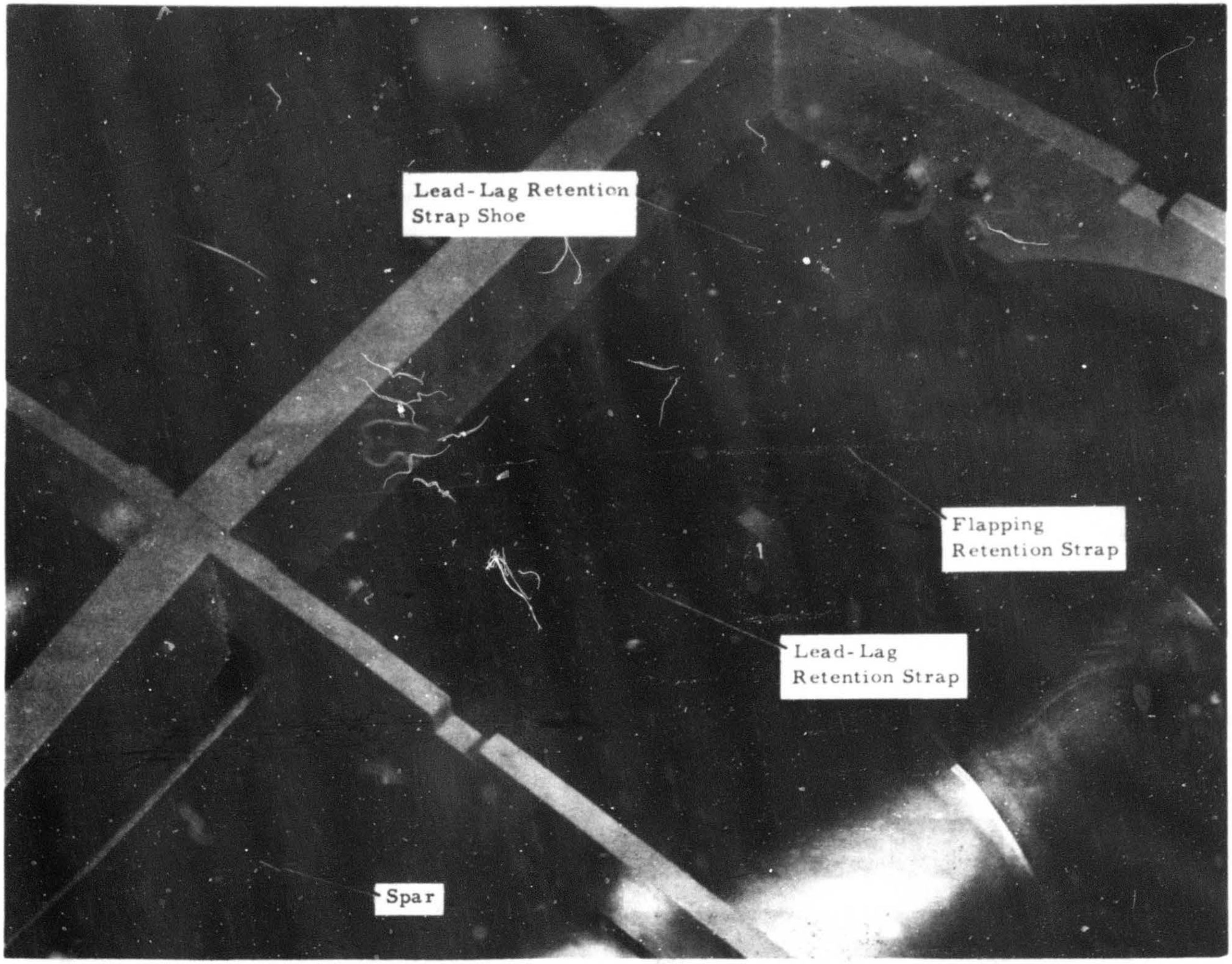


Figure 63. Lead-Lag Strap Retention.

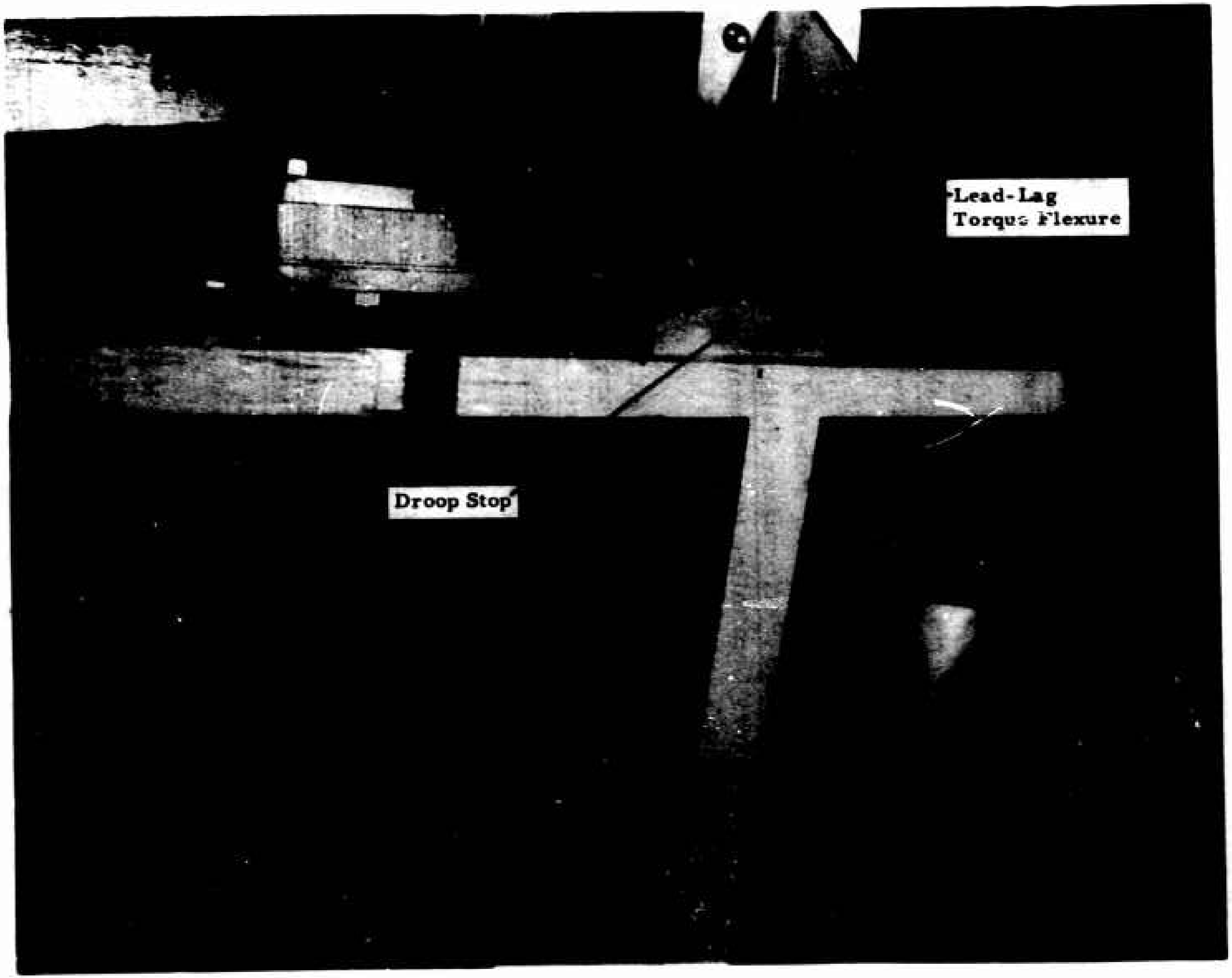


Figure 64. View Looking Up At Lead-Lag Hinge.

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# APPENDIX I
# SUMMARY WEIGHT STATEMENT
# AND DETAILED WEIGHT CALCULATIONS

**MIL-STD-451, PART I**

**NAME**__________
**DATE**__________

**PAGE**__________
**MODEL** 395 Config 2
**REPORT**__________

## SUMMARY WEIGHT STATEMENT
## ROTORCRAFT ONLY
**ESTIMATED ~~CALCULATED ACTUAL~~**

**(Cross out those not applicable)**

## HOT CYCLE HEAVY-LIFT HELICOPTER
## CONFIGURATION 2

**CONTRACT**__________
**ROTORCRAFT, GOVERNMENT NUMBER**__________
**ROTORCRAFT, CONTRACTOR NUMBER**__________
**MANUFACTURED BY** Hughes Tool Company - Aircraft Division

| | | **Main** | **Auxiliary** |
|---|---|---|---|
| **Engine** | **Manufactured by** | General Electric | |
| | **Model** | GE-1 | |
| | **Number** | 2 | |
| **Propeller** | **Manufactured by** | | |
| | **Model** | | |
| | **Number** | | |

ROTORCRAFT
SUMMARY WEIGHT STATEMENT
WEIGHT EMPTY

NAME__________
DATE__________

PAGE__________
MODEL 395 Config. 2
REPORT__________

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | | | | | | | |
| 2 | ROTOR GROUP | | | | | | 5440 |
| 3 | BLADE ASSEMBLY | | | | | 3806 | |
| 4 | HUB | | | | | 1634 | |
| 5 | HINGE AND BLADE RETENTION | | | | | | |
| 6 | | FLAP | PING | | | | |
| 7 | | LEAD | LAG | | | | |
| 8 | | PITC | H | | | | |
| 9 | | FOLD | ING | | | | |
| 10 | WING GROUP | | | | | | |
| 11 | WING PANELS—BASIC STRUCTURE | | | | | | |
| 12 | CENTER SECTION—BASIC STRUCTURE | | | | | | |
| 13 | INTERMEDIATE PANEL—BASIC STRUCTURE | | | | | | |
| 14 | OUTER PANEL—BASIC STRUCTURE—INCL TIPS | | | LBS | | | |
| 15 | SECONDARY STRUC—INCL FOLD MECH | | | LBS | | | |
| 16 | AILERONS—INCL BALANCE WTS | | | LBS | | | |
| 17 | FLAPS | | | | | | |
| 18 | —TRAILING EDGE | | | | | | |
| 19 | —LEADING EDGE | | | | | | |
| 20 | SLATS | | | | | | |
| 21 | SPOILERS | | | | | | |
| 22 | | | | | | | |
| 23 | TAIL GROUP | | | | | | 1040 |
| 24 | TAIL ROTOR | | | | | 70 | |
| 25 | —BLADES | | | | | | |
| 26 | —HUB | | | | | | |
| 27 | STABILIZER—BASIC STRUCTURE | | | | | 970 | |
| 28 | FINS—BASIC STRUCTURE—INCL DORSAL | | | LBS | | | |
| 29 | SECONDARY STRUCTURE—STABILIZER AND FINS | | | | | | |
| 30 | ELEVATOR—INCL BALANCE WEIGHT | | | LBS | | | |
| 31 | RUDDER—INCL BALANCE WEIGHT | | | LBS | | | |
| 32 | | | | | | | |
| 33 | BODY GROUP | | | | | | 2843 |
| 34 | FUSELAGE OR HULL—BASIC STRUCTURE | | | | | | |
| 35 | BOOMS—BASIC STRUCTURE | | | | | | |
| 36 | SECONDARY STRUCTURE—FUSELAGE OR HULL | | | | | | |
| 37 | —BOOMS | | | | | | |
| 38 | —DOORS, PANELS & MISC | | | | | | |
| 39 | | | | | | | |
| 40 | | | | | | | |
| 41 | ALIGHTING GEAR—LAND TYPE | | | | | | 2185 |
| 42 | LOCATION | ROLLING | STRUCT | CONTROLS | | | |
| 43 | | ASSEMBLY | | | | | |
| 44 | | | | | | | |
| 45 | | | | | | | |
| 46 | | | | | | | |
| 47 | | | | | | | |
| 48 | | | | | | | |
| 49 | | | | | | | |
| 50 | ALIGHTING GEAR GROUP—WATER TYPE | | | | | | |
| 51 | LOCATION | FLOATS | STRUTS | CONTROLS | | | |
| 52 | | | | | | | |
| 53 | | | | | | | |
| 54 | | | | | | | |
| 55 | | | | | | | |
| 56 | | | | | | | |
| 57 | | | | | | | |

* Wheels, Brakes, Tires, Tubes and Air.

NAME__________
DATE__________

## ROTORCRAFT
## SUMMARY WEIGHT STATEMENT
## WEIGHT EMPTY—Continued

PAGE__________
MODEL 395 Config. 2
REPORT__________

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | | | | | | | |
| 2 | FLIGHT CONTROLS GROUP | | | | | | 1414 |
| 3 | COCKPIT CONTROLS | | | | | 33 | |
| 4 | AUTOMATIC STABILIZATION | | | | | 959 | |
| 5 | SYSTEM CONTROLS—ROTOR NON ROTATING | | | | | 260 | |
| 6 | ROTATING | | | | | | |
| 7 | —FIXED WING | | | | | | |
| 8 | | | | | | 162 | |
| 9 | | | | | | | |
| 10 | ENGINE SECTION OR NACELLE GROUP | | | | | | 460 |
| 11 | INBOARD | | | | | | |
| 12 | CENTER | | | | | | |
| 13 | OUTBOARD | | | | | | |
| 14 | DOORS, PANELS AND MISC | | | | | | |
| 15 | | | | | | | |
| 16 | PROPULSION GROUP | | | | | | 2474 |
| 17 | | | X AUXILIARY X | | X MAIN X | | |
| 18 | ENGINE INSTALLATION | | | | | 1627 | |
| 19 | ENGINE | | | | | ** | |
| 20 | TIP BURNERS | | | | | | |
| 21 | LOAD COMPRESSOR | | | | | | |
| 22 | REDUCTION GEAR BOX, ETC | | | | | | |
| 23 | ACCESSORY GEAR BOXES AND DRIVES | | | | | | |
| 24 | SUPERCHARGER—FOR TURBOS | | | | | | |
| 25 | AIR INDUCTION SYSTEM | | | | | * | |
| 26 | EXHAUST SYSTEM | | | | | * | |
| 27 | COOLING SYSTEM | | | | | | |
| 28 | LUBRICATING SYSTEM | | | | | 64 | |
| 29 | TANKS | | | | | | |
| 30 | BACKING BD, TANK SUP & PADDING | | | | | | |
| 31 | COOLING INSTALLATION | | | | | | |
| 32 | PLUMBING, ETC | | | | | | |
| 33 | FUEL SYSTEM | | | | | * | |
| 34 | TANKS—UNPROTECTED | | | | | | |
| 35 | —PROTECTED | | | | | | |
| 36 | BACKING BD, TANK SUP & PADDING | | | | | | |
| 37 | PLUMBING, ETC | | | | | | |
| 38 | WATER INJECTION SYSTEM | | | | | | |
| 39 | ENGINE CONTROLS | | | | | 60 | |
| 40 | STARTING SYSTEM | | | | | 70 | |
| 41 | PROPELLER INSTALLATION | | | | | | |
| 42 | DRIVE SYSTEM | | | | | 653 | |
| 43 | GEAR BOXES | | | | | | |
| 44 | LUBE SYSTEM | | | | | | |
| 45 | CLUTCH AND MISC | | | | | | |
| 46 | TRANSMISSION DRIVE | | | | | | |
| 47 | ROTOR SHAFT | | | | | | |
| 48 | JET DRIVE | | | | | | |
| 49 | | | | | | | |
| 50 | | | | | | | |
| 51 | | | | | | | |
| 52 | AUXILIARY POWER PLANT GROUP | | | | | | 160 |
| 53 | | | | | | | |
| 54 | | | | | | | |
| 55 | | | | | | | |
| 56 | | | | | | | |
| 57 | | | | | | | |

*Weight for these items included in engine installation weight
**Engine weight confidential

ROTORCRAFT
SUMMARY WEIGHT STATEMENT
WEIGHT EMPTY—Continued

NAME
DATE

PAGE
MODEL 395 Config. 2
REPORT

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | | | | | | | |
| 2 | | | | | | | |
| 3 | | | | | | | |
| 4 | INSTRUMENT AND NAVIGATIONAL EQUIPMENT GROUP | | | | | | 180 |
| 5 | INSTRUMENTS | | | | | | |
| 6 | NAVIGATIONAL EQUIPMENT | | | | | | |
| 7 | | | | | | | |
| 8 | | | | | | | |
| 9 | HYDRAULIC AND PNEUMATIC GROUP | | | | | | 711 |
| 10 | HYDRAULIC | | | | | | |
| 11 | PNEUMATIC | | | | | | |
| 12 | | | | | | | |
| 13 | | | | | | | |
| 14 | ELECTRICAL GROUP | | | | | | 742 |
| 15 | A C SYSTEM | | | | | | |
| 16 | D C SYSTEM | | | | | | |
| 17 | | | | | | | |
| 18 | | | | | | | |
| 19 | ELECTRONICS GROUP | | | | | | 150 |
| 20 | EQUIPMENT | | | | | | |
| 21 | INSTALLATION | | | | | | |
| 22 | | | | | | | |
| 23 | | | | | | | |
| 24 | ARMAMENT GROUP—INCL GUNFIRE PROTECTION | | | | LBS | | |
| 25 | | | | | | | |
| 26 | FURNISHINGS AND EQUIPMENT GROUP | | | | | | 300 |
| 27 | ACCOMMODATIONS FOR PERSONNEL | | | | | | |
| 28 | MISCELLANEOUS EQUIPMENT | X INCL | | LBS | BALLAST X | | |
| 29 | FURNISHINGS | | | | | | |
| 30 | EMERGENCY EQUIPMENT | | | | | | |
| 31 | | | | | | | |
| 32 | | | | | | | |
| 33 | | | | | | | |
| 34 | AIR CONDITIONING AND ANTI-ICING EQUIPMENT | | | | | | 100 |
| 35 | AIR CONDITIONING | | | | | | |
| 36 | ANTI-ICING | | | | | | |
| 37 | | | | | | | |
| 38 | | | | | | | |
| 39 | PHOTOGRAPHIC GROUP | | | | | | |
| 40 | EQUIPMENT | | | | | | |
| 41 | INSTALLATION | | | | | | |
| 42 | | | | | | | |
| 43 | AUXILIARY GEAR GROUP | | | | | | |
| 44 | AIRCRAFT HANDLING GEAR | | | | | | |
| 45 | LOAD HANDLING GEAR | | | | | | 1400 |
| 46 | ATO GEAR | | | | | | |
| 47 | | | | | | | |
| 48 | | | | | | | |
| 49 | | | | | | | |
| 50 | | | | | | | |
| 51 | | | | | | | |
| 52 | | | | | | | |
| 53 | | | | | | | |
| 54 | MANUFACTURING VARIATION | | | | | | |
| 55 | | | | | | | |
| 56 | | | | | | | |
| 57 | TOTAL—WEIGHT EMPTY—PAGES 2, 3 AND 4 | | | | | | 19,599 |

PAGE____________
NAME____________
DATE____________

**SUMMARY WEIGHT STATEMENT**
**USEFUL LOAD GROSS WEIGHT**

MODEL 395 Config. 2
REPORT____________

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | LOAD CONDITION | | | Transport | | Heavy Lift | |
| 2 | | | | 12 ton | | 20 ton | |
| 3 | CREW—NO. 3 | | | 600 | | 600 | |
| 4 | PASSENGERS—NO. | | | | | | |
| 5 | FUEL LOCATION | TYPE | GALS | | | | |
| 6 | UNUSABLE Fuselage | JP-4 | | 100 | | 100 | |
| 7 | INTERNAL | | | 7831 | | 3901 | |
| 8 | | | | | | | |
| 9 | | | | | | | |
| 10 | | | | | | | |
| 11 | EXTERNAL | | | | | | |
| 12 | | | | | | | |
| 13 | | | | | | | |
| 14 | | | | | | | |
| 15 | BOMB BAY | | | | | | |
| 16 | | | | | | | |
| 17 | | | | | | | |
| 18 | | | | | | | |
| 19 | OIL | | | 30 | | 30 | |
| 20 | UNUSABLE | | | | | | |
| 21 | ENGINE | | | | | | |
| 22 | | | | | | | |
| 23 | | | | | | | |
| 24 | | | | | | | |
| 25 | BAGGAGE - Crew Kits | | | 50 | | 50 | |
| 26 | CARGO - Payload | | | 24,000 | | 40,000 | |
| 27 | | | | | | | |
| 28 | ARMAMENT | | | | | | |
| 29 | GUNS—LOCATION TYPE** | QUANTITY | CALIBER | | | | |
| 30 | | | | | | | |
| 31 | | | | | | | |
| 32 | | | | | | | |
| 33 | | | | | | | |
| 34 | AMM | | | | | | |
| 35 | | | | | | | |
| 36 | | | | | | | |
| 37 | | | | | | | |
| 38 | BOMB INSTL* | | | | | | |
| 39 | BOMBS | | | | | | |
| 40 | | | | | | | |
| 41 | TORPEDO INSTL* | | | | | | |
| 42 | TORPEDOES | | | | | | |
| 43 | | | | | | | |
| 44 | ROCKET INSTL* | | | | | | |
| 45 | ROCKETS | | | | | | |
| 46 | | | | | | | |
| 47 | EQUIPMENT—PYROTECHNICS | | | | | | |
| 48 | —PHOTOGRAPHIC | | | | | | |
| 49 | | | | | | | |
| 50 | —OXYGEN | | | | | | |
| 51 | | | | | | | |
| 52 | —MISCELLANEOUS | | | | | | |
| 53 | | | | | | | |
| 54 | | | | | | | |
| 55 | USEFUL LOAD | | | 32,611 | | 44,681 | |
| 56 | Weight Empty | | | 19,599 | | 19,599 | |
| 57 | GROSS WEIGHTS—PAGES 2-6 | | | 52,260 | | 64,280 | |

* If not specified as Weight Empty.

** Fixed, Flexible, etc.

NAME
DATE

PAGE
MODEL 395 Config. 3
REPORT

## SUMMARY WEIGHT STATEMENT
## ROTORCRAFT ONLY
## ESTIMATED ~~CALCULATED~~ ~~ACTUAL~~

(Cross out those not applicable)

## HOT CYCLE HEAVY-LIFT HELICOPTER
## CONFIGURATION 3

CONTRACT
ROTORCRAFT, GOVERNMENT NUMBER
ROTORCRAFT, CONTRACTOR NUMBER
MANUFACTURED BY Hughes Tool Company - Aircraft Division

| | | Main | Auxiliary |
|---|---|---|---|
| Engine | Manufactured by | General Electric | |
| | Model | GE-1 | |
| | Number | 2 | |
| Propeller | Manufactured by | | |
| | Model | | |
| | Number | | |

NAME____________ ROTORCRAFT PAGE____________

SUMMARY WEIGHT STATEMENT MODEL 395 Config. 3

DATE____________ WEIGHT EMPTY REPORT____________

| | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| 1 | | | | | | | | |
| 2 | ROTOR GROUP | | | | | | 5440 | |
| 3 | BLADE ASSEMBLY | | | | | 3806 | | |
| 4 | HUB | | | | | 1634 | | |
| 5 | HINGE AND BLADE RETENTION | | | | | | | |
| 6 | | FLAP | PING | | | | | |
| 7 | | LEAD | LAG | | | | | |
| 8 | | PITC | H | | | | | |
| 9 | | FOLD | ING | | | | | |
| 10 | WING GROUP | | | | | | | |
| 11 | WING PANELS—BASIC STRUCTURE | | | | | | | |
| 12 | CENTER SECTION—BASIC STRUCTURE | | | | | | | |
| 13 | INTERMEDIATE PANEL—BASIC STRUCTURE | | | | | | | |
| 14 | OUTER PANEL—BASIC STRUCTURE—INCL TIPS | | | LBS | | | | |
| 15 | SECONDARY STRUC—INCL FOLD MECH | | | LBS | | | | |
| 16 | AILERONS—INCL BALANCE WTS | | | LBS | | | | |
| 17 | FLAPS | | | | | | | |
| 18 | —TRAILING EDGE | | | | | | | |
| 19 | —LEADING EDGE | | | | | | | |
| 20 | SLATS | | | | | | | |
| 21 | SPOILERS | | | | | | | |
| 22 | | | | | | | | |
| 23 | TAIL GROUP | | | | | | 1062 | |
| 24 | TAIL ROTOR | | | | | 70 | | |
| 25 | —BLADES | | | | | | | |
| 26 | —HUB | | | | | | | |
| 27 | STABILIZER—BASIC STRUCTURE | | | | | 992 | | |
| 28 | FINS—BASIC STRUCTURE—INCL DORSAL | | | LBS | | | | |
| 29 | SECONDARY STRUCTURE—STABILIZER AND FINS | | | | | | | |
| 30 | ELEVATOR—INCL BALANCE WEIGHT | | | LBS | | | | |
| 31 | RUDDER—INCL BALANCE WEIGHT | | | LBS | | | | |
| 32 | | | | | | | | |
| 33 | BODY GROUP | | | | | | 3615 | |
| 34 | FUSELAGE OR HULL—BASIC STRUCTURE | | | | | | | |
| 35 | BOOMS—BASIC STRUCTURE | | | | | | | |
| 36 | SECONDARY STRUCTURE—FUSELAGE OR HULL | | | | | | | |
| 37 | —BOOMS | | | | | | | |
| 38 | —DOORS, PANELS & MISC | | | | | | | |
| 39 | | | | | | | | |
| 40 | | | | | | | | |
| 41 | ALIGHTING GEAR—LAND TYPE | | | | | | 2300 | Fixed |
| 42 | LOCATION | ROLLING* | STRUCT | CONTROLS | | | (2810) | Retract. |
| 43 | | ASSEMBLY | | | | | | |
| 44 | | | | | | | | |
| 45 | | | | | | | | |
| 46 | | | | | | | | |
| 47 | | | | | | | | |
| 48 | | | | | | | | |
| 49 | | | | | | | | |
| 50 | ALIGHTING GEAR GROUP—WATER TYPE | | | | | | | |
| 51 | LOCATION | FLOATS | STRUTS | CONTROLS | | | | |
| 52 | | | | | | | | |
| 53 | | | | | | | | |
| 54 | | | | | | | | |
| 55 | | | | | | | | |
| 56 | | | | | | | | |
| 57 | | | | | | | | |

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* Wheels, Brakes, Tires, Tubes and Air.

NAME____________
DATE____________

### ROTORCRAFT
### SUMMARY WEIGHT STATEMENT
### WEIGHT EMPTY—Continued

PAGE____________
MODEL 395 Config. 3
REPORT____________

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | | | | | | | |
| 2 | FLIGHT CONTROLS GROUP | | | | | | 1445 |
| 3 | COCKPIT CONTROLS | | | | | 33 | |
| 4 | AUTOMATIC STABILIZATION | | | | | 984 | |
| 5 | SYSTEM CONTROLS—ROTOR NON ROTATING | | | | | 266 | |
| 6 | ROTATING | | | | | | |
| 7 | —FIXED WING | | | | | | |
| 8 | Intermediate Linkages | | | | | 162 | |
| 9 | | | | | | | |
| 10 | ENGINE SECTION OR NACELLE GROUP | | | | | | 460 |
| 11 | INBOARD | | | | | | |
| 12 | CENTER | | | | | | |
| 13 | OUTBOARD | | | | | | |
| 14 | DOORS, PANELS AND MISC | | | | | | |
| 15 | | | | | | | |
| 16 | PROPULSION GROUP | | | | | | 2478 |
| 17 | | | X AUXILIARY X | | X MAIN X | | |
| 18 | ENGINE INSTALLATION | | | | | 1627 | |
| 19 | ENGINE | | | | | ** | |
| 20 | TIP BURNERS | | | | | | |
| 21 | LOAD COMPRESSOR | | | | | | |
| 22 | REDUCTION GEAR BOX, ETC | | | | | | |
| 23 | ACCESSORY GEAR BOXES AND DRIVES | | | | | | |
| 24 | SUPERCHARGER—FOR TURBOS | | | | | | |
| 25 | AIR INDUCTION SYSTEM | | | | | * | |
| 26 | EXHAUST SYSTEM | | | | | * | |
| 27 | COOLING SYSTEM | | | | | | |
| 28 | LUBRICATING SYSTEM | | | | | 64 | |
| 29 | TANKS | | | | | | |
| 30 | BACKING BD, TANK SUP & PADDING | | | | | | |
| 31 | COOLING INSTALLATION | | | | | | |
| 32 | PLUMBING, ETC | | | | | | |
| 33 | FUEL SYSTEM | | | | | * | |
| 34 | TANKS—UNPROTECTED | | | | | | |
| 35 | —PROTECTED | | | | | | |
| 36 | BACKING BD, TANK SUP & PADDING | | | | | | |
| 37 | PLUMBING, ETC | | | | | | |
| 38 | WATER INJECTION SYSTEM | | | | | | |
| 39 | ENGINE CONTROLS | | | | | 60 | |
| 40 | STARTING SYSTEM | | | | | 70 | |
| 41 | PROPELLER INSTALLATION | | | | | 657 | |
| 42 | DRIVE SYSTEM | | | | | | |
| 43 | GEAR BOXES | | | | | | |
| 44 | LUBE SYSTEM | | | | | | |
| 45 | CLUTCH AND MISC | | | | | | |
| 46 | TRANSMISSION DRIVE | | | | | | |
| 47 | ROTOR SHAFT | | | | | | |
| 48 | JET DRIVE | | | | | | |
| 49 | | | | | | | |
| 50 | | | | | | | |
| 51 | | | | | | | |
| 52 | AUXILIARY POWER PLANT GROUP | | | | | | 160 |
| 53 | | | | | | | |
| 54 | | | | | | | |
| 55 | | | | | | | |
| 56 | | | | | | | |
| 57 | | | | | | | |

*Weight for these items included in engine installation weight
**Engine weight confidential

NAME________________
DATE________________

## ROTORCRAFT
## SUMMARY WEIGHT STATEMENT
## WEIGHT EMPTY—Continued

PAGE________________
MODEL 395 Config. 3
REPORT________________

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | | | | | | | |
| 2 | | | | | | | |
| 3 | | | | | | | 180 |
| 4 | INSTRUMENT AND NAVIGATIONAL EQUIPMENT GROUP | | | | | | |
| 5 | INSTRUMENTS | | | | | | |
| 6 | NAVIGATIONAL EQUIPMENT | | | | | | |
| 7 | | | | | | | |
| 8 | | | | | | | |
| 9 | HYDRAULIC AND PNEUMATIC GROUP | | | | | | 731 |
| 10 | HYDRAULIC | | | | | | |
| 11 | PNEUMATIC | | | | | | |
| 12 | | | | | | | |
| 13 | | | | | | | |
| 14 | ELECTRICAL GROUP | | | | | | 749 |
| 15 | A C SYSTEM | | | | | | |
| 16 | D C SYSTEM | | | | | | |
| 17 | | | | | | | |
| 18 | | | | | | | |
| 19 | ELECTRONICS GROUP | | | | | | 150 |
| 20 | EQUIPMENT | | | | | | |
| 21 | INSTALLATION | | | | | | |
| 22 | | | | | | | |
| 23 | | | | | | | |
| 24 | ARMAMENT GROUP—INCL GUNFIRE PROTECTION | | | | LBS | | |
| 25 | | | | | | | |
| 26 | FURNISHINGS AND EQUIPMENT GROUP | | | | | | 300 |
| 27 | ACCOMMODATIONS FOR PERSONNEL | | | | | | |
| 28 | MISCELLANEOUS EQUIPMENT | X INCL | | LBS | BALLASTX | | |
| 29 | FURNISHINGS | | | | | | |
| 30 | EMERGENCY EQUIPMENT | | | | | | |
| 31 | | | | | | | |
| 32 | | | | | | | |
| 33 | | | | | | | |
| 34 | AIR CONDITIONING AND ANTI-ICING EQUIPMENT | | | | | | 100 |
| 35 | AIR CONDITIONING | | | | | | |
| 36 | ANTI-ICING | | | | | | |
| 37 | | | | | | | |
| 38 | | | | | | | |
| 39 | PHOTOGRAPHIC GROUP | | | | | | |
| 40 | EQUIPMENT | | | | | | |
| 41 | INSTALLATION | | | | | | |
| 42 | | | | | | | |
| 43 | AUXILIARY GEAR GROUP | | | | | | |
| 44 | AIRCRAFT HANDLING GEAR | | | | | | |
| 45 | LOAD HANDLING GEAR | | | | | | 1400 |
| 46 | ATO GEAR | | | | | | |
| 47 | | | | | | | |
| 48 | | | | | | | |
| 49 | | | | | | | |
| 50 | | | | | | | |
| 51 | | | | | | | |
| 52 | | | | | | | |
| 53 | | | | | | | |
| 54 | MANUFACTURING VARIATION | | | | | | |
| 55 | | | | | | | |
| 56 | | | | | | | |
| 57 | TOTAL—WEIGHT EMPTY—PAGES 2, 3 AND 4 | | | | | (Fixed Gear) | 20,570 |

(Retract Gear) 21,080

NAME

DATE

## SUMMARY WEIGHT STATEMENT
USEFUL LOAD GROSS WEIGHT

PAGE

MODEL 395 Config. 3

REPORT

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | LOAD CONDITION | | | Transport | | Heavy Lift | |
| 2 | | | | 12 ton | | 20 ton | |
| 3 | CREW—NO. 3 | | | 600 | | 600 | |
| 4 | PASSENGERS—NO. | | | | | | |
| 5 | FUEL LOCATION | TYPE | GALS | | | | |
| 6 | UNUSABLE Fuselage | JP-4 | | 100 | | 100 | |
| 7 | INTERNAL | | | 6884 | | 4131 | |
| 8 | | | | | | | |
| 9 | | | | | | | |
| 10 | | | | | | | |
| 11 | EXTERNAL | | | | | | |
| 12 | | | | | | | |
| 13 | | | | | | | |
| 14 | | | | | | | |
| 15 | BOMB BAY | | | | | | |
| 16 | | | | | | | |
| 17 | | | | | | | |
| 18 | | | | | | | |
| 19 | OIL | | | | | | |
| 20 | UNUSABLE | | | | | | |
| 21 | ENGINE | | | 30 | | 30 | |
| 22 | | | | | | | |
| 23 | | | | | | | |
| 24 | | | | | | | |
| 25 | BAGGAGE - Crew Kits | | | 50 | | 50 | |
| 26 | CARGO - Payload | | | 24,000 | | 40,000 | |
| 27 | | | | | | | |
| 28 | ARMAMENT | | | | | | |
| 29 | GUNS—LOCATION TYPE** | QUANTITY | CALIBER | | | | |
| 30 | | | | | | | |
| 31 | | | | | | | |
| 32 | | | | | | | |
| 33 | | | | | | | |
| 34 | AMM | | | | | | |
| 35 | | | | | | | |
| 36 | | | | | | | |
| 37 | | | | | | | |
| 38 | BOMB INSTL* | | | | | | |
| 39 | BOMBS | | | | | | |
| 40 | | | | | | | |
| 41 | TORPEDO INSTL* | | | | | | |
| 42 | TORPEDOES | | | | | | |
| 43 | | | | | | | |
| 44 | ROCKET INSTL* | | | | | | |
| 45 | ROCKETS | | | | | | |
| 46 | | | | | | | |
| 47 | EQUIPMENT—PYROTECHNICS | | | | | | |
| 48 | —PHOTOGRAPHIC | | | | | | |
| 49 | | | | | | | |
| 50 | —OXYGEN | | | | | | |
| 51 | | | | | | | |
| 52 | —MISCELLANEOUS | | | | | | |
| 53 | | | | | | | |
| 54 | | | | | | | |
| 55 | USEFUL LOAD | | | 31,664 | | 44,911 | |
| 56 | Weight Empty (Fixed Gear) | | | 20,570 | | 20,570 | |
| 57 | GROSS WEIGHTS—PAGES 3-6 | | | 52,234 | | 65,481 | |

* If not specified as Weight Empty.

** Fixed, Flexible, etc.

NAME____________
DATE____________

PAGE____________
MODEL 395 Config. 4
REPORT____________

# SUMMARY WEIGHT STATEMENT
# ROTORCRAFT ONLY

**ESTIMATED ~~— CALCULATED — ACTUAL~~**

**(Cross out those not applicable)**

## HOT CYCLE HEAVY-LIFT HELICOPTER
## CONFIGURATION 4

CONTRACT____________
ROTORCRAFT, GOVERNMENT NUMBER____________
ROTORCRAFT, CONTRACTOR NUMBER____________
MANUFACTURED BY Hughes Tool Company - Aircraft Division

| | | Main | Auxiliary |
|---|---|---|---|
| Engine | Manufactured by | General Electric | |
| | Model | GE-1 | |
| | Number | 2 | |
| Propeller | Manufactured by | | |
| | Model | | |
| | Number | | |

NAME________________
DATE________________

## ROTORCRAFT
## SUMMARY WEIGHT STATEMENT
## WEIGHT EMPTY

PAGE________________
MODEL 395 Config. 4
REPORT________________

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | | | | | | | |
| 2 | ROTOR GROUP | | | | | | 5440 |
| 3 | BLADE ASSEMBLY | | | | | 3806 | |
| 4 | HUB | | | | | 1634 | |
| 5 | HINGE AND BLADE RETENTION | | | | | | |
| 6 | | FLAP | PING | | | | |
| 7 | | LEAD | LAG | | | | |
| 8 | | PITC | H | | | | |
| 9 | | FOLD | ING | | | | |
| 10 | WING GROUP | | | | | | |
| 11 | WING PANELS—BASIC STRUCTURE | | | | | | |
| 12 | CENTER SECTION—BASIC STRUCTURE | | | | | | |
| 13 | INTERMEDIATE PANEL—BASIC STRUCTURE | | | | | | |
| 14 | OUTER PANEL—BASIC STRUCTURE—INCL TIPS | | | LBS | | | |
| 15 | SECONDARY STRUC—INCL FOLD MECH | | | LBS | | | |
| 16 | AILERONS—INCL BALANCE WTS | | | LBS | | | |
| 17 | FLAPS | | | | | | |
| 18 | —TRAILING EDGE | | | | | | |
| 19 | —LEADING EDGE | | | | | | |
| 20 | SLATS | | | | | | |
| 21 | SPOILERS | | | | | | |
| 22 | | | | | | | |
| 23 | TAIL GROUP | | | | | | 1068 |
| 24 | TAIL ROTOR | | | | | 70 | |
| 25 | —BLADES | | | | | | |
| 26 | —HUB | | | | | | |
| 27 | STABILIZER—BASIC STRUCTURE | | | | | 998 | |
| 28 | FINS—BASIC STRUCTURE—INCL DORSAL | | | LBS | | | |
| 29 | SECONDARY STRUCTURE—STABILIZER AND FINS | | | | | | |
| 30 | ELEVATOR—INCL BALANCE WEIGHT | | | LBS | | | |
| 31 | RUDDER—INCL BALANCE WEIGHT | | | LBS | | | |
| 32 | | | | | | | |
| 33 | BODY GROUP | | | | | | 3575 |
| 34 | FUSELAGE OR HULL—BASIC STRUCTURE | | | | | | |
| 35 | BOOMS—BASIC STRUCTURE | | | | | | |
| 36 | SECONDARY STRUCTURE—FUSELAGE OR HULL | | | | | | |
| 37 | —BOOMS | | | | | | |
| 38 | —DOORS, PANELS & MISC | | | | | | |
| 39 | | | | | | | |
| 40 | | | | | | | |
| 41 | ALIGHTING GEAR—LAND TYPE | | | | | | 2852 |
| 42 | LOCATION * | ROLLING | STRUCT | CONTROLS | | | |
| 43 | | ASSEMBLY | | | | | |
| 44 | | | | | | | |
| 45 | | | | | | | |
| 46 | | | | | | | |
| 47 | | | | | | | |
| 48 | | | | | | | |
| 49 | | | | | | | |
| 50 | ALIGHTING GEAR GROUP—WATER TYPE | | | | | | |
| 51 | LOCATION | FLOATS | STRUTS | CONTROLS | | | |
| 52 | | | | | | | |
| 53 | | | | | | | |
| 54 | | | | | | | |
| 55 | | | | | | | |
| 56 | | | | | | | |
| 57 | | | | | | | |

* Wheels, Brakes, Tires, Tubes and Air.

NAME
DATE

**ROTORCRAFT**
**SUMMARY WEIGHT STATEMENT**
**WEIGHT EMPTY—Continued**

PAGE
MODEL 395 Config. 4
REPORT

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | | | | | | | |
| 2 | FLIGHT CONTROLS GROUP | | | | | | 1454 |
| 3 | COCKPIT CONTROLS | | | | | 33 | |
| 4 | AUTOMATIC STABILIZATION | | | | | | |
| 5 | SYSTEM CONTROLS—ROTOR NON ROTATING | | | | | 990 | |
| 6 | ROTATING | | | | | 269 | |
| 7 | —FIXED WING | | | | | | |
| 8 | Intermediate Linkages | | | | | 162 | |
| 9 | | | | | | | |
| 10 | ENGINE SECTION OR NACELLE GROUP | | | | | | 460 |
| 11 | INBOARD | | | | | | |
| 12 | CENTER | | | | | | |
| 13 | OUTBOARD | | | | | | |
| 14 | DOORS, PANELS AND MISC | | | | | | |
| 15 | | | | | | | |
| 16 | PROPULSION GROUP | | | | | | 2479 |
| 17 | | | X AUXILIARY | X | X | MAIN X | |
| 18 | ENGINE INSTALLATION | | | | | 1627 | |
| 19 | ENGINE | | | | | ** | |
| 20 | TIP BURNERS | | | | | | |
| 21 | LOAD COMPRESSOR | | | | | | |
| 22 | REDUCTION GEAR BOX, ETC | | | | | | |
| 23 | ACCESSORY GEAR BOXES AND DRIVES | | | | | | |
| 24 | SUPERCHARGER—FOR TURBOS | | | | | | |
| 25 | AIR INDUCTION SYSTEM | | | | | * | |
| 26 | EXHAUST SYSTEM | | | | | * | |
| 27 | COOLING SYSTEM | | | | | | |
| 28 | LUBRICATING SYSTEM | | | | | 64 | |
| 29 | TANKS | | | | | | |
| 30 | BACKING BD, TANK SUP & PADDING | | | | | | |
| 31 | COOLING INSTALLATION | | | | | | |
| 32 | PLUMBING, ETC | | | | | | |
| 33 | FUEL SYSTEM | | | | | * | |
| 34 | TANKS—UNPROTECTED | | | | | | |
| 35 | —PROTECTED | | | | | | |
| 36 | BACKING BD, TANK SUP & PADDING | | | | | | |
| 37 | PLUMBING, ETC | | | | | | |
| 38 | WATER INJECTION SYSTEM | | | | | | |
| 39 | ENGINE CONTROLS | | | | | 60 | |
| 40 | STARTING SYSTEM | | | | | 70 | |
| 41 | PROPELLER INSTALLATION | | | | | | |
| 42 | DRIVE SYSTEM | | | | | 658 | |
| 43 | GEAR BOXES | | | | | | |
| 44 | LUBE SYSTEM | | | | | | |
| 45 | CLUTCH AND MISC | | | | | | |
| 46 | TRANSMISSION DRIVE | | | | | | |
| 47 | ROTOR SHAFT | | | | | | |
| 48 | JET DRIVE | | | | | | |
| 49 | | | | | | | |
| 50 | | | | | | | |
| 51 | | | | | | | |
| 52 | AUXILIARY POWER PLANT GROUP | | | | | | 160 |
| 53 | | | | | | | |
| 54 | | | | | | | |
| 55 | | | | | | | |
| 56 | | | | | | | |
| 57 | | | | | | | |

*Weight for these items included in engine installation weight

**Engine weight confidential

NAME__________
DATE__________

## ROTORCRAFT
## SUMMARY WEIGHT STATEMENT
## WEIGHT EMPTY—Continued

PAGE__________
MODEL 395 Config. 4
REPORT__________

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | | | | | | | |
| 2 | | | | | | | 180 |
| 3 | | | | | | | |
| 4 | INSTRUMENT AND NAVIGATIONAL EQUIPMENT GROUP | | | | | | |
| 5 | INSTRUMENTS | | | | | | |
| 6 | NAVIGATIONAL EQUIPMENT | | | | | | |
| 7 | | | | | | | |
| 8 | | | | | | | |
| 9 | HYDRAULIC AND PNEUMATIC GROUP | | | | | | 735 |
| 10 | HYDRAULIC | | | | | | |
| 11 | PNEUMATIC | | | | | | |
| 12 | | | | | | | |
| 13 | | | | | | | |
| 14 | ELECTRICAL GROUP | | | | | | 752 |
| 15 | A C SYSTEM | | | | | | |
| 16 | D C SYSTEM | | | | | | |
| 17 | | | | | | | |
| 18 | | | | | | | |
| 19 | ELECTRONICS GROUP | | | | | | 150 |
| 20 | EQUIPMENT | | | | | | |
| 21 | INSTALLATION | | | | | | |
| 22 | | | | | | | |
| 23 | | | | | | | |
| 24 | ARMAMENT GROUP—INCL GUNFIRE PROTECTION | | | | LBS | | |
| 25 | | | | | | | |
| 26 | FURNISHINGS AND EQUIPMENT GROUP | | | | | | 300 |
| 27 | ACCOMMODATIONS FOR PERSONNEL | | | | | | |
| 28 | MISCELLANEOUS EQUIPMENT | X INCL | | LBS | BALLASTX | | |
| 29 | FURNISHINGS | | | | | | |
| 30 | EMERGENCY EQUIPMENT | | | | | | |
| 31 | | | | | | | |
| 32 | | | | | | | |
| 33 | | | | | | | |
| 34 | AIR CONDITIONING AND ANTI-ICING EQUIPMENT | | | | | | 100 |
| 35 | AIR CONDITIONING | | | | | | |
| 36 | ANTI-ICING | | | | | | |
| 37 | | | | | | | |
| 38 | | | | | | | |
| 39 | PHOTOGRAPHIC GROUP | | | | | | |
| 40 | EQUIPMENT | | | | | | |
| 41 | INSTALLATION | | | | | | |
| 42 | | | | | | | |
| 43 | AUXILIARY GEAR GROUP | | | | | | |
| 44 | AIRCRAFT HANDLING GEAR | | | | | | |
| 45 | LOAD HANDLING GEAR | | | | | | 1400 |
| 46 | ATO GEAR | | | | | | |
| 47 | | | | | | | |
| 48 | | | | | | | |
| 49 | | | | | | | |
| 50 | | | | | | | |
| 51 | | | | | | | |
| 52 | | | | | | | |
| 53 | | | | | | | |
| 54 | MANUFACTURING VARIATION | | | | | | |
| 55 | | | | | | | |
| 56 | | | | | | | |
| 57 | TOTAL—WEIGHT EMPTY—PAGES 2, 3 AND 4 | | | | | | 21,105 |

NAME________________
DATE________________

**SUMMARY WEIGHT STATEMENT**
**USEFUL LOAD GROSS WEIGHT**

PAGE________________
MODEL 395 Config. 4
REPORT________________

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | LOAD CONDITION | | | Transport | | Heavy Lift | |
| 2 | | | | 12 ton | | 20 ton | |
| 3 | CREW—NO. | | | 600 | | 600 | |
| 4 | PASSENGERS—NO. | | | | | | |
| 5 | FUEL LOCATION | TYPE | GALS | | | | |
| 6 | UNUSABLE Fuselage | JP-4 | | 100 | | 100 | |
| 7 | INTERNAL | | | 8272 | | 4312 | |
| 8 | | | | | | | |
| 9 | | | | | | | |
| 10 | | | | | | | |
| 11 | EXTERNAL | | | | | | |
| 12 | | | | | | | |
| 13 | | | | | | | |
| 14 | | | | | | | |
| 15 | BOMB BAY | | | | | | |
| 16 | | | | | | | |
| 17 | | | | | | | |
| 18 | | | | | | | |
| 19 | OIL | | | 30 | | 30 | |
| 20 | UNUSABLE | | | | | | |
| 21 | ENGINE | | | | | | |
| 22 | | | | | | | |
| 23 | | | | | | | |
| 24 | | | | | | | |
| 25 | BAGGAGE - Crew Kits | | | 50 | | 50 | |
| 26 | CARGO - Payload | | | 24,000 | | 40,000 | |
| 27 | | | | | | | |
| 28 | ARMAMENT | | | | | | |
| 29 | GUNS—LOCATION TYPE** | QUANTITY | CALIBER | | | | |
| 30 | | | | | | | |
| 31 | | | | | | | |
| 32 | | | | | | | |
| 33 | | | | | | | |
| 34 | AMM | | | | | | |
| 35 | | | | | | | |
| 36 | | | | | | | |
| 37 | | | | | | | |
| 38 | BOMB INSTL.* | | | | | | |
| 39 | BOMBS | | | | | | |
| 40 | | | | | | | |
| 41 | TORPEDO INSTL.* | | | | | | |
| 42 | TORPEDOES | | | | | | |
| 43 | | | | | | | |
| 44 | ROCKET INSTL.* | | | | | | |
| 45 | ROCKETS | | | | | | |
| 46 | | | | | | | |
| 47 | EQUIPMENT—PYROTECHNICS | | | | | | |
| 48 | —PHOTOGRAPHIC | | | | | | |
| 49 | | | | | | | |
| 50 | —*OXYGEN | | | | | | |
| 51 | | | | | | | |
| 52 | —MISCELLANEOUS | | | | | | |
| 53 | | | | | | | |
| 54 | | | | | | | |
| 55 | USEFUL LOAD | | | 33,052 | | 45,092 | |
| 56 | Weight Empty | | | 21,105 | | 21,105 | |
| 57 | GROSS WEIGHTS—PAGES 2-6 | | | 54,157 | | 66,197 | |

* If not specified as Weight Empty.

** Fixed, Flexible, etc.

NAME____________
DATE____________

ROTOR GROUP
BLADE ASSEMBLY

PAGE____________
MODEL____________
REPORT____________

| 1 | | NACA Section | | Trans. | Root | | |
|---|---|---|---|---|---|---|---|
| 2 | | .0014 | .0018 | Sect. | Sect. | | |
| 3 | | | | | | | |
| 4 | FRONT SPAR—UPPER CAP | | | | | | |
| 5 | —LOWER CAP | | | | | | |
| 6 | —WEB & STIFFENERS | | | | | | |
| 7 | —JOINTS, SPLICES & FASTNR | | | | | | |
| 8 | | | | | | | |
| 9 | INTER SPAR—UPPER CAP @25% Chord | 83.7 | 526.5 | 85.8 | | | |
| 10 | —LOWER CAP | | | | | | |
| 11 | —WEB & STIFFENERS | | | | | | |
| 12 | —JOINTS, SPLICES & FASTNR | | | | | | |
| 13 | | | | | | | |
| 14 | REAR SPAR—UPPER CAP | | | | | | |
| 15 | —LOWER CAP | | | | | | |
| 16 | —WEB & STIFFENERS | | | | | | |
| 17 | —JOINTS, SPLICES & FASTNR | | | | | | |
| 18 | | | | | | | |
| 19 | INTERSPAR STRUCTURE | | | | | | |
| 20 | COVERING & STIFFENERS | 186.9 | 503.1 | 39.0 | | | |
| 21 | RIBS | 40.2 | 124.9 | 112.2 | | | |
| 22 | FILLER | | | | | | |
| 23 | JOINTS, SPLICES & FASTNR | | | | | | |
| 24 | Flexures | 37.5 | 96.0 | 6.0 | | | |
| 25 | | | | | | | |
| 26 | LEADING EDGE | | | | | | |
| 27 | LEADING EDGE MEMBER | | | | | | |
| 28 | COVERING & STIFFENERS | | | | | | |
| 29 | RIBS | | | | | | |
| 30 | FILLER | | | | | | |
| 31 | JOINTS, SPLICES & FASTNR | | | | | | |
| 32 | | | | | | | |
| 33 | TRAILING EDGE | | | | | | |
| 34 | TRAILING EDGE MEMBER | 3.9 | 12.8 | | | | |
| 35 | COVERING & STIFFENERS | 39.0 | 103.5 | 14.40 | | | |
| 36 | RIBS | 8.1 | 24.2 | | | | |
| 37 | FILLER | | | | | | |
| 38 | JOINTS, SPLICES & FASTNR | 1.1 | 2.6 | | | | |
| 39 | | | | | | | |
| 40 | TIPS—IF NOT INTEGRAL + Cascade | 36.0 | | | | | |
| 41 | INTERNAL DUCT STRUCTURE + Insul. | 124.8 | 327.3 | | | | |
| 42 | Articulated Duct | | | | 204.0 | | |
| 43 | | | | | | | |
| 44 | Damper | | | 198.0 | | | |
| 45 | Damper Arm | | | | 13.5 | | |
| 46 | Balance Section for Damper | | | 14.4 | | | |
| 47 | BALANCE WEIGHTS for 23 to 28% | 124.7 | 155.8 | | | | |
| 48 | Lead-Lag Flexure | | | 63.9 | | | |
| 49 | TRIM TAB & FITTING | | | | | | |
| 50 | ROOT END ATTACHMENT | | | | 78.6 | | |
| 51 | FITTINGS - Torque Box | | | | 277.5 | | |
| 52 | FASTENERS | | | | | | |
| 53 | EXTERIOR FINISH | | | | | | |
| 54 | Fairing | | | | 48.0 | | |
| 55 | Droop Stops | | | | 54.6 | | |
| 56 | TOTAL—BLADE ASSEMBLY | 685.9 | 1876.7 | 533.7 | 676.2 | | |
| 57 | | | | | | | 3772.5 |

NAME________ 
DATE________

**ROTOR GROUP**
**HINGES AND HUB**

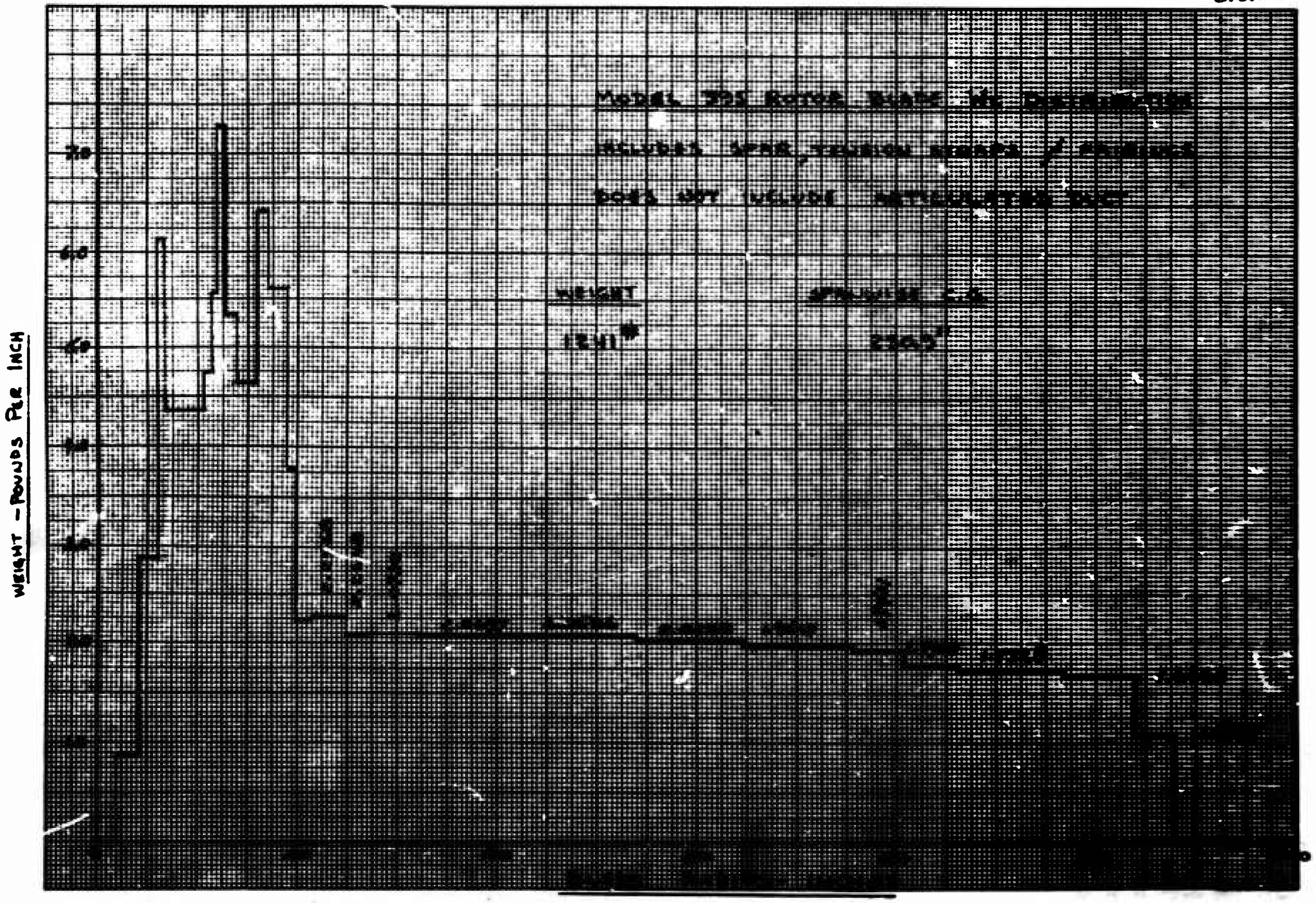
PAGE________
MODEL________
REPORT________

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1 | | X HUB X | X | HINGE & BLADE RETENTION | | | X |
| 2 | | | FLAPPING | LEAD—LAG | PITCH | FOLDING | |
| 3 | | | | | | | |
| 4 | HOUSING - Hub | 369.0 | | | | | |
| 5 | GIMBAL RING | | | | | | |
| 6 | YOKE | | | | | | |
| 7 | UNIVERSAL JOINTS | | | | | | |
| 8 | Support-Shaft to Hub | 37.0 | | | | | |
| 9 | | | | | | | |
| 10 | | | | | | | |
| 11 | | | | | | | |
| 12 | Bearings-Blade Feathering | 17.0 | | | | | |
| 13 | BEARINGS (2) - Main | 288.0 | | | | | |
| 14 | SEALS, SPACERS & RETAINERS | 153.0 | | | | | |
| 15 | BLADE GRIPS | | | | | | |
| 16 | BALANCE WEIGHTS | | | | | | |
| 17 | STOPS and Supports | 27.0 | | | | | |
| 18 | LUB SYS INCL LBS OIL | | | | | | |
| 19 | FAIRINGS & DUST COVERS | 24.0 | | | | | |
| 20 | GREASE | | | | | | |
| 21 | | | | | | | |
| 22 | | | | | | | |
| 23 | | | | | | | |
| 24 | | | | | | | |
| 25 | FITTINGS | | | | | | |
| 26 | PINS | | | | | | |
| 27 | LINKS | | | | | | |
| 28 | DAMPERS OR RESTRAINERS | | | | | | |
| 29 | DRAG BRACE | | | | | | |
| 30 | TENSION STRAP ASSEMBLY | 121.5 | | | | | |
| 31 | | | | | | | |
| 32 | | | | | | | |
| 33 | Shaft-Fixed | 286.0 | | | | | |
| 34 | SHAFTS - Rotating | 251.0 | | | | | |
| 35 | PITCH ARMS | | | | | | |
| 36 | | | | | | | |
| 37 | | | | | | | |
| 38 | BLADE FOLD—MECHANISM | | | | | | |
| 39 | —ACTUATORS | | | | | | |
| 40 | —CONTROLS | | | | | | |
| 41 | —LOCKS | | | | | | |
| 42 | *—PLUMBING | | | | | | |
| 43 | *—CIRCUITRY | | | | | | |
| 44 | —SUPPORTS | | | | | | |
| 45 | | | | | | | |
| 46 | | | | | | | |
| 47 | | | | | | | |
| 48 | | | | | | | |
| 49 | | | | | | | |
| 50 | FASTENERS and Insulation | 60.0 | | | | | |
| 51 | | | | | | | |
| 52 | | | | | | | |
| 53 | EXTERIOR FINISH | | | | | | |
| 54 | | | | | | | |
| 55 | | | | | | | |
| 56 | TOTAL—HINGES AND HUB | 1633.5 | | | | | |
| 57 | TOTAL—ROTOR GROUP—PAGES 2 and 3 | | | | | | 5439 |

* Main distribution point to actuating unit.

8/3/66

8/3/66


MODEL 395 ROTOR BLADE Wt. DISTRIBUTION
INCLUDES SPAR, TENSION STRAPS / FAIRINGS
DOES NOT INCLUDE [illegible] DUCT
WEIGHT
1241#
SPANWISE C.G.
2305"
WEIGHT - POUNDS PER INCH
7.0
6.0
5.0
4.0
3.0
2.0
1.0

## DETAILED WEIGHT CALCULATIONS

The following detailed weight calculations were made using design layouts and structural analysis of the blade, hub, and associated rotor parts. The weight of the blade nonbending material consisting of skin panels, ribs, and chordwise balance material was calculated; then the spar area required for the resulting centrifugal loads was determined and its weight calculated.

WEIGHT CALCULATIONS FOR BLADE SECTION (.75R TO TIP)

NACA 0014 SECTION (REF. DRAWING 395-0919)
(20" SEGMENT)

| **AFT SEGMENT** (AL. ALLOY) | W | WT/IN | X-L.E | $\bar{w}x$ |
|---|---|---|---|---|
| SKIN 67.8×20×.016×.101 | 2.195 | .1028 | 45.93 | 4.7148 |
| CHANNEL 9.3×20×.012×.101 | .225 | .0112 | 27.50 | .3080 |
| DOUBLER .8×20×.016×.101 | .026 | .0013 | 27.50 | .0358 |
| DOUBLER .8×20×.012×.101 | .019 | .0010 | 29.50 | .0295 |
| RIBS (4) 67.8×.012×.101×4 | .426 | .0213 | 36.60 | .7796 |
| END COVER 4.2×25.8×.101×.003 | .033 | .0017 | 36.60 | .0622 |
| HARDWARE (SAME AS XV-9A) | .056 | .0028 | 27.50 | .0770 |
| TOTAL AFT SEGMENT | | .1481$^{\#4}$ | 40.54 | 6.0044 |
| **MAIN SEGMENT** | | | | |
| SKIN (TITANIUM) | | | | |
| INNER 58.6×18×.012×.17 | 2.152 | .1076 | | |
| (.75-.8) OUTER 58.6×20×.008×.17 | (1.594) | (.0799) | | |
| (.8-.9) OUTER 58.6×20×.007×.17 | (1.395) | (.0698) | | |
| (.9-1.0) OUTER 58.6×20×.006×.17 | (1.195) | (.0598) | | |
| CORRUGATION 54.0×16.5×.83×3.333×.007×.17 | 2.933 | .1466 | | |
| TOTAL SKIN (.75-.8 %R) | | .3339$^{\#1}$ | 13.7 | 4.5744 |
| " " (.8-.9 %R) | | .3240$^{\#2}$ | 13.7 | 4.4388 |
| " " (.9-1.0 %R) | | .3140$^{\#3}$ | 13.7 | 4.3018 |
| AFT CLOSURE WEB (HI TEMP METAL @.3#/in³) | | | | |
| (.75-.8) WEB 9.0×20×.020×.3 | (1.080) | (.0540)$^{\#1}$ | 27.25 | 1.4715 |
| (.8-1.0) WEB 9.0×20×.018×.3 | (.972) | (.0486)$^{\#2,3}$ | 27.25 | 1.3244 |
| MID RIB (CRES) 113.8in²×.020×.286 | .651 | .0326 | 13.0 | .4238 |
| RIB DOUBLER (CRES) 30.9in²×.020×.286 | .177 | .0088 | 17.0 | .1496 |
| FWD STIFFENER (CRES) 5.8in²×.052×.286 | .086 | .0043 | 3.4 | .0146 |
| AFT STIFFENER (CRES) 7.0in²×.052×.286 | .104 | .0052 ✓ | 26.3 | .1368 |
| FWD DUCT STRAP (CRES) 17.5in²×.035×.286 | .175 | .0087 | 13.5 | .1174 |
| AFT DUCT STRAP (CRES) 18.7in²×.035×.286 | .187 | .0094 | 16.6 | .1560 |
| TOTAL RIB | | .0690$^{\#}$ | 14.67 | .9982 |

0704

WEIGHT CALCULATIONS FOR BLADE
SECTION .75R TO TIP (CONT'D)

| | W | WT/IN | $\bar{X}_{L.E.}$ | $\bar{W}X$ |
|---|---|---|---|---|
| BAL. WT. CLIPS (4) .8in² × .020 × .286 × 4 | .018 | .0009# | 1.0 | .0009 |
| 85% WEB 8×20.5 × .020 × .286 −4.50×2 | .756 | .0378# | 15.4 | .5821 |
| OUTPLATE STRIP 1.95 × 18 × .040 × .286 | .360 | .0180# | 15.15 | .2727 |
| UPPER CHANNEL 2.3 × 18 × .020 × .286 | .237 | .0118# | 15.15 | .1788 |
| LOWER CHANNEL (SAME AS UPPER CHANNEL) | .237 | .0118# | 15.16 | .1788 |
| END RIB - FWD 19.84 × .020 × .286 × 2 | .227 | .0113# | 2.8 | .0316 |
| END RIB - MID 15.73 × .020 × .286 × 4 | .360 | .0180# | 15.6 | .2808 |
| FLEXURE 4 × 61.5 × .025 × .30 | 1.845 | .0922# | 15.12 | 1.3941 |
| FLEX. MAT 1.3 × 3.2 × .025 × .3 | .031 | .0016# | 26.8 | .0429 |
| | | | | |
| FWD DUCT (INCONEL) | | | | |
| DUCT 32.4 × 20 × .010 × .298 | 1.931 | .0966 | | |
| BELLOWS 28.8 × 3.5 × .020 × .298 | .601 | .0300 | | |
| INSULATION .0012 × 28.8 × 20 | .691 | .0346 | | |
| STIFFENERS 28.8 × .4 × .012 × .298 × 4 | .165 | .0082 | | |
| TIE CHANNEL 1.3 × 4.8 × .030 × .286 | .054 | .0027 | | |
| | | .1721# | 9.0 | 1.5489 |
| AFT DUCT (INCONEL) | | | | |
| DUCT 32.0 × 20 × .010 × .298 | 1.907 | .0954 | | |
| BELLOWS 26.8 × 3.5 × .020 × .298 | .559 | .0280 | | |
| INSULATION .0012 × 26.8 × 20 | .643 | .0322 | | |
| STIFFENERS 26.8 × .4 × .012 × .298 × 4 | .153 | .0077 | | |
| TIE CHANNEL 1.3 × 4.8 × .030 × .286 | .054 | .0027 | | |
| | | .1660# | 21.0 | 3.4860 |

| | | | |
|---|---|---|---|
| TOTAL BLADE SECTION - NO SPAR (.75-.8%R) (405 - 432) | 1.1465 | 18.36 | 21.0461 |
| (.8-.9%R)(432 - 486) | 1.1312 | 18.34 | 20.7634 |
| (.9-1.0%R)(486-540) | 1.1212 | 18.40 | 20.6264 |

## WEIGHT CALCULATIONS FOR BLADE CONSTANT SECTION TO .75R

NACA 0018 SECTION (REF. DRAWING 395-0919)
(20" SEGMENT)

| | | W | $\bar{w}$ | X-LP | $\bar{w}$X |
|---|---|---|---|---|---|
| **AFT SEGMENT** (AL ALLOY) | | | | | |
| | SKIN 68.4×20×.016×.101 | 2.189 | .1094 | 43.33 | 4.7408 |
| | CHANNEL 11.5×20×.012×.101 | .279 | .0139 | 28.5 | .3995 |
| | DOUBLER .8×20×.016×.101 | .026 | .0013 | 27.5 | .0358 |
| | DOUBLER .8×20×.012×.101 | .019 | .0010 | 27.5 | .0275 |
| | RIBS(4) 99.17×.012×.101×4 | .481 | .0240 | 36.6 | .8784 |
| | END COVER 5.35×2.58×.101×.003 | .042 | .0021 | 36.6 | .0769 |
| | HARDWARE | .056 | .0028 | 27.5 | .0770 |
| | TOTAL AFT SEGMENT | | .1544$^{\#}$ | 40.26 | 6.2154 |
| MAIN SEGMENT | | | | | |
| | SKIN (TITANIUM) | | | | |
| | INNER 60.0×.012×.17×18 | 2.203 | .1102 | | |
| (.2-.6) | OUTER 60.0×.011×.17×20 | (2.244)$^{\#1}$ | (.1122) | | |
| (.6-.75) | OUTER 60.0×.008×.17×20 | (1.632)$^{\#2}$ | (.0816) | | |
| | CORRUGATION .83×5.333×55.4×16.5×.0072×.17 | 3.009 | .1504 | | |
| | TOTAL SKIN (.2-.6) | | .3728$^{\#1}$ | 13.7 | 5.1074 |
| | " " (.6-.75) | | .3422$^{\#2}$ | 13.7 | 4.6881 |
| AFT CLOSURE WEB | | | | | |
| (.2-.6) | WEB 11.5×20×.022×.3 | (1.518) | (.0759)$^{\#1}$ | 27.25 | 2.0683 |
| (.6-.75) | WEB 11.5×20×.021×.3 | (1.449) | (.0724)$^{\#2}$ | 27.25 | 1.9729 |
| | MID RIB (CRES) 150.1 in$^2$×.020×.286 | .858 | .0429 | 13.0 | .5577 |
| | RIB DOUBLER (CRES) 42.4 in$^2$×.020×.286 | .242 | .0121 | 17.0 | .2057 |
| | FWD STIFF (CRES) 7.2 in$^2$×.052×.286 | .107 | .0054 | 3.0 | .0164 |
| | AFT STIFF (CRES) 9.2×.052×.286 | .137 | .0068 | 26.3 | .1788 |
| | FWD DUCT STRAP (CRES) 20.2 in$^2$×.035×.286 | .202 | .0101 | 13.2 | .1333 |
| | AFT DUCT STRAP (CRES) 22.0 in$^2$×.035×.286 | .220 | .0110 | 16.9 | .1859 |
| | TOTAL RIB | | .0883$^{\#}$ | 14.49 | 1.2798 |

0704

WEIGHT CALCULATIONS FOR BLADE
CONSTANT SECTION TO .75R (CONT'D)

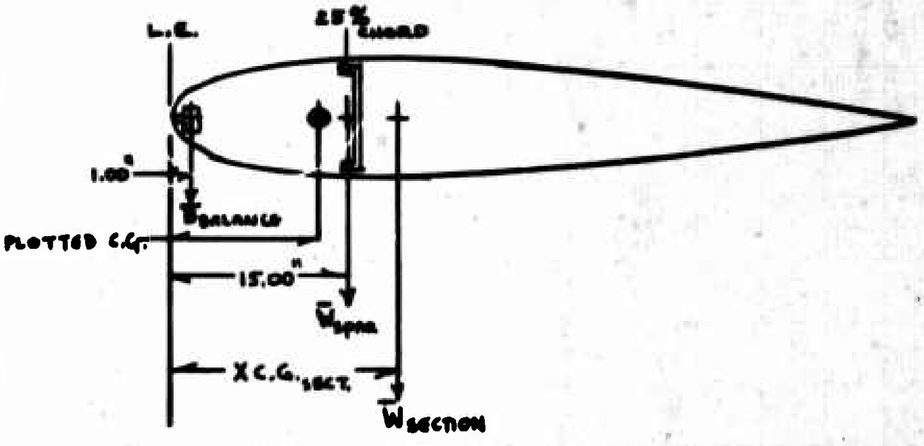
| | $w$ | $\bar{w}$ | $x$-LE | $\bar{w}x$ |
|---|---|---|---|---|
| BAL. WT. CLIPS (4) $.8in^2 \times .020 \times .286 \times 4$ | .015 | .0009* | 1.0 | .0009 |
| 25% WEB $10.4 \times 20.5 \times .020 \times .286 - 4.80 \times 2$ | 1.038 | .0519* | 15.4 | .7993 |
| NUTPLATE STRIP $1.75 \times 18 \times .040 \times .286$ | .360 | .0180* | 15.15 | .2727 |
| UPPER CHANNEL $3.2 \times 18 \times .020 \times .286$ | .329 | .0165* | 15.5 | .2558 |
| LOWER CHANNEL (SAME AS UPPER CHANNEL) | .329 | .0165* | 15.5 | .2558 |
| END RIB-FWD $25.1 \times .020 \times .286 \times 2$ | .287 | .0144* | 2.8 | .0403 |
| END RIB-MID $27.2 \times .020 \times .286 \times 4$ | .622 | .0311* | 15.6 | .4852 |
| FLEXURE $4 \times 68 \times .025 \times .3$ | 2.040 | .1020* | 15.12 | 1.5422 |
| FLEXURE HAT (SAME AS 0014) | .031 | .0016* | 26.8 | .0429 |
| | | | | |
| FWD DUCT (INCONEL) | | | | |
| DUCT $36.6 \times 20 \times .010 \times .298$ | 2.181 | .1090 | | |
| BELLOWS $32 \times 3.5 \times .020 \times .298$ | .668 | .0334 | | |
| INSULATION (MIN-K WRAP) $.0012 \times 32 \times 20$ | .768 | .0384 | | |
| STIFFENERS $32 \times .4 \times .012 \times .298 \times 4$ | .183 | .0092 | | |
| TIE CHANNEL $1.3 \times 4.8 \times .030 \times .286$ | .054 | .0027 | | |
| | | .1927* | 8.0 | 1.5416 |
| | | | | |
| AFT DUCT (INCONEL) | | | | |
| DUCT $3.1416 \times 9.3 \times 20 \times .010 \times .298$ | 1.741 | .0870 | | |
| BELLOWS $30 \times 3.5 \times .020 \times .298$ | .626 | .0313 | | |
| INSULATION $.0012 \times 30 \times 20$ | .720 | .0360 | | |
| STIFFENERS $30 \times .4 \times .012 \times .298 \times 4$ | .172 | .0086 | | |
| TIE CHANNEL $1.3 \times 4.8 \times .030 \times .286$ | .054 | .0027 | | |
| | | .1656* | 21.2 | 3.5107 |
| | | | | |
| TOTAL BLADE SECTION-NO SPAR (.2-.6)(108-324) | | 1.3026 | 17.98 | 23.4183 |
| (.6-.75)(324-405) | | 1.2685 | 18.06 | 22.9036 |

## Calculations for Blade Leading Edge Balance Weight

Each typical blade section is balanced about the plotted chordwise center of gravity for that section. The plotted center of gravity was selected from the following graph at the midpoint (spanwise) of the section.

As an example:

$\bar{M}$ = chordwise moment of item (about Leading Edge)
$\bar{W}$ = unit weight per inch of item
$\bar{B}$ = unit balance weight per inch



$$\bar{M}_{sect} = \bar{W}_{sect} \times X\,C.G._{section}$$
$$\bar{M}_{spar} = \bar{W}_{spar} \times 15.00$$
$$\bar{M}_{balance} = \bar{B}_{balance} \times 1.00 = \bar{B}$$

and

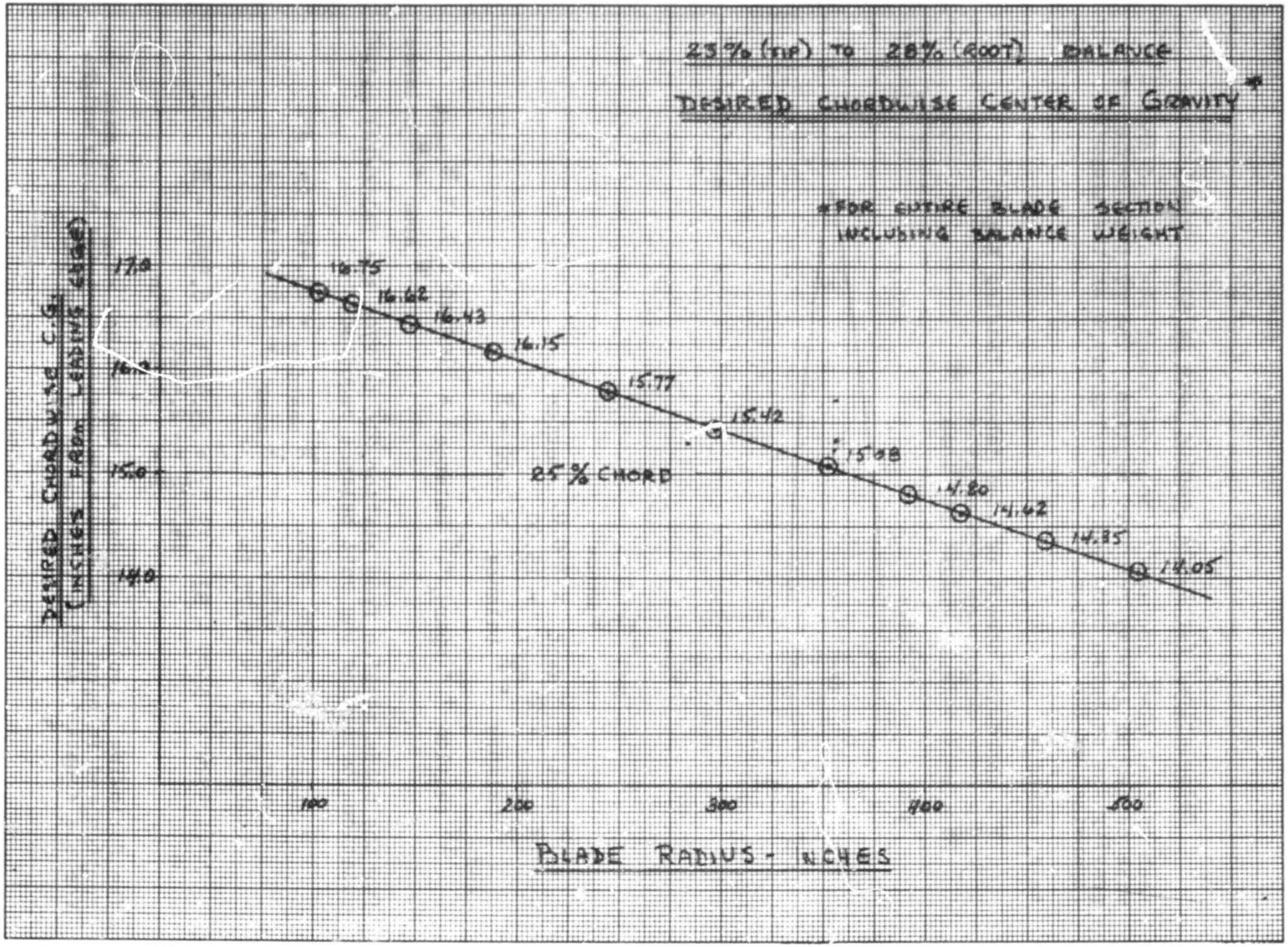
$$(\bar{W}_{section} + \bar{W}_{spar} + \bar{B}_{balance}) \times \text{Plotted C.G.} = \bar{M}_{sect} + \bar{M}_{spar} + \bar{M}_{balance}$$

For the blade section from station 216 to 270, the following calculations determine the unit balance required.

$\bar{W}_{section}$ = 1.3026 #/in.    $\bar{M}_{section}$ = 23.4183 in. #
$\bar{W}_{spar}$ = .5900 #/in.    $\bar{M}_{spar}$ = 8.8500 in. #
Plotted C.G. = 15.77 in.

$$\therefore (1.3026 + .5900 + \bar{B}) \times 15.77 = 23.4183 + 8.8500 + \bar{B}$$
$$14.77\bar{B} = 2.4220$$
$$\bar{B} = .1640 \text{ \#/in.}$$

The following calculations develop the balance weight for each section and integrate the results to obtain the total blade weight for the constant section (station 96 to station 540).


23% (TIP) TO 28% (ROOT) BALANCE
DESIRED CHORDWISE CENTER OF GRAVITY*
*FOR ENTIRE BLADE SECTION INCLUDING BALANCE WEIGHT
DESIRED CHORDWISE C.G. (INCHES FROM LEADING EDGE)
17.0
16.0
15.0
14.0
16.75
16.62
16.43
16.15
15.77
15.42
15.08
14.80
14.62
14.35
14.05
25% CHORD
100
200
300
400
500
BLADE RADIUS - INCHES

BALANCE CALCULATIONS FOR
23% TO 28% CHORD BALANCE

## 23% TO 28% BALANCE

| STA | $\bar{\omega}$ | X(LE) | $\bar{\omega}$X | BALANCE CALC |
|---|---|---|---|---|
| 96-108 | 1.3026 | 17.98 | 23.4183 | (2.0266+B)×16.75=34.2783+B |
| SPAR | .724 | 15.00 | 10.8600 | ∴ B = .0211 #/IN |
| BALANCE | .0211 | 1.00 | .0211 | |
| | 2.0477 | 16.75 | 34.2994 | |
| 108-135 | 1.3026 | 17.98 | 23.4183 | (2.0266+B)×16.62=34.2783+B |
| SPAR | .724 | 15.00 | 10.8600 | B= .0382#/IN |
| BALANCE | .0382 | 1.00 | .0382 | |
| | 2.0648 | 16.62 | 34.3165 | |
| 135-162 | 1.3026 | 17.98 | 23.4183 | (2.0076+B)×16.43=33.9933+B |
| SPAR | .705 | 15.00 | 10.5750 | B = .0654#/in |
| BALANCE | .0654 | 1.00 | .0654 | |
| | 2.0730 | 16.43 | 34.0587 | |
| 162-216 | 1.3026 | 19.98 | 23.4183 | (1.9626+B)×16.15=33.8183+B |
| SPAR | .66 | 15.00 | 9.9000 | B=.1071 #/in |
| BALANCE | .1071 | 1.00 | .1071 | |
| | 2.0697 | 16.15 | 33.9254 | |
| 216-270 | 1.3026 | 17.98 | 23.4183 | (1.8926+B)×15.77=32.2683+B |
| SPAR | .59 | 15.00 | 8.8500 | B = .1640#/in |
| BALANCE | .1640 | 1.00 | .1640 | |
| | 2.0566 | 15.77 | 32.4323 | |
| 270-324 | 1.3026 | 17.98 | 23.4183 | (1.8126+B)×15.42=31.0683+B |
| SPAR | .51 | 15.00 | 7.6500 | B= .2162 #/in |
| BALANCE | .2162 | 1.00 | .2162 | |
| | 2.0288 | 15.42 | 31.2845 | |
| 324-378 | 1.2685 | 18.06 | 22.9036 | (1.6985+B)×15.08=29.3536+B |
| SPAR | .43 | 15.00 | 6.4500 | B= .2656#/in |
| BALANCE | .2656 | 1.00 | .2656 | |
| | 1.9641 | 15.08 | 29.6192 | |
| 378-405 | 1.2685 | 18.06 | 22.9036 | (1.6385+B)×14.80=28.4536+B |
| SPAR | .37 | 15.00 | 5.5500 | B= .3046#/in |
| BALANCE | .3046 | 1.00 | .3046 | |
| | 1.9431 | 14.80 | 28.7582 | |

## 23% TO 27% BALANCE (CONT'D)

| STA | $\bar{w}$ | x(LE) | $\bar{w}$x | BALANCE CALC |
|---|---|---|---|---|
| 405-432 | 1.1465 | 18.36 | 21.0461 | (1.4665+B)×14.62=25.8461+B |
| SPAR | .32 | 15.00 | 4.8007 | B = .3235 #/in |
| BALANCE | .3235 | 1.00 | .3235 | |
| | 1.7900 | 14.62 | 26.1696 | |
| 432-486 | 1.1312 | 18.36 | 20.7634 | (1.3812+B)X1435=24.5134+B |
| SPAR | .25 | 15.00 | 3.7500 | B = .3516 #/in |
| BALANCE | .3516 | 1.00 | .3516 | |
| | 1.7328 | 14.35 | 24.8660 | |
| 486-522 | 1.1212 | 18.40 | 20.6264 | (1.2812+B)X14.05=23.0264+B |
| SPAR | .16 | 15.00 | 2.4000 | B = .3851 #/in |
| BALANCE | .3851 | 1.00 | .3851 | |
| | 1.6663 | 14.05 | 23.4115 | |
| 522-540 | .4866 | 7.50 | 3.6495 | |
| CASCADE | .6111 | 15.00 | 9.1665 | |
| BALANCE | ~ | 1.0 | | |
| | 1,0977 | 11.68 | 12.8160 | |

(13.85) for 23%

19.5%

INTEGRATION TO OBTAIN TOTAL
CONSTANT BLADE WEIGHT

## INTEGRATE BLADE
## 23% TO 28% BALANCE

| STA | $\bar{w}$ | L | W | X | R | WX | WR |
|---|---|---|---|---|---|---|---|
| 96-108 | 2.0477 | 12 | 24.57 | 16.75 | 102.0 | 411.55 | 2506.1 |
| 108-135 | 2.0648 | 27 | 55.75 | 16.62 | 121.5 | 926.56 | 6773.6 |
| 135-162 | 2.0730 | 27 | 55.97 | 16.43 | 148.5 | 919.59 | 8311.5 |
| 162-216 | 2.0697 | 54 | 111.76 | 16.15 | 189.0 | 1804.92 | 21122.6 |
| 216-270 | 2.0566 | 54 | 111.06 | 15.77 | 243.0 | 1751.42 | 26987.6 |
| 270-324 | 2.0288 | 54 | 109.56 | 15.42 | 297.0 | 1689.42 | 32539.3 |
| 324-378 | 1.9641 | 54 | 106.06 | 15.08 | 351.0 | 1599.38 | 37227.1 |
| 378-405 | 1.9431 | 27 | 52.46 | 14.80 | 391.5 | 776.41 | 20538.1 |
| 405-432 | 1.7900 | 27 | 48.33 | 14.62 | 418.5 | 706.58 | 20226.1 |
| 432-486 | 1.7328 | 54 | 93.57 | 14.35 | 459.0 | 1342.73 | 42948.6 |
| 486-522 | 1.6663 | 36 | 59.99 | 14.05 | 504.0 | 842.86 | 30235.0 |
| 522-540 | 1.0977 | 18 | 19.76 | 11.68 | 531.0 | 230.80 | 10492.6 |
| TOTAL BLADE<br>(OUTBD STA 96) | | | 848.8 # | 15.32" | 306.2" | 13002.22 | 259908.2 |

CF @ 750 = $2.841 \times 259908 \times 159^2 \times 10^{-5}$ = 186600 # HOVER

CF @ 675 = $143^2$ = 151000 # CRUISE

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BALANCE CALCULATIONS FOR
25% CONSTANT BALANCE

## 25% BALANCE

| STA | $\bar{w}$ | X(LE) | $\bar{w}$X | BALANCE CALC. |
|---|---|---|---|---|
| 96-135 | 1.3026 | 17.98 | 23.4183 | (2.0266+B)×15.0=34.2983+B |
| SPAR | .724 | 15.00 | 10.8600 | B=.2771#/IN |
| BALANCE | .2771 | 1.00 | .2771 | |
| | 2.3037 | 15.00 | 34.5554 | |
| 135-162 | 1.3026 | 17.98 | 23.4183 | (2.0076+B)15.0=33.9933+B |
| SPAR | .705 | 15.00 | 10.5750 | B=.2771#/IN |
| BALANCE | .2771 | 1.00 | .2771 | |
| | 2.2847 | 15.00 | 34.2704 | |
| 162-216 | 1.3026 | 17.98 | 23.4183 | (1.9626+B)×15.0=33.3183+B |
| SPAR | .660 | 15.00 | 9.9000 | B=.2771#/IN |
| BALANCE | .2771 | 1.00 | .2771 | |
| | 2.2397 | 15.00 | 33.5954 | |
| 216-270 | 1.3026 | 17.98 | 23.4183 | (1.8926+B)×15.0=32.2683+B |
| SPAR | .5900 | 15.00 | 8.8500 | B=.2771#/in |
| BALANCE | .2771 | 1.00 | .2771 | |
| | 2.1697 | 15.00 | 32.5454 | |
| 270-324 | 1.3026 | 17.98 | 23.4183 | (1.8126+B)×15.0=31.0683+B |
| SPAR | .510 | 15.00 | 7.6500 | B=.2771#/IN |
| BALANCE | .2771 | 1.00 | .2771 | |
| | 2.0897 | 15.00 | 31.3454 | |
| 324-378 | 1.2685 | 18.06 | 22.9036 | (1.6985+B)×15.0=29.3536+B |
| SPAR | .430 | 15.00 | 6.4500 | B=.2769#/in |
| BALANCE | .2769 | 1.00 | .2769 | |
| | 1.9754 | 15.00 | 29.6305 | |
| 378-405 | 1.2685 | 18.06 | 22.9036 | (1.6385+B)×15.0=28.4536+B |
| SPAR | .370 | 15.00 | 5.5500 | B=.2769#/in |
| BALANCE | .2769 | 1.00 | .2769 | |
| | 1.9154 | 15.00 | 28.7305 | |

3700

25% BALANCE (CONTINUED)

| STA | $\overline{W}$ | X(LE) | $\overline{W}X$ | BALANCE CALC |
|---|---|---|---|---|
| 405-432 | 1.1465 | 18.36 | 21.0461 | (1.4665+B)x15.00 = 25,8461+B |
| SPAR | .320 | 15.00 | 4.8000 | B = .2749 #/in |
| BALANCE | .2749 | 1.00 | .2749 | |
| | 1.7414 | 15.00 | 26.1210 | |
| 432-486 | 1.1312 | 18.36 | 20.7634 | (1.3812+B)x15.0 = 24.5134+B |
| SPAR | .2500 | 15.00 | 3.7500 | B = .2711 #/in |
| BALANCE | .2711 | 1.00 | .2711 | |
| | 1.6523 | 15.00 | 24.7845 | |
| 486-522 | 1.1212 | 18.40 | 20.6264 | (1.2812+B)x15.0 = 23.0264+B |
| SPAR | .160 | 15.00 | 2.4000 | B = .2720 #/in |
| BALANCE | .2720 | 1.00 | .2720 | |
| | 1.5532 | 15.00 | 23.2984 | |
| 522-540 | .4866 | 7.50 | 3.6495 | |
| CASCADE | .6111 | 15.00 | 9.1665 | |
| BALANCE | ~ | 1.00 | ~ | |
| | 1.0977 | 11.68<br>(19.5%)<br>(15.00% FOR 25%) | 12.8160 | |

INTEGRATION TO OBTAIN TOTAL BLADE WEIGHT

INTEGRATE BLADE (25% BALANCE)

| STA | $\bar{\omega}$ | L | W | X(L.E.) | R | WX | WR |
|---|---|---|---|---|---|---|---|
| 96-135 | 2.3037 | 39 | 89.84 | 15.0 | 110.5 | 1342.6 | 9927.3 |
| 135-162 | 2.2847 | 27 | 61.69 | 15.0 | 148.5 | 925.4 | 9161.0 |
| 162-216 | 2.2397 | 54 | 120.94 | 15.0 | 189.0 | 1814.1 | 22859.7 |
| 216-270 | 2.1697 | 54 | 117.16 | 15.0 | 243.0 | 1757.4 | 28469.9 |
| 270-324 | 2.0897 | 54 | 112.84 | 15.0 | 297.0 | 1692.6 | 33513.6 |
| 324-378 | 1.9754 | 54 | 106.67 | 15.0 | 351.0 | 1600.0 | 37441. |
| 378-405 | 1.9154 | 27 | 51.72 | 16.0 | 391.5 | 775.8 | 20248.4 |
| 405-432 | 1.7414 | 27 | 47.02 | 16.0 | 418.5 | 706.3 | 19677.9 |
| 432-486 | 1.6523 | 54 | 89.22 | 15.0 | 459.0 | 1338.3 | 40952.0 |
| 486-522 | 1.5532 | 36 | 55.92 | 15.0 | 504.0 | 838.8 | 28183.7 |
| 522-540 | 1.0977 | 18 | 19.76 | 11.68 | 531.0 | 230.80 | 10492.6 |
| TOTAL BLADE (OUTBD STA 96) | | | 872.8# | 14.92" | 299.0" | 13026.1 | 260925.2 |

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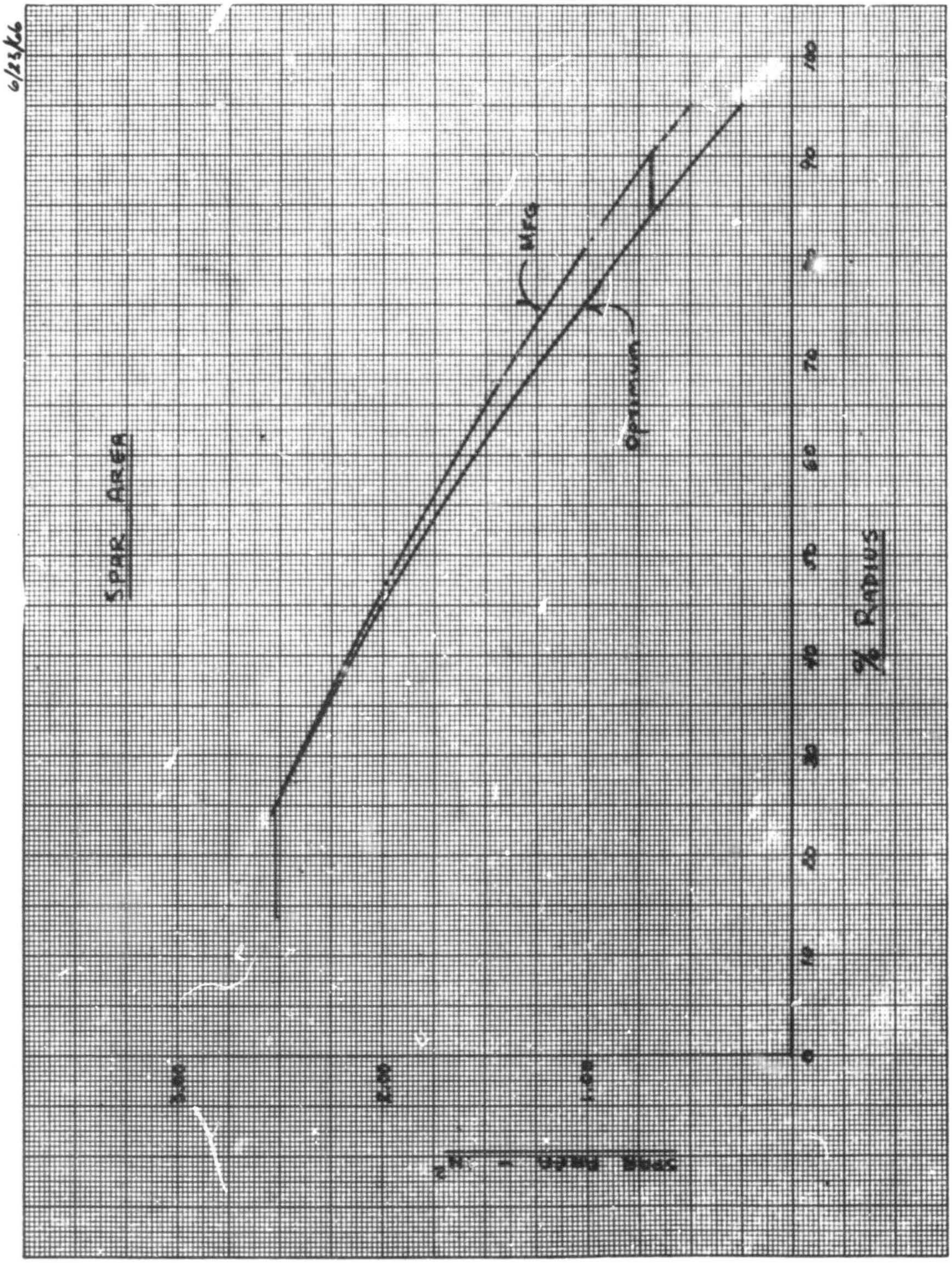
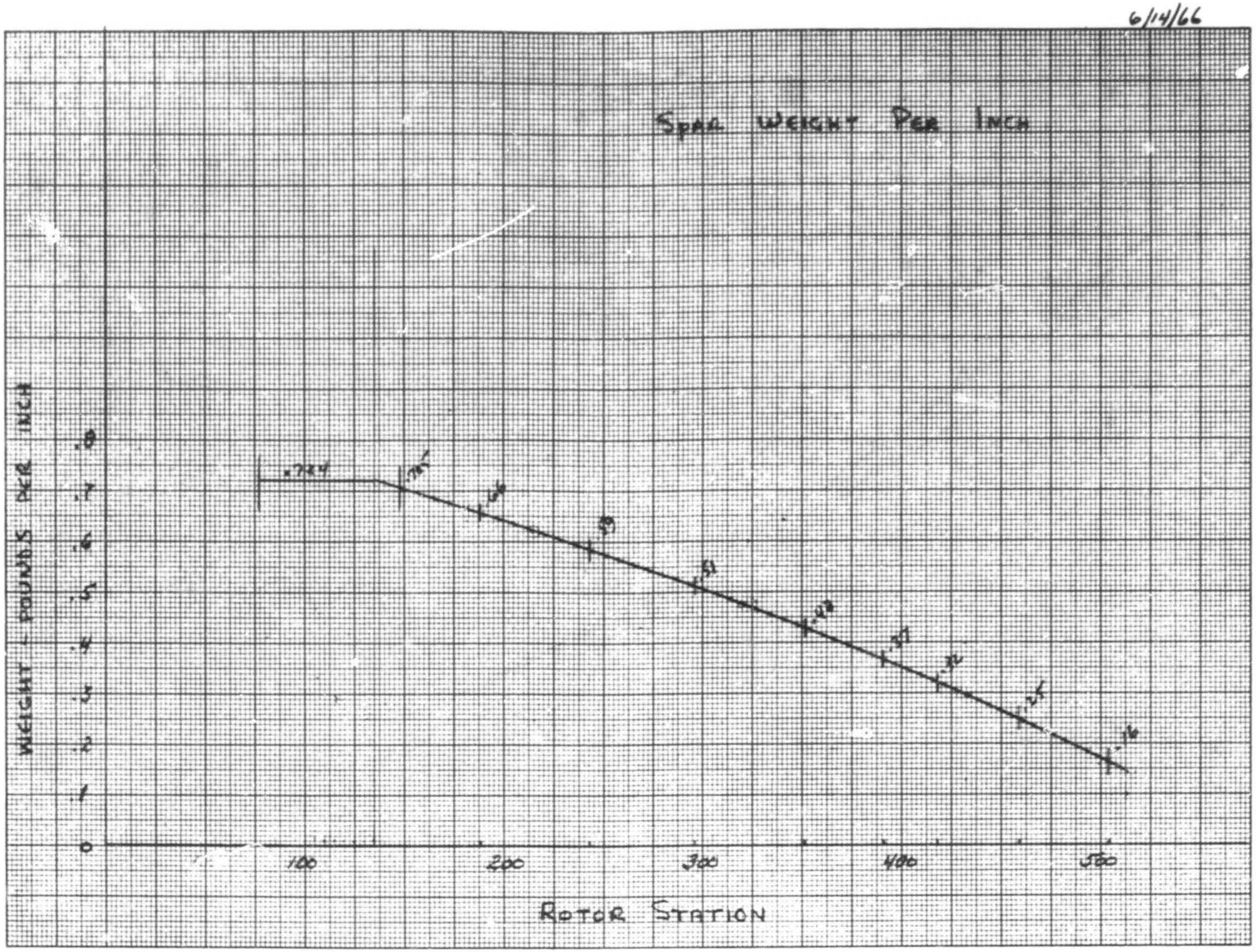

6/23/66
SPAR AREA
MFG
OPTIMUM
% RADIUS
0
10
20
30
40
50
60
70
80
90
100
1.00
2.00
3.00
SPAR AREA – IN²

SPAR WEIGHT PER INCH
WEIGHT - POUNDS PER INCH
0
.1
.2
.3
.4
.5
.6
.7
.8
100
200
300
400
500
ROTOR STATION
.724
.705
.66
.59
.51
.43
.37
.32
.25
.16

6/14/66

WEIGHT CALCULATION FOR BLADE
WEIGHT (OTHER THAN CONSTANT SECT.)

## LEAD-LAG FLEXURE

$177.6\ \text{IN}^2 \times .42\ell \times .286 =$ 21.3# EACH

## L.E. BALANCE (EXTRA AREA FOR DAMPER LOADS)

$.75\ \text{in}^2$ TO 0 for 50" - TIE IN AT R = 100

BALANCE @ STA 100 : $.074\ \text{in}^2$

$\therefore$ NEED $.750\ \text{in}^2 - .074\ \text{in}^2 = .676\ \text{in}^2 \longrightarrow 0\ \text{in}^2$ @ 150

$\frac{.676}{2} \times 50 \times .286 =$ 4.8# PER BLADE

## TIP CASCADE (.025 INCONEL) (REF. DRAWING 395-0919)

| | | | |
|---|---|---|---|
| XU-9A | | | |
| STRUCTURE | 5.03# | $\therefore 5.03 \times \frac{\text{Chd}}{31.5} = 5.03\,\frac{60}{31.5} =$ | 9.58# |
| VANES | .72# | | |
| (31.5" CHORD) | .10 in | $.15 \times \frac{8.4}{5} \times 7 = 2.12$ | 2.12 |
| | | | 11.7# say 12# |

## FLAPPING STRAP (.50ℓ CRES)

| | |
|---|---|
| CONSTANT SECTION | $99.9\ \text{in}^2$ |
| INBOARD END | $75.2\ \text{in}^2$ |
| OUTBOARD END | $107.0\ \text{in}^2$ |

$282.1 \times .5\ell \times .286 =$ 40.5#

## DAMPER (LEAD-LAG)

369A Damper : 2.1# WITH 1" RADIUS
Per Stress HCH would require 5" Radius and 1¼ the depth of the 369A damper.

$\therefore 2.1^{\#} \times \frac{5^2}{1^2} \times 1.25 = 66^{\#}$ 66#

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BLADE COMPONENTS WEIGHT
CALCULATIONS (CONT'D)

**ARTICULATED DUCT**

TRI-DUCT TO SLIP JOINT (9.5" DIA - 2 REQ.)
XV-9A = 10.85# @ 8.5" DIA
∴ $10.85 \times \frac{9.5}{8.5} = 12^{\#}$
$+ 18"L \times 9.5 \times 3.1416 \times .12 \times .286 = 3^{\#}$ } $15^{\#} \times 2 =$ $30^{\#}$

SLIP JOINT (2 REQ.)
XV-9A = 7.88#
∴ $7.88 \times \frac{9.5}{8.5} = 8.8^{\#} \times 2 =$ $18^{\#}$

SLIP JOINT TO STA 96 (2 REQ.) L = 28"

$28 \times 9.5 \times 3.1416 \times .020 \times .286 = 4.8^{\#} \times 2$ $10^{\#}$

BELLOWS (2 REQ.) (12" DIA)

XV-9A = 3.9# @ 9" DIA

∴ $3.9 \times \frac{12}{9} = 5^{\#} \times 2$ $10^{\#}$

TOTAL ARTICULATED DUCT $68^{\#}$

**TORQUE BOX FAIRING**

ASSUME .062 BOLTIRON (SAME AS XV-9A)

$130 \times 33 = 4300\ \text{in}^2 \times .062 \times .049 = 12^{\#}$
$+$ 30% LOCAL STIFFENING $\frac{4}{16^{\#}}$ $\underline{16^{\#}}$

**STUB SPAR** STA 78 TO 96

$.724^{\#}/\text{IN} \times 18" = 13.03 + 1.5 \times$ BEEF-UP
$13.03 + 15.54 =$ $\underline{28.6^{\#}}$

BLADE COMPONENTS WEIGHT
CALCULATIONS (CONT'D)

**TRANSITION SECTION** (STA 67 TO 96) (REF. DRAWING 395-0924)

SKIN (USE NACA 0018 SKIN)
.4027 #/IN x 30" = 12.1#

CLOSURE CHANNEL
7 x 30 x .015 x .286 = .9

OUTBOARD BULKHEAD
CAP 65" x .25 16.25
WEB 74.2 $in^2$ x .06 4.45
20.70 $in^3$ x .284 = 5.9#

MID STABILIZING RIBS (2)
SAME AS OUTBD RIB = 5.9 x 2 = 11.9#

INBOARD BULKHEAD
316.8 $in^2$ x .06 19.01
11 x .6 x 2 13.20
34 x 4 x .125 x 1 12.00
2 x 2.5 x 1 x 4 20.00
69.21 $in^3$ x .284 = 19.7#

TRAILING EDGE (SEE NACA 0018)
.1611 #/IN x 30" = 4.8#

TORSION FLEXURE (2) EST 2#

TOTAL TRANSITION SECTION 59.2#

**HUB FAIRING** (USE XV-9A TYPE COVER) 45"DIA 35"HIGH

XV-9A = 8.33# @ 3050 $in^2$

$8.33 \times \frac{7660\ in^2}{3050\ in^2} =$ 20.9#

STIFFENING + HARDWARE 3.# 24#

**DROOP STOP** (T1.)

2 $in^2$ x 30 60
3 $in^2$ x 12 36
1 $in^2$ x 6 x 3 18
114 $in^3$ x .16 = 18.2#

**DAMPER ARM ( P/O TORQUE BOX)**

$32.4''L \times .4 in^2$ 12.96

$8.8 \times 6.4 \times .05$ $\underline{2.82}$

$15.78 \times .286 =$ $\underline{4.5^{\#}}$

BLADE COMPONENTS WEIGHT
CALCULATIONS (CONT'D)

**TORQUE BOX TO HUB TRUSS** (STEEL TUBING)

(1"DIA × .083WALL) .0679 #/IN × 20" · 1.4 + 1.0 END FITTINGS = 2.4
(1"DIA × .083WALL) .0679 #/IN × 12" = .8 + 1.0 = 1.8
2 (2"DIA × .095WALL) .1615 × 16" = 2.6 + 1.0 (2 REQ.) = 7.2
4 (2"DIA × .083WALL) .1420 × 15" 2.1 + 1.0 (4 REQ.) = 12.4
MISC. HARDWARE, WELD, ETC. 2.4

26.2#

**TORQUE BOX** (TITANIUM)

SKIN 17 × 3.1416 × 30.5 × .08 × .17 = 22.2#

INBOARD FRAME $17.26 \times 3.1416 \times 2.5 \times .1$
$(18.8^2 - 17.0^2) \times .7854 \times .25$
$18.8^2 \times .7854 \times .25$
$6.4 \times 4 \times 1.4$
$-7.4 \times 8 \times .25$
} $116.65 in^3 \times .17 = 19.8$#

OUTBOARD FRAME $17 \times 3.1416 \times 7.6 \times .08$
$-5.6 \times 44 \times .08$
$17^2 \times .7854 \times .08$
$17^2 \times .7854 \times .10$
$-22.8 \times 2 \times .10$
$-22.8 \times 2 \times .08$
$8 \times .4 \times 7 \times .4$
} $135.01 in^3 \times .17 = 23.0$#

MINOR FRAMES (3)
$17 \times 3.1416 \times 1 \times .08$
$17^2 \times .7854 \times .08$
} $22.43 \times .17 \times 3$ = 11.4#

STRAP SHOES $6.2 \times 3.6 \times 6 \times 4 \times .17$ = 9.1#

TRUSS TIE-IN FITTINGS (1.5# EA. × 4) = 6.0#

HARDWARE 1.0#

TOTAL TORQUE BOX 92.5#

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## HUB AND SHAFT CALCULATIONS

**HUB SUPPORT COLUMN (FIXED SHAFT)** (REF. DWG. 395-0932)

| SECTION | | |
|---|---|---|
| 1 | $1.1 \times .15 \times 3.1416 \times 37.8$ | 19.58 |
| 2 | $.35 \times .55 \times 3.1416 \times 33.0$ | 22.99 |
| 3 | $2.85 \times .4 \times 3.1416 \times 37.8$ | 135.36 |
| 4 | $1.5 \times .5 \times 3.1416 \times 37.642$ | 168.32 |
| 5 | $.4\ in^2 \times 3.1416 \times 35.7 \times 2$ | 89.72 |
| 6 | $24.0 \times .1 \times 3.1416 \times 38.1$ | 287.29 |
| 7 | $.121\ in^2 \times 3.1416 \times 37.6 \times 2$ | 28.68 |
| 8 | $3.35 \times .5 \times 3.1416 \times 37.8$ | 198.90 |
| 9 | $7.8 \times .1 \times 3.1416 \times 38.0$ | 93.10 |
| 10 | $(44.0^2 - 38.2^2) .7854 \times .1$ | 37.45 |
| 11 | $1.0 \times .5 \times 3.1416 \times 44.5$ | 69.90 |
| 12 | $(36.0D - 35.0D) \times .1 \times 5$ | 27.88 |

$1169\ in^3 \times .286 = \underline{334.3}$ #

**UPPER BEARING** MODEL 485 BEARING = 198# @ 48" DIA (SEE NEXT PAGE FOR CALC)

BEARING $198 \times \frac{35\ DIA}{48\ DIA} = \underline{144}$ # (2 REQ) $\underline{288}$ #

**LOWER BEARING** (SEE NEXT PAGE) $\underline{74}$ #

**UPPER RETAINER**

| | |
|---|---|
| $(39.8^2 - 33.6^2) \times .7854 \times .1$ | 35.74 |
| $.5 \times .6 \times 3.1416 \times 35.2$ | 33.18 |
| $(35.6^2 - 34.0^2) \times .7854 \times .05$ | 4.37 |

$73.3\ in^3 \times .286 = \underline{21}$ #

**UPPER FLANGE**

| | |
|---|---|
| $(39.8^2 - 37.8^2) \times .7854 \times .1$ | 12.19 |
| $.1 \times .8 \times 3.1416 \times 38.0$ | 9.55 |

$21.7\ in^3 \times .286 = \underline{6.2}$ #

**LOWER SEAL** (AL. AL.)

| | |
|---|---|
| $.28 \times 4 \times 3.1416 \times 34.4$ | 108.07 |
| $8.8 \times .2 \times 3.1416 \times 37$ | 204.58 |

$312.6\ in^3 \times .101 = \underline{31.6}$ #

MODEL 485 MAIN BEARING CALCULATIONS (TO RATIO TO MODEL 395)

UPPER BEARING STEEL
DIAMETER OF BEARINGS 1.625 in., 92 REQUIRED
TOTAL-BEARING WEIGHT

$$\frac{3.14 \times (1.625)^3}{6} \text{IN}^3 \times 92 \times .285^{\#}/\text{IN}^3 = 58.6^{\#}$$

INNER AND OUTER RACES

$$48 \text{ IN} \times 3.14 \times 2.6 \text{ IN} \times 1.25 \text{ IN} \times .285^{\#}/\text{IN}^3 = 139.6^{\#}$$

TOTAL 48" DIA BEARING $\underline{198^{\#}}$

RE-CALC 8/23/66

LOWER BEARING (2 REQ.)
BALLS 1.375 DIA, 67 REQ

$$\frac{3.14 \times 1.375^3}{6} \times 67 \times .285 = 26.0$$

RACES $35 \times 3.14 \times 2.6 \times 1.25 \times .285 = 102.0$

BEARING RETAINER $\underline{16.0}$

$144^{\#} \times 2 = \underline{288^{\#}}$

UPPER BEARING
BALLS .875 DIA × 94 REQ.

$$\frac{3.14 \times .875^3}{6} \times 94 \times .285 = 9.4$$

RACES $35 \times 3.14 \times 1.8 \times 1.0 \times .285 = 56.4$

BEARING RETAINER $\underline{8.0}$

$74^{\#}$ $\underline{74^{\#}}$

## HUB & SHAFT (CONTINUED)

**ROTATING SHAFT** (TO POINT "A" ONLY) (CRES)

| SECTION | | |
|---|---|---|
| 1 | 1.75 × .65 × 3.1416 × 32.8 | 117.21 |
| 2 | $(32.2^2 - 27.2^2) \times .7854 \times .1$ | 23.33 |
| 3 | .3 × 1.5 × 3.1416 × 27.4 | 38.74 |
| 4 | .5 × .25 × 3.1416 × 27.4 | 10.76 |
| 5 | 2.6 × .4 × 3.1416 × 32.5 | 106.18 |
| 6 | 24.5 × .11 × 3.1416 × 32.2 | 272.62 |
| 7 | 2.7 × .4 × 3.1416 × 32.4 | 109.93 |
| 8 | $(32.0^2 - 27.3^2) \times .7854 \times .1$ | 21.89 |
| 9 | 2.25 × .35 × 3.1416 × 26.8 | 66.30 |
| 10 | 6 × .25 × 3.1416 × 32.4 | 152.68 |
| 11 | 2.6 × .05 × 3.1416 × 32.5 | 13.27 |
| 12 | .55 × .25 × 3.1416 × 32.2 | 13.91 |
| 13 | (32D − 31D) × .11 × 6 | 32.66 |

$979.4 \text{ in}^3 \times .286 = \underline{280}$#

**LOWER BEARING RETAINER** (CRES)

| | |
|---|---|
| .42 × 2.4 × 3.1416 × 33.0 | 104.50 |
| .4 × .6 × 3.1416 × 33.4 | 25.18 |

$129.7 \text{ in}^3 \times .286 = \underline{37}$#

**ACCESSORY DRIVE GEAR** (NOT PART OF ROTOR GROUP WEIGHT) (CRES)

| | |
|---|---|
| 2.6 × .25 × 3.1416 × 32.8 | 66.98 |
| .9 × .4 × 3.1416 × 34.0 | 38.42 |
| 2.25 × .6 × 3.1416 × 35.4 | 150.10 |

$255.5 \text{ in}^3 \times .286 = \underline{73}$#

**DROOP STOP SUPPORT** (STEEL - 3 REQ.)

1" wall × .083 .0679 × 22 × 2 = 2.98 + 2.0 Fitting = 5.0
1" wall × .083 .0679 × 30 = 1.36 + 1.0 fitting = 2.4
DROOP STOP PAD 6 × 2 × .5 × .286 = 1.7

9.1
× 3

27.3#

**FEATHERING BEARINGS IN HUB**

$(4.4^2-3.0^2) \times .7854 \cdot 2.5 \times .286 \times 3$ 17#

**HUB**

| | | |
|---|---|---|
| UPPER & LOWER PLATES | 2908.8 in² × .1 = | 290.9 |
| STRAP TIE-IN WEB | 186.8 Long × 2.38 in² = | 444.6 |
| CROSS-TIE | 38.4 Long × 1.0 in² = | 38.4 |
| CROSS-TIE | 45.6 Long × 2.0 in² = | 91.2 |
| CROSS-TIE | 60.0 Long × 1.0 in² = | 60.0 |
| STRAP WEB TIE | 456.0 Long × .8 in² = | 364.8 |

1289.9 in³ × .286 = 369#

**HUB SUPPORT** (3 REQ.)

| | | |
|---|---|---|
| UPPER PAD | 3.76 × 1.2 × .5 × 3 | 6.768 |
| LOWER PAD | 1.50 × 4.2 × .5 × 2 | 6.300 |
| OUTER STRUT | .7 in² × 8.75 × 2 | 12.250 |
| INNER STRUT | .7 in² × 9.75 × 2 | 13.650 |

38.968 in³ × .284 = 11.07#

× 3

33.2

+ HARDWARE 4.0

37.2# 37.2

PROPULSION "Y" & TRI-DUCT WEIGHT
CALCULATIONS

## PROPULSION- HOT GAS DUCTING - REF. STUDY DRAWING 395-0925

### TRI-DUCT

OUTER DUCT 24" - INNER DUCT 17" (STRESSED SKIN)
SKIN .30#/in³ MATERIAL .015" THICK

| | | | |
|---|---|---|---|
| DUCT | 17×3.1416×50×.015 | 40.06 | |
| | 17×3.1416×42×.015 | 33.65 | |
| | 17×3.1416×42×.006×2 | 26.92 | |
| | 9×3.1416×30×.015×3 | 38.17 | |
| | 9×3.1416×30×.015×3 | 38.17 | |
| | 24×3.1416×50×.015 | 56.54 | |
| | | 233.5 in³ × .3 | 70.0# |
| DUCT TO BLADE FLANGE -14"DIA. | XV-9A=2.44# @10"D ∴ 2.44×$\frac{14}{10}$×6 | | 20.4# |
| STIFFENING (SAME AS XV-9A)×2 | | | 10.6# |
| LOWER FLANGE | .25×1.2×17×3.1416 | 16.02 | |
| | .1×1.2×25×3.1416 | 9.42 | |
| | .15×.6×25.5×3.1416 | 7.21 | |
| | .1×.5×3.5×3 | .53 | |
| | | 33.2 in³ ×.3 | 10.0# |
| INSULATION | 26×3.1416×50 | 4084 | |
| | 9×3.1416×30×6 | 5272 | |
| | | 9356 × .0012#/in³ | 11.2# |
| | TOTAL TRI-DUCT | | 122.2# |

### Y TO TRI-DUCT SEAL

| | | |
|---|---|---|
| TRI-DUCT SEAL HSNG | 53.41in³ × .286 = | 15.3# |
| Y DUCT SEAL HSNG | 52.91in³ × .286 = | 15.1 |
| SPRINGS & WASHERS | = | 1.0 |
| CARBON SEALS-INNER | 3.3in³ × .33 × 2 = | 2.2 |
| -OUTER | 4.9in³ × .33 = | 1.6 |
| TOTAL SEAL | | 35.2# |

## HOT GAS DUCTING – (CONTINUED)

### Y DUCT (FOR CONFIGURATION #2)

| | | |
|---|---|---|
| 17×3.1416×85×.015 | 68.09 | |
| 17×3.1416×45×.015 | 36.05 | |
| 24×3.1416×40×.015 | 45.24 | |
| 17×3.1416×40×.015 | 32.04 | |
| 17×3.1416×2×40×.006 | 25.64 | |
| | 2021 in$^3$×.3 | 62.1# |
| DIVERTER VALVE ADAPTER<br>SAME AS XV-9A ×2 | | 5.9# |
| BELLOWS(2) @ 18" DIA<br>XV-9A · 3.16# @ 12"<br>$3.16 \times \frac{18}{12} \times 2$ | | 9.5# |
| FLANGE TO TRI-DUCT<br>SAME AS FLANGE ON TRI-DUCT | | 10.0# |
| INSULATION<br>18×3.1416×45×2<br>25×3.1416×40 | <br>5089.4<br>3141.6<br>8231×.0012#/in$^3$ | 9.9# |
| TOTAL Y DUCT | | 99.4# |

TOTAL HOT GAS DUCTING
(FROM DIVERTER VALVE TO
BLADE ARTICULATED DUCT) 255#

SUMMARY - ROTATING CONTROLS
WEIGHT CALCULATIONS

## SUMMARY
## FLIGHT CONTROLS
## (ROTATING HEAD CONTROLS)

REF. DRAWING 395-0932

| | | |
|---|---|---|
| STATIONARY SWASHPLATE | | 319.0 |
| SWASH PLATE | 249.0 | |
| ARMS | 70.0 | |
| ROTATING SWASHPLATE | | 224.7 |
| SWASHPLATE | 184.0 | |
| ARMS | 40.7 | |
| SWASHPLATE BEARINGS | | 200.0 |
| SWASHPLATE BEARING BALL | | 75.2 |
| TORQUE SCISSORS | | 11.5 |
| VERTICAL LINKS + FITTINGS | | 29.3 |
| PITCH ARM (IN TORQUE BOX WT.) | | — |
| HYDRAULIC CYLINDERS & MOUNTS | | 390.0 |
| TOTAL ROTATING CONTROLS | | 1250# |

## SWASHPLATE ACCESSORY WEIGHT CALCULATIONS

**SWASHPLATE SUPPORT** AL

$19IN \times 1.5(STIFF) \times 0.2\,IN \times 42IN \times \pi \times 0.1LB/IN^3 =$ 75.2#

**TORQUE SCISSORS** AL

$8.0IN \times 0.2\,IN \times 24IN \times 0.1LB/IN^3 \times 3REQ =$ 11.5#

**PUSH-PULL LINKS** ST

35"LONG $A = 0.82\,IN^2$

$35IN \times 0.82\,IN^2 \times 0.285LB/IN^3 \times 3REQ =$ 24.5#

**END FITTINGS (PUSH-PULL LINKS)**

0.8LB EA × 6 REQ 4.8#

**SWASHPLATE ARMS - STATIONARY** ST

$45\,IN \times (2 \times 0.75IN \times 0.25IN + 8.0IN \times 0.15IN)$
$\times 0.285\,LB/IN^3 \times 3REQ =$ 70.0#

**SWASHPLATE ARMS - ROTATING** ST

$24IN \times (2 \times 1.0IN \times 0.1875IN + 8.0IN \times 0.2IN)$
$\times 0.285LB/IN^3 \times 3REQ =$ 40.7#

8704

## SWASHPLATES — WEIGHT CALCULATIONS

**ROTATING SWASHPLATE** STEEL

$51\,\text{IN} \times \pi \times (2.5 \times 0.7 + 3.6 \times .15 + 2.5 \times 0.7)\,\text{IN}^2 \times 0.285\,\text{LB/IN}^3 = \underline{184^{\#}}$

**STATIONARY SWASHPLATE** ST

$$46\,\text{IN} \times \pi \times (1.25 \times 1.0 + 5.2 \times 0.1 + 1.0 \times 1.25 + 1.4 \times 0.5 + 2.3 \times 0.25 + 3.5 \times 0.5)\,\text{IN}^2 \times 0.285\,\text{LB/IN}^3 = \underline{249^{\#}}$$

**SWASHPLATE BEARINGS**

BALLS ST

$$\left(\frac{.75\,\text{IN}}{2}\right)^3 \times \frac{4}{3}\pi \times 0.285\,{}^{\#}/\text{IN}^3 \times 300\,\text{REQ} = 18.8^{\#}$$

BEARING RETAINERS ST

$0.3\,\text{IN} \times 6.75\,\text{IN} \times 47.6\,\text{IN} \times \pi \times 0.285\,\text{LB/IN}^3 = 86.3^{\#}$

INNER AND OUTER RACES ST

$0.25\,\text{IN} \times 7\,\text{IN} \times 47.6\,\text{IN} \times \pi \times 0.285\,\text{LB/IN}^3 = 74.6^{\#}$

SEALS NEOPRENE RUBBER

$1.7\,\text{IN} \times 0.25\,\text{IN} \times 47.6\,\text{IN} \times \pi \times 0.072\,\text{LB/IN}^3 = 4.6^{\#}$

SEAL RETAINERS ST

$0.55\,\text{IN} \times 0.05\,\text{IN} \times 47.6\,\text{IN} \times \pi \times 0.285\,\text{LB/IN}^3 = 1.2^{\#}$

GREASE

$0.5\,\text{IN} \times 4\,\text{IN} \times 47.6\,\text{IN} \times \pi \times 0.036\,\text{LB/IN}^3 = 10.8^{\#}$

MISCELLANEOUS $\underline{3.7^{\#}}$

TOTAL WEIGHT - SWASHPLATE BEARINGS $\underline{200^{\#}}$

## ACTUATORS, SWASHPLATE-WEIGHT CALCULATIONS

### ACTUATORS

$P_{LIMIT} = 83,300$ LBS  STROKE = 8 IN  PRESSURE = 3000 LBS
BORE DIA = 6 IN  $L_{CYL} = 33$ IN  ROD DIA = 1.75 IN
$L_{ROD} = 29$ IN

(CYL WALL) $t = \frac{3 \times P \times B}{2 \times 160,000} = \frac{3 \times 3000 \times 6}{320,000} = .2$ IN

(ROD AREA) $A = \frac{1.5 P_L}{\sigma} = \frac{1.5 \times 83,300}{120,000} = 1.04 \text{ IN}^2$

CYL WT = $B \times \pi \times t \times L_{CYL} \times 0.285 \text{ LB/IN}^3 =$

$6 \times 3.14 \times 0.2 \times 33 \times 0.285 =$  35.4 LBS

CYL ENDS AND DIVISION  ST

$B^2 \times 0.7854 \times 1.5 \times 0.285 =$  12.1 LBS

ROD WT  ST
$A \times L_{ROD} \times 0.285$

$\left(\frac{1.75}{2}\right)^2 \times \pi \times 29 \times 0.285 =$  19.9 LBS

PISTON VALVE WT  ST

$B^2 \times 0.7854 \times \Delta L \times 0.285$

$6^2 \times 0.7854 \times 1.0 \times 0.285 =$  8.1 LBS

SERVO VALVE  9.0 LBS

FLUID WEIGHT
$B^2 \times .7854 \times L \times 0.038 \text{ LB/IN}^3$

$6^2 \times 0.7854 \times 22 \times 0.038 =$  23.6 LBS

MISCELLANEOUS AND MOUNTS  21.9 LBS

WEIGHT PER ACTUATOR  130.0 LBS

3 REQ  390 LBS

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# APPENDIX II
## PRELIMINARY STRUCTURAL ANALYSIS

The preliminary structual analysis of the Heavy-Lift Hot Cycle Helicopter Rotor System is contained in this Appendix as outlined below.

I. Basic Rotor Configuration

II. Weight Data

III. Temperature Data

IV. Design Loads

V. Materials and Allowable Stresses

VI. Stress Analysis

  a. Rotor Blade
  b. Blade Retention System
  c. Hub
  d. Rotor Shaft Bearings
  e. Flight Controls
  f. Hot Gas Ducting

## I. BASIC ROTOR CONFIGURATION (Figure 16 and 17)

The rotor is composed of three blades that are attached to the hub by retention straps. The retention straps transfer the blade centrifugal force to the hub. The inboard end of the blade is mounted in the hub with a feathering bearing. The retention straps provide the flexibility to allow the blade to flap and feather about the feathering bearing.

At the connection of the retention straps to the blade is located a lead-lag flexure. The lead-lag motion is controlled by a damper mounted at the leading edge of the blade.

The hub is attached to the rotor shaft by three multimember attachment fittings spaced equally about the circumference of the shaft.

The rotor shaft is supported by a lower bearing loaded by thrust and radial load and an upper bearing loaded radially.

The swashplate is mounted on the rotor shaft support between the upper and lower bearing. The rotating swashplate is attached directly to the blade pitch arm by a single tension-compression member. This design provides a short direct load path as well as a rigid control system.

BASIC DATA

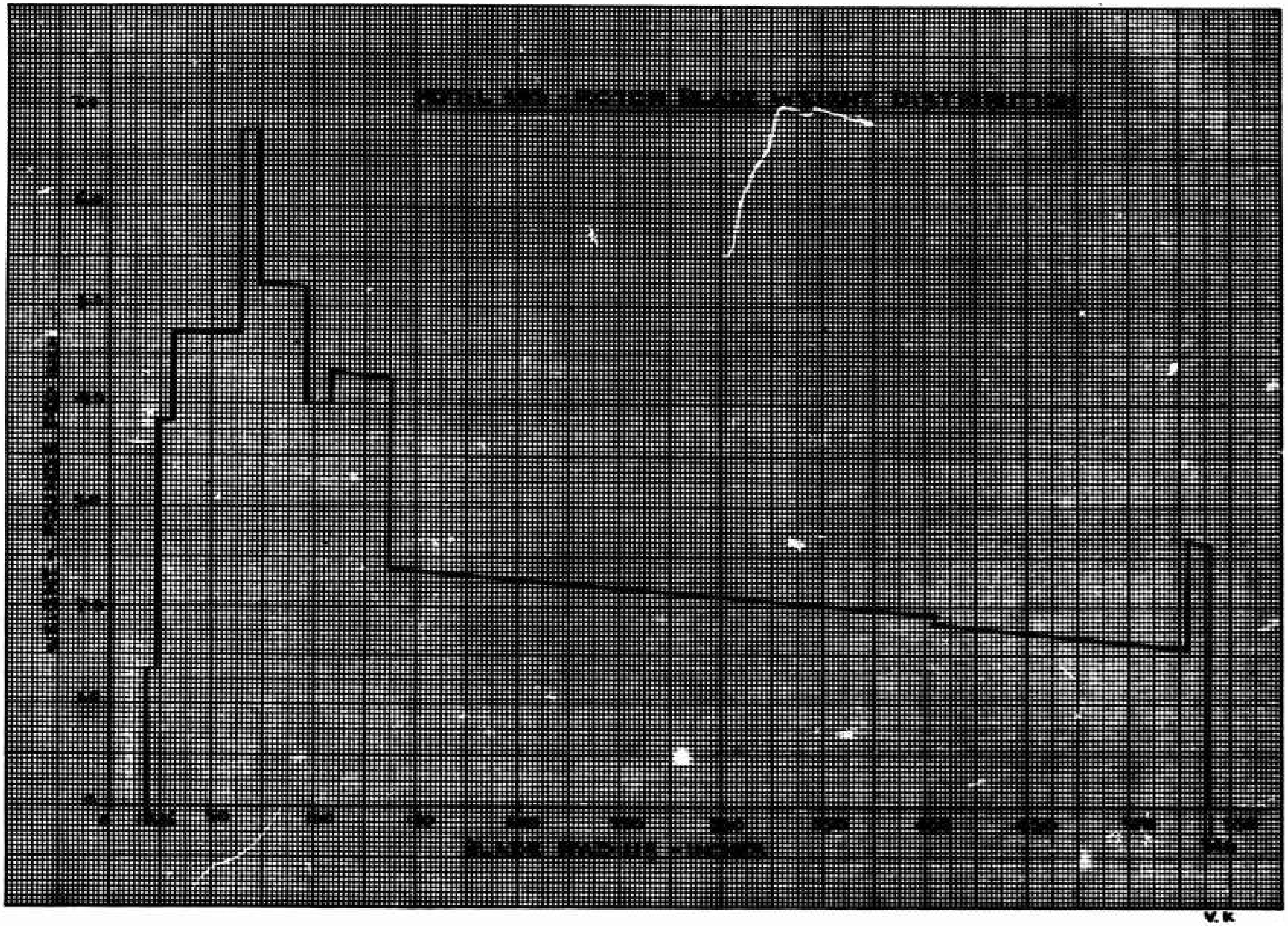
| | |
|---|---|
| Rotor radius | 45.0 ft |
| Chord | 60.0 in. |
| Airfoil section (blade) | |
| NACA 0018 | Root to 3/4 radius |
| NACA 0014 | 3/4 radius to tip |
| Number of blades | 3 |
| Blade twist | -8° |
| Flapping hinge offset (% blade radius) | 4.2% |
| Rotor shaft attitude | 5° fwd |
| Design gross weight | 65,700 lb |

## II. WEIGHT DATA

The weights used in this report were chosen early in the design program. Therefore they do not quite agree with the latest weight data based on the final preliminary design drawings.

However, the values used are conservative. The blade weight used is 1,363 pounds as against the latest value of 1,307 pounds. This results in the blade and hub being designed for slightly higher centrifugal forces than required.

The total weight of the blades, hub, and rotor-shaft used is 5,000 pounds as against the latest value of 5,400 pounds. This results in the rotor shaft and bearings being designed for slightly higher loads than required, as the greater weight would give some additional relief to counteract the lift loads.



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## III. TEMPERATURE DATA

The temperatures and gas pressures used in the analysis of the hot gas duct system are based on the engine characteristics of the General Electric GE-1 turbine engine.

The temperature spectrum for long-time operation of the ducting system is based on the mission requirements as determined from the contract. Mission requirements indicate that the gas temperatures will be below 1100°F for 80 percent of the time and below 1300°F for 95 percent of the time. Thus, the gas temperature spectrum used for structural design (see following page) is conservative.

This spectrum used for checking the ducting system for long-time operation as determined by the 0.2 percent creep allowable of the ducting material has been arbitrarily limited to a minimum gas temperature of 1300°F.

This conservatism has not caused any weight penalty, as the blade ducting gages are determined by fabrication requirements.

## III TEMPERATURE DATA

### GAS TEMPERATURE AND PRESSURE SPECTRUM

| CONDITION | TIME HOURS | GAS TEMP °F | DUCT* °F |
|---|---|---|---|
| EMERGENCY | .33 | 1490° | 1390° |
| TAKEOFF MAX | 300.00 | 1425° | 1325° |
| MILITARY | 1200.00 | 1350° | 1250° |
| MILITARY POWER STD. DAY OR NORMAL POWER HOT DAY | 2099.67 | 1300° | 1200° |
|  | 3600.00 |  |  |

*DUCT WALL TEMP = GAS TEMP. - 100°F FROM XV-9 DATA

| T °F | T+460 | t | $t_{NR}$ | $t/t_{NR}$ |
|---|---|---|---|---|
| 1390 | 1850 | .33 | 75. | .0044 |
| 1325 | 1785 | 300.00 | 468. | .6410 |
| 1250 | 1710 | 1200.00 | 4680. | .2570 |
| 1200 | 1660 | 2099.67 | 23450. | .0896 |
|  |  | 3600.00 |  | .9920 |

$(T+460)(20+\log_{10} t) = 40.47(10^3)$ $T = 1260°F$

T = LONG TIME EQUIVALENT DUCT WALL TEMPERATURE FOR 3600 HOURS OF OPERATION

### SUMMARY OF DUCT WALL TEMPERATES & PRESSURE

| CONDITION | TEMPERATURE DUCT WALL | GAS PRESSURE |
|---|---|---|
| PEAK VALUES | 1390°F | 39.6 #/IN² |
| 3600 HOUR OPERATION | 1260°F | 35.0 #/IN² |

# IV DESIGN LOADS

The basic rotor structural design criteria has been previously shown in the Structure section of this report.

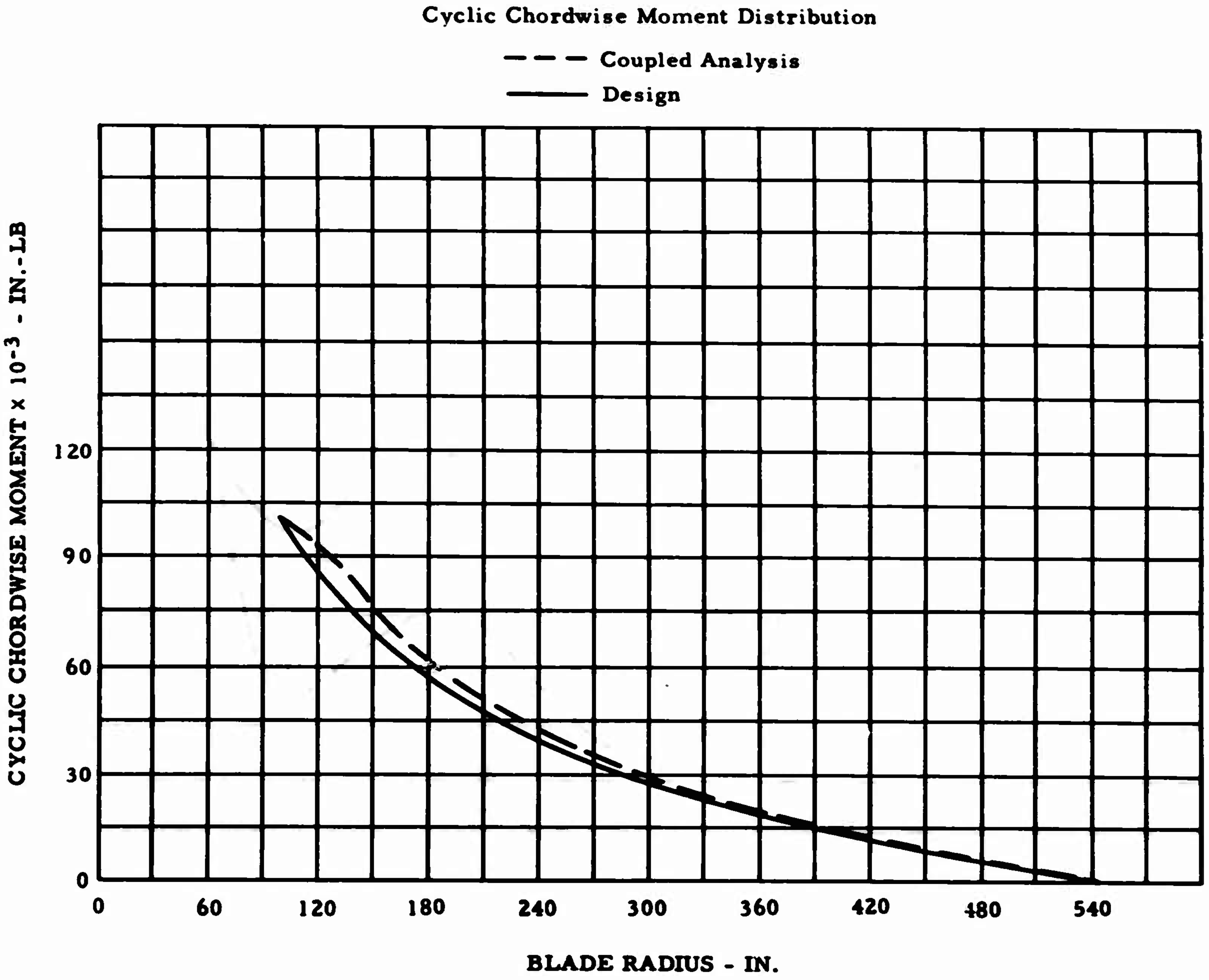
The main rotor design loads were developed from test data available from flight strain surveys on the XV-9A, OH-6, and H-34 helicopters. Since the XV-9A rotor is rigid in-plane (no lead-lag hinge), the effect of chord-flap coupling was compensated for in the use of blade loads from that helicopter.

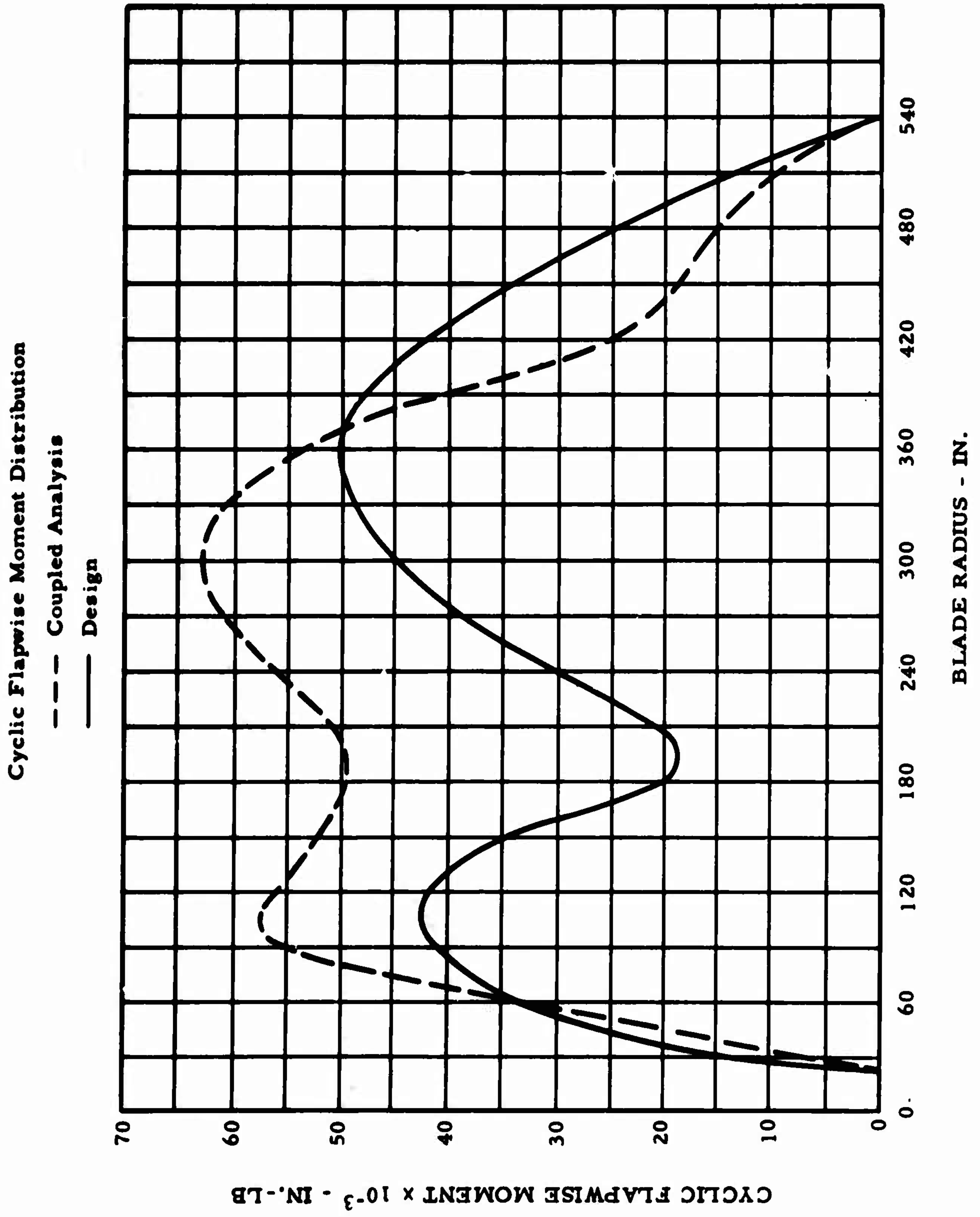
Bending moments were scaled proportional to gross weight times rotor radius, torsional moments were scaled proportional to gross weight times blade chord, and shears were scaled proportional to gross weight. Chordwise moment was based on lead-lag damping twice as large as required to prevent ground resonance.

After the blade was designed to carry the loads based on the method described above, the resulting blade properties were used in a fully-coupled dynamic analysis of the design flight condition to corroborate the design loads. The satisfactory correlation of the design loads and computed loads are shown by the following two curves on the next two pages.

The cyclic chordwise moment distribution shows good agreement between the design values and the results of the coupled analysis. The cyclic flapwise moment distribution from the coupled analysis has higher values than used in design between blade station 60 to 375. However, the blade as designed has sufficient strength to accommodate these higher values as shown in the Stress Analysis section.

The structural weight data used in determining the design loads was obtained from detailed analysis of the blade and hub design.


Cyclic Chordwise Moment Distribution
– – – Coupled Analysis
—— Design
CYCLIC CHORDWISE MOMENT × 10-3 - IN.-LB
120
90
60
30
0
0
60
120
180
240
300
360
420
480
540
BLADE RADIUS - IN.



Cyclic Flapwise Moment Distribution

## BLADE LOADS SUMMARY

| SYMBOL | LOAD DESCRIPTION | FATIGUE CONDITION | LIMIT MANEUVER |
|---|---|---|---|
| $M_{F.2R}$ | FLAPPING MOMENT | 36,011 ± 43,213 | 110,632 ± 110,632 |
| $M_{F.8R}$ | FLAPPING MOMENT | 33,031 ± 40,393 | 65,150 ± 65,150 |
| $V_Z$ | FEATHERING BALL SHEAR - VERTICAL | 839 ± 1,009 | 1,009 ± 1,009 |
| $M_{c.2R}$ | CHORDWISE MOMENT | ±100,000 | ± 500,000 |
| $M_{c.8R}$ | CHORDWISE MOMENT | ± 6,250 | ± 37,500 |
| $V_{x.2R}$ | CHORDWISE SHEAR | ± 370 | ±1850 |
| $V_{x.8R}$ | CHORDWISE SHEAR | ± 95 | ± 475 |
| $M_{T.2R}$ | BLADE TORSION | 63,000 ± 75,500 | 228,000 ± 228,000 |
| $M_{T.8R}$ | BLADE TORSION | 31,000 ± 37,000 | 112,000 ± 112,000 |
| $M_{T_{ROOT}}$ | BLADE TORSION | $64{,}000 \pm 76{,}000 + M_{STRAPS}$ | $230{,}000 \pm 230{,}000 + M_{STRAPS}$ |
| $\beta$ | FLAPPING ANGLE | $8^\circ + 5^\circ \cos(\psi - 30^\circ)$ | $13^\circ + 10^\circ \cos(\psi - 30^\circ)$ |
| $\xi$ | LEAD-LAG ANGLE | $1.25^\circ \sin\psi$ | $3^\circ \sin\psi$ |
| $\theta$ | FEATHERING ANGLE | $7^\circ - 12^\circ \sin\psi$ | $14^\circ - 14^\circ \sin\psi$ |
| – | TIP SPEED | 675 F.P.S | *787 F.P.S |

* SPARS, RETENTION STRAPS, ATTACHMENTS AND HUB ARE CHECKED FOR DIRECT LOAD FROM BLADE CENTRIFUGAL FORCE ONLY DUE TO 840 F.P.S TIP SPEED. (787 F.P.S = 105% OF 750 F.P.S)

---

A SECOND LIMIT CONDITION IS CONSIDERED IN WHICH ALL LOADS AND MOTIONS ARE THE SAME AS ABOVE, EXCEPT FLAPPING ANGLE BECOMES $15^\circ + 10^\circ \cos(\psi - 30)$ AND TIP SPEED IS 675 F.P.S

- CONT. -

BLADE MOTION CAPABILITY IS PROVIDED AS FOLLOWS:

| | |
|---|---|
| FLAPPING | +25°<br>-6.0° |
| COLLECTIVE PITCH (BLADE) | +14° MAX (AT 75% RAD)<br>+1° MIN. |
| COLLECTIVE PITCH (CONTROLS) | +14°<br>±1° |
| CYCLIC PITCH - FORWARD | 14° |
| - AFT. | 14° |
| - LATERAL | ± 6° |

BLADE RETENTION STRAPS ARE UNTWISTED WHEN THE 75% RADIUS BLADE STATION IS AT 8°

| | |
|---|---|
| LEAD LAG MOTION | ± 1.25° CRUISE<br>± 3.00° MANEUVER |

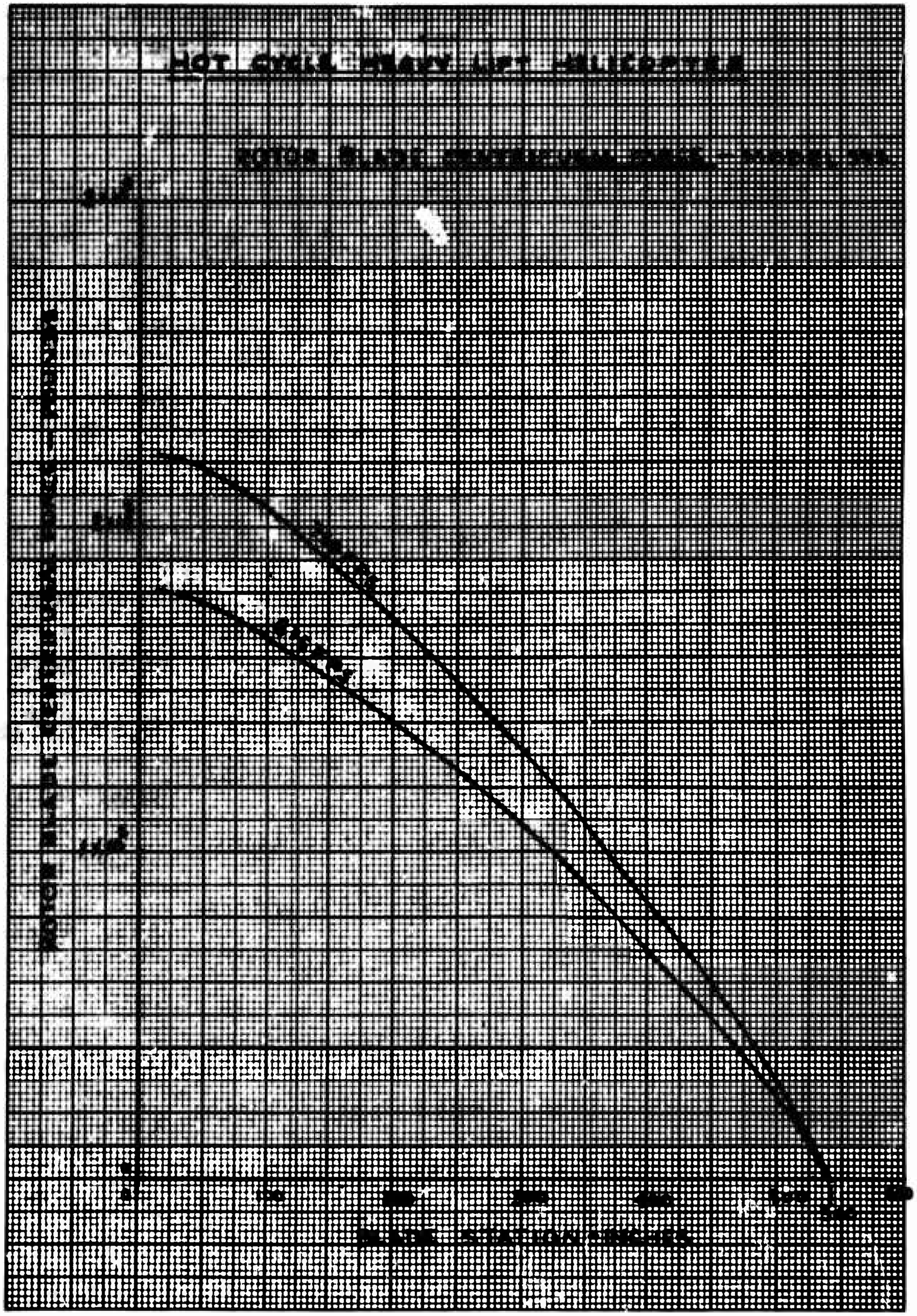
## IV DESIGN LOADS

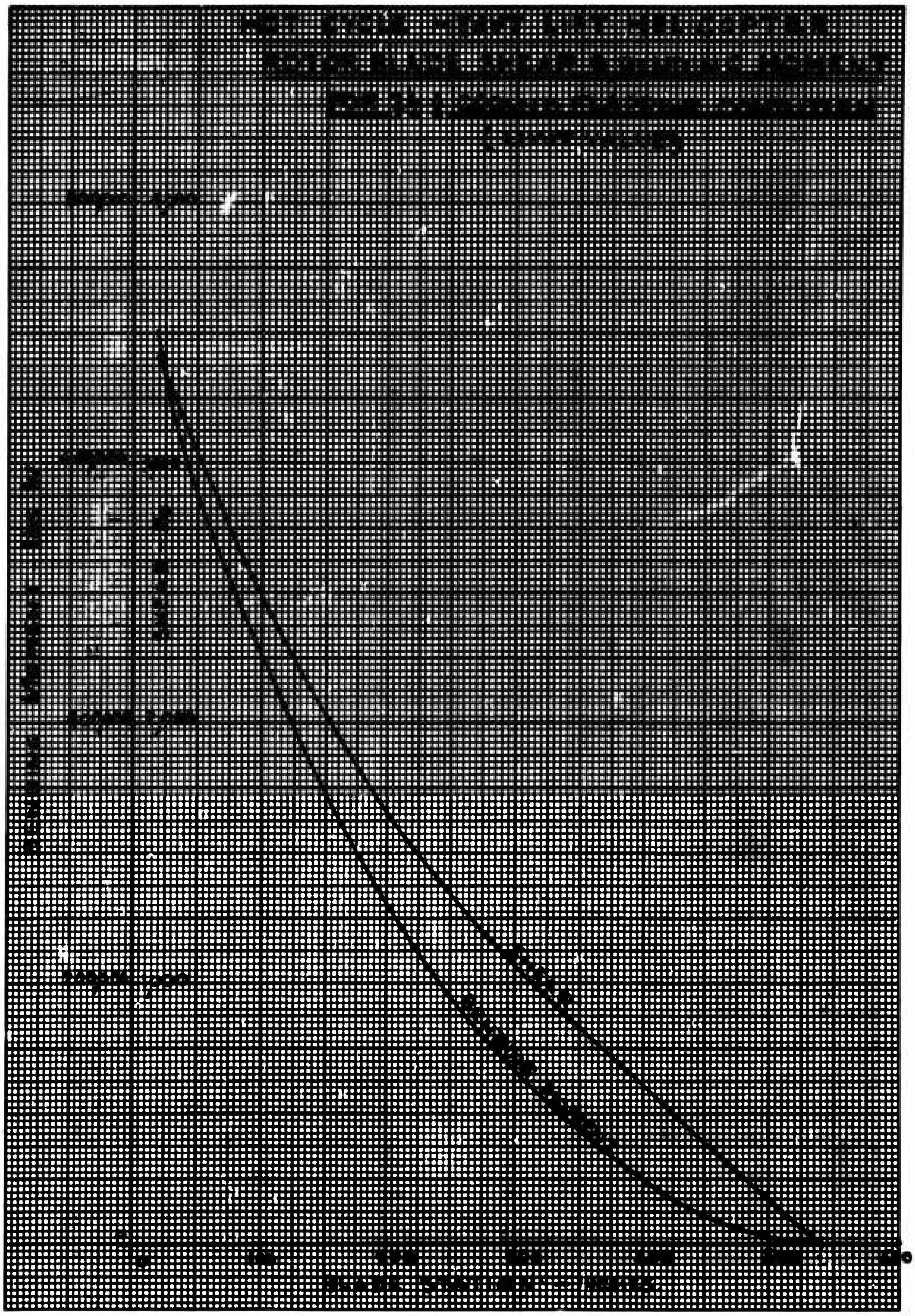
- BLADE CENTRIFUGAL FORCE AT 675 F.P.S TIP SPEED

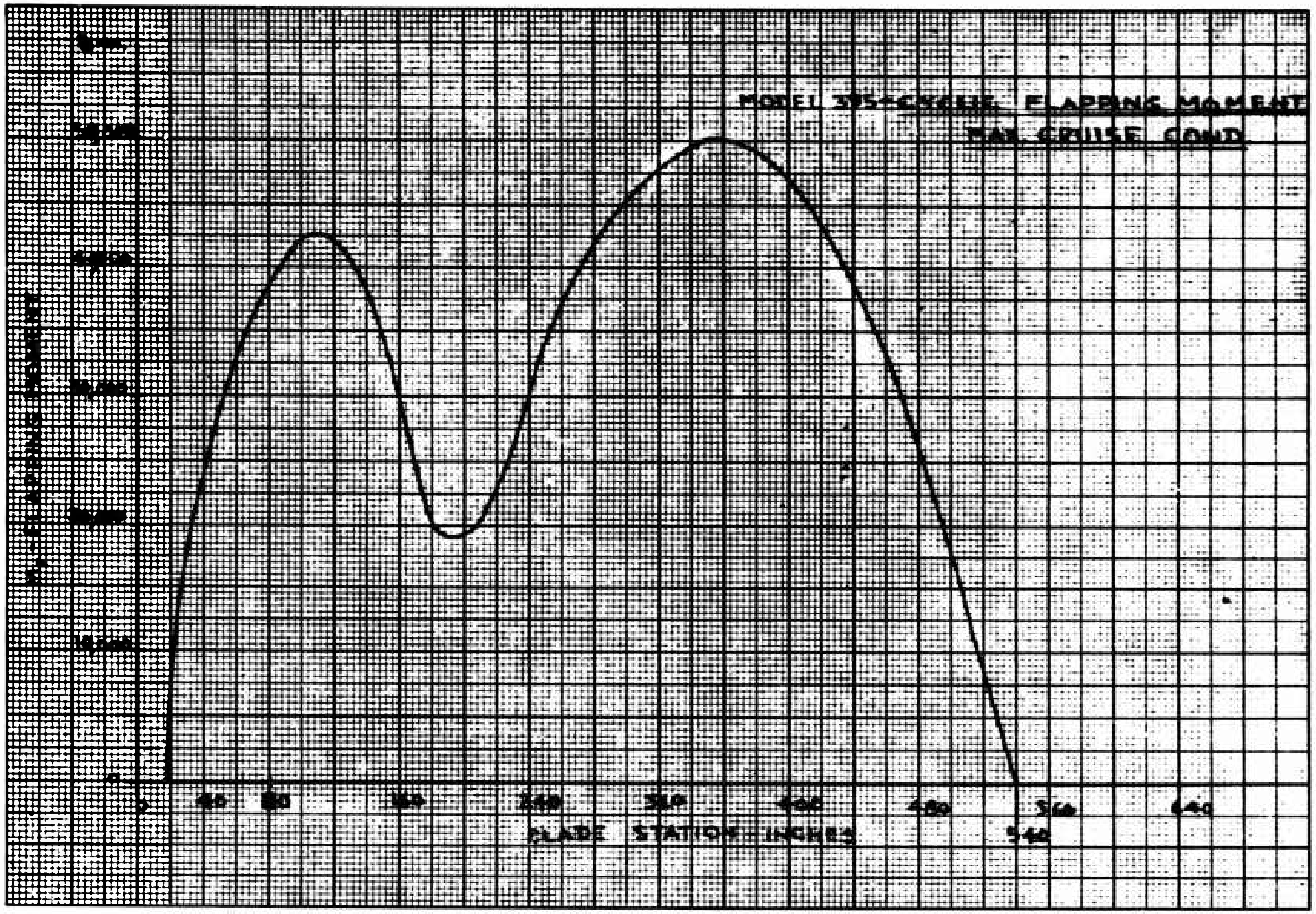
| STA | ΔSTA. | #/IN AVG. | $\Delta W_T$ | $\lambda$ RADIUS TO C.G. | $g = \frac{\lambda \omega^2}{386}$ | $\Delta C.F. = \frac{\Delta W_T \lambda \omega^2}{386}$ | TOTAL C.F @ 675FPS | TOTAL C.F @ 750 FPS |
|---|---|---|---|---|---|---|---|---|
| 18 | | | | | | | 180,455 | 222,320 |
| | 6 | 1.350 | .8 | 21 | 12.20 | 98 | | |
| 24 | | | | | | | 180,357 | 222,199 |
| | 8 | 3.85 | 30 | 28 | 16.30 | 489 | | |
| 32 | | | | | | | 179,868 | 221,597 |
| | 33 | 4.72 | 155 | 48.5 | 28.24 | 4,377 | | |
| 65 | | | | | | | 175,491 | 216,205 |
| | 9 | 6.75 | 60 | 69.50 | 40.50 | 2,430 | | |
| 74 | | | | | | | 173,061 | 213,211 |
| | 23 | 5.20 | 119 | 85.50 | 49.80 | 5,926 | | |
| 97 | | | | | | | 167,135 | 205,910 |
| | 11 | 4.00 | 44 | 102.50 | 59.43 | 2,615 | | |
| 108 | | | | | | | 164,520 | 202,687 |
| | 34 | 4.30 | 146 | 125 | 72.82 | 10,632 | | |
| 142 | | | | | | | 153,[illegible] | 189,590 |
| | 160 | 2.25 | 360 | 222 | 129.40 | 46,580 | | |
| 302 | | | | | | | 107,308 | 132,203 |
| | 73 | 2.02 | 147 | 338.50 | 197.32 | 29,006 | | |
| 375 | | | | | | | 78,302 | 96,460 |
| | 30 | 1.940 | 58 | 390 | 227.35 | 13,186 | | |
| 405 | | | | | | | 65,116 | 80,223 |
| | 45 | 1.750 | 76.50 | 427.5 | 248.87 | 19,038 | | |
| 450 | | | | | | | 46,078 | 56,768 |
| | 79 | 1.650 | 130. | 489.5 | 285 | 37,050 | | |
| 529 | | | | | | | 9,028 | 11,122 |
| | 11 | 2.650 | 29 | 534.5 | 311.30 | 9,028 | | |
| 540 | | | | | | | 0 | 0 |
| TOTAL | | | 1362.50 | | | | | |

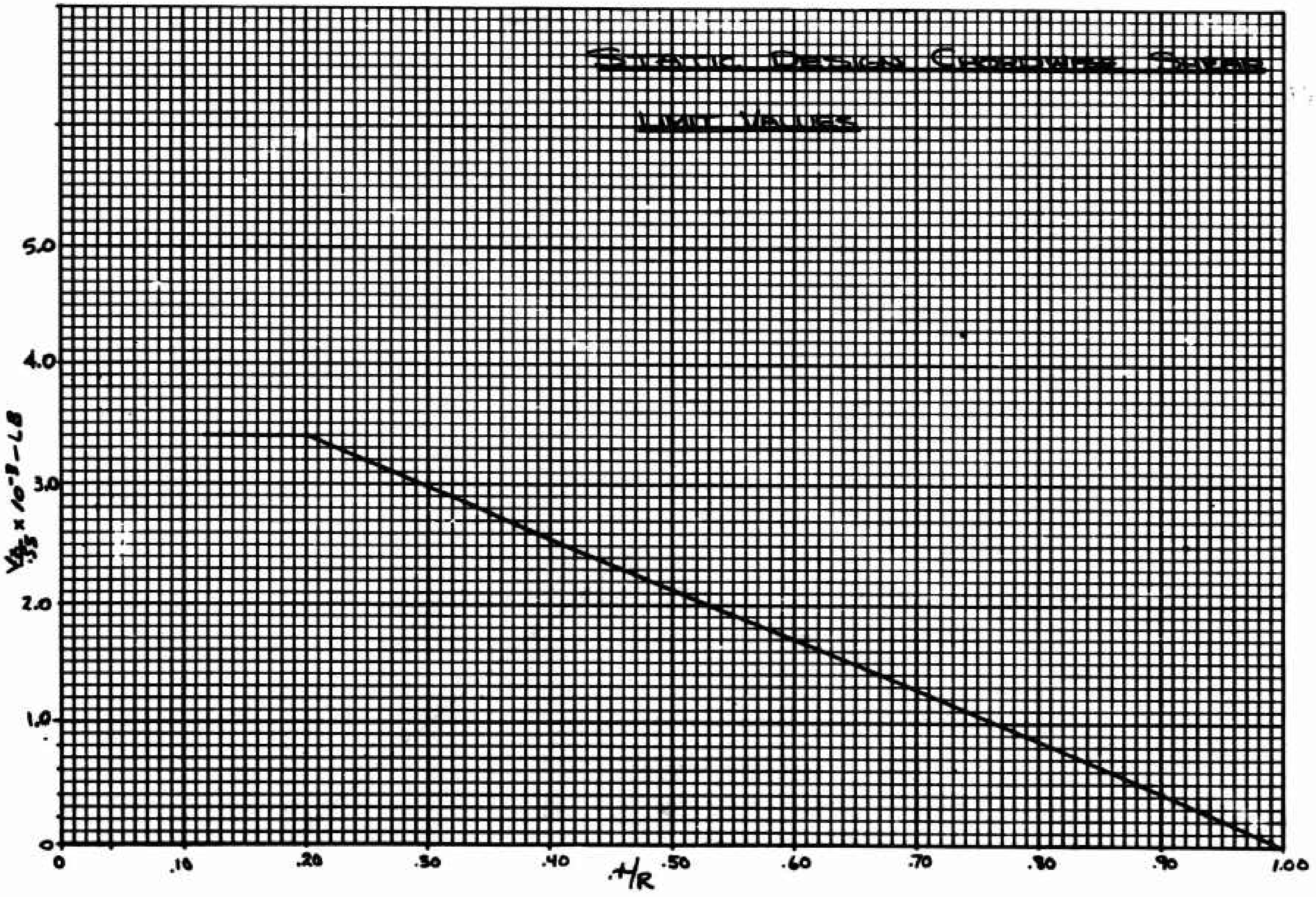
$* \frac{675 \text{ FPS}}{2\pi\lambda} = \frac{675}{2\pi(45)} = 2.38 \text{ R.P.S}, \ \omega = 2.38 \times 2\pi = 15 \text{ Rad/sec.} \ ; \ \omega^2 = 225$

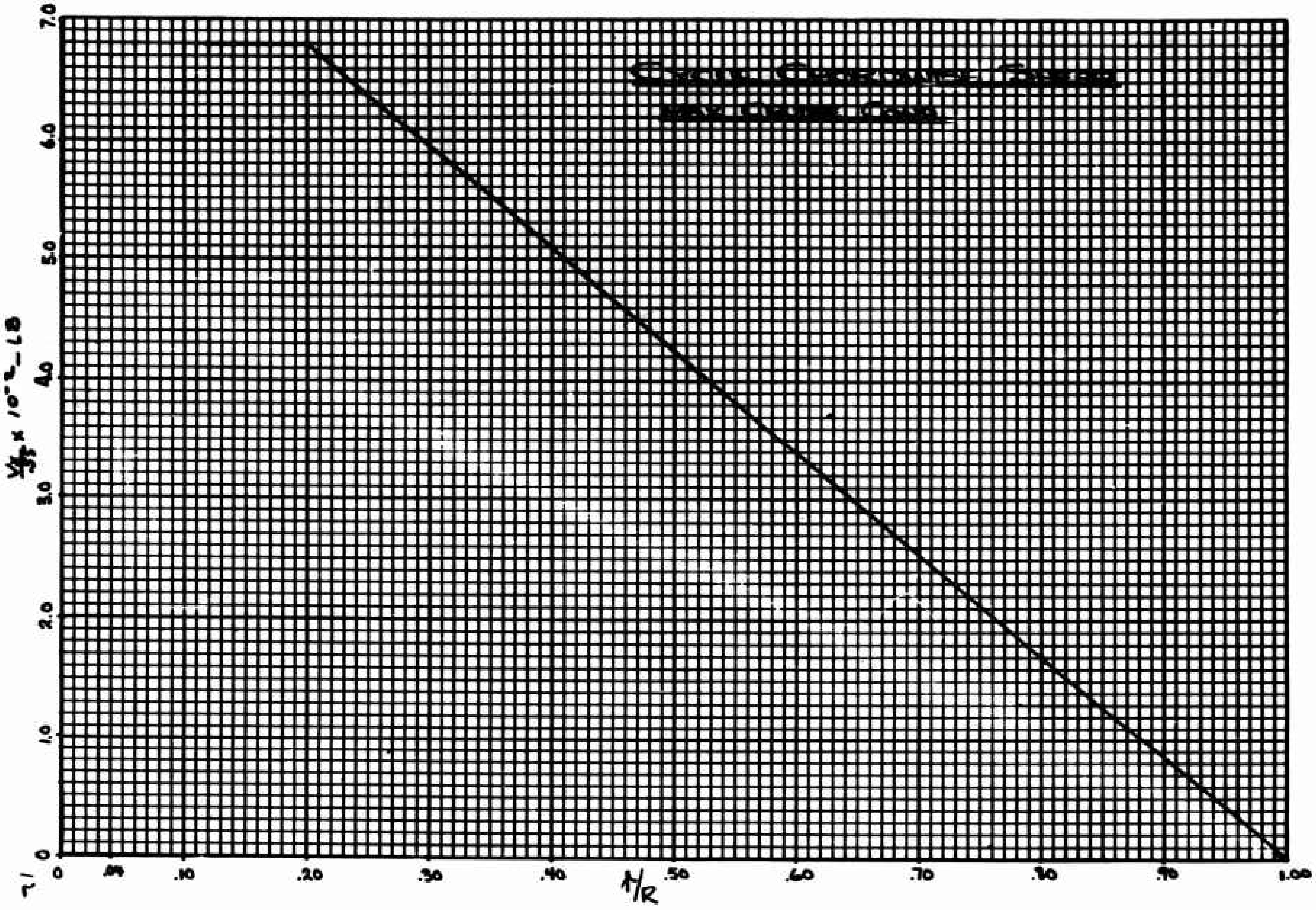
3784


HOT CYCLE HEAVY LIFT HELICOPTER
ROTOR BLADE CENTRIFUGAL FORCE
BLADE STATION - INCHES

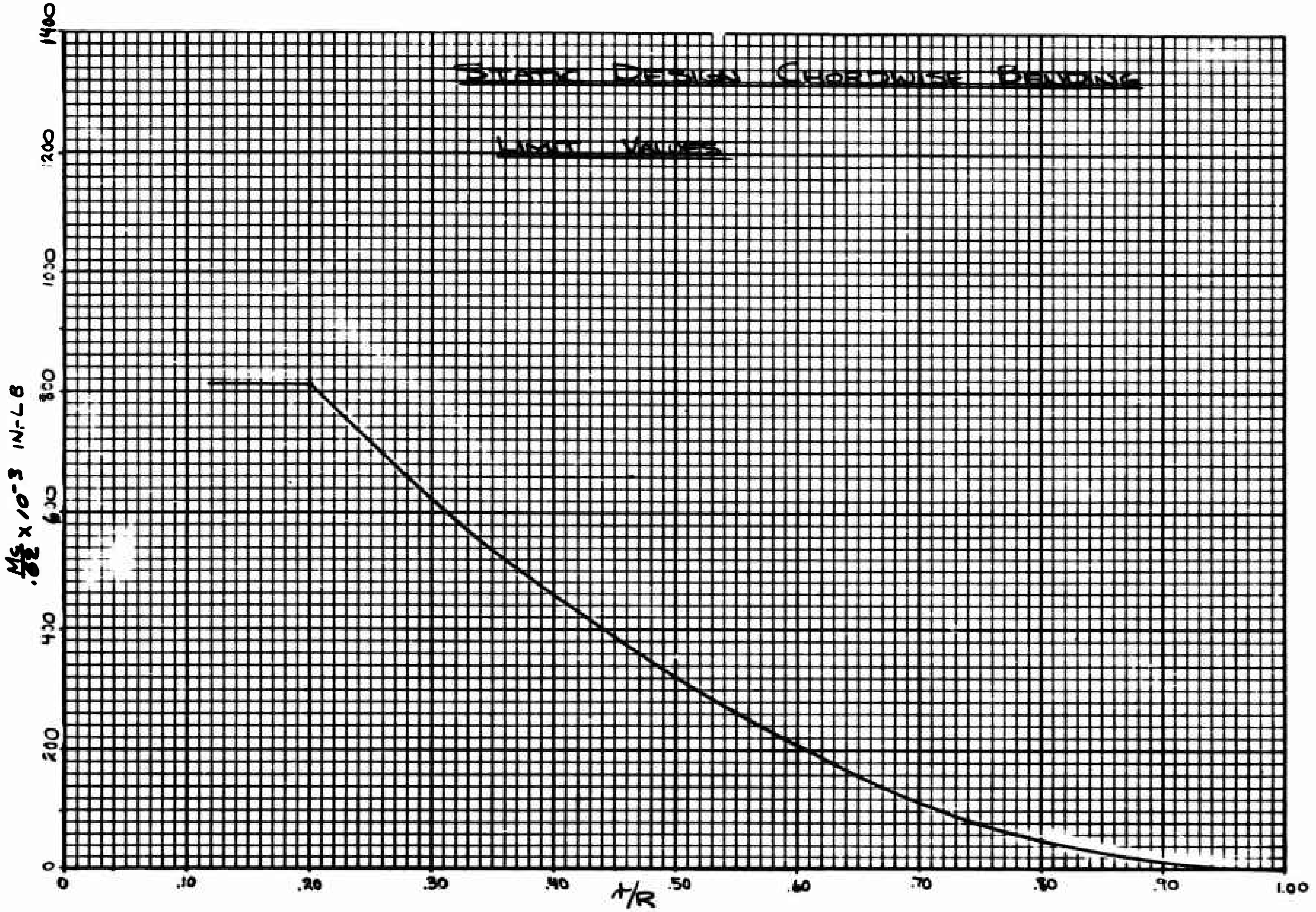


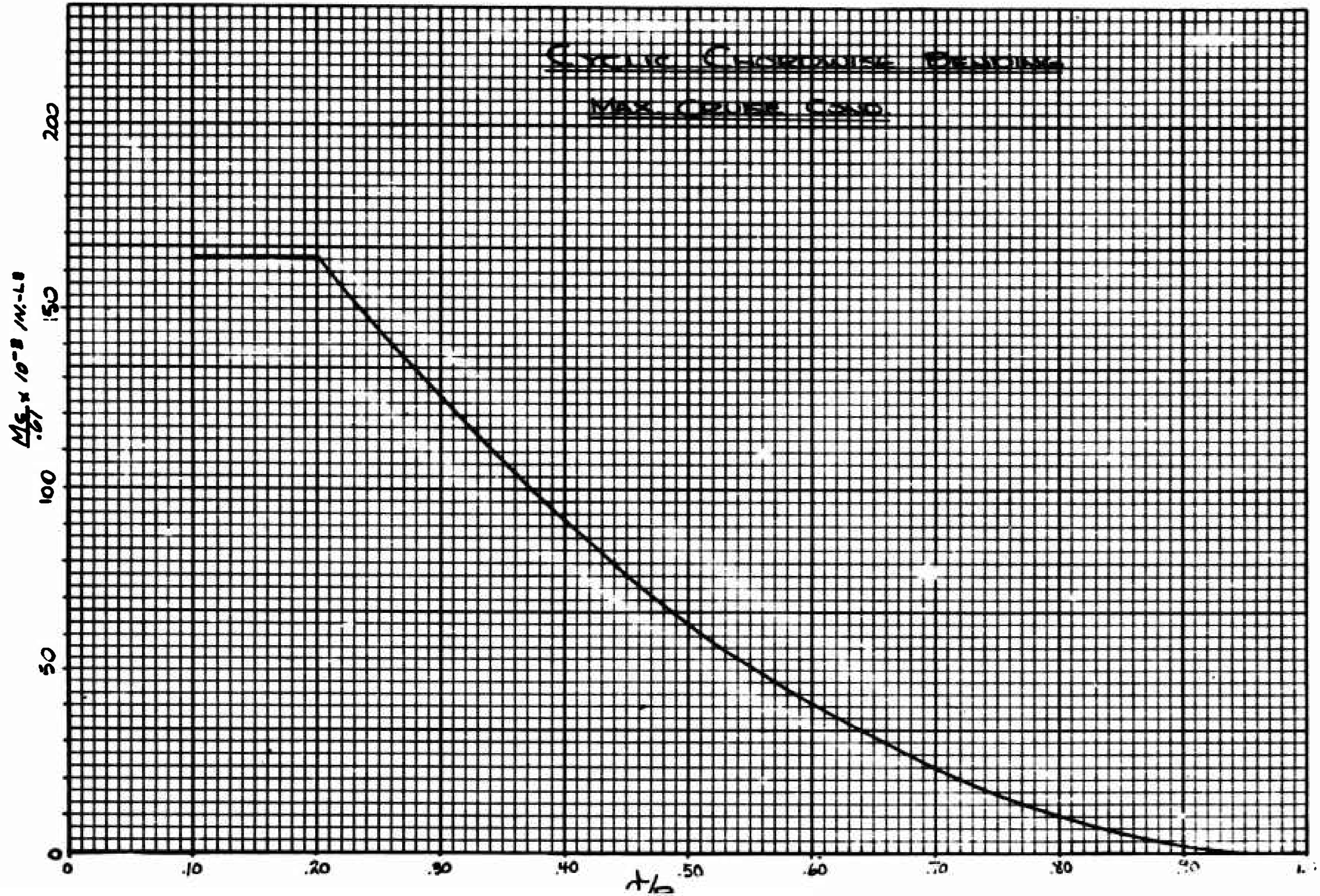

MODEL 395-CYCLIC FLAPPING MOMENT
MAX CRUISE COND
M_F - FLAPPING MOMENT
0
10,000
20,000
30,000
40,000
50,000
60,000
0
40
80
160
240
320
400
480
560
640
540
BLADE STATION - INCHES

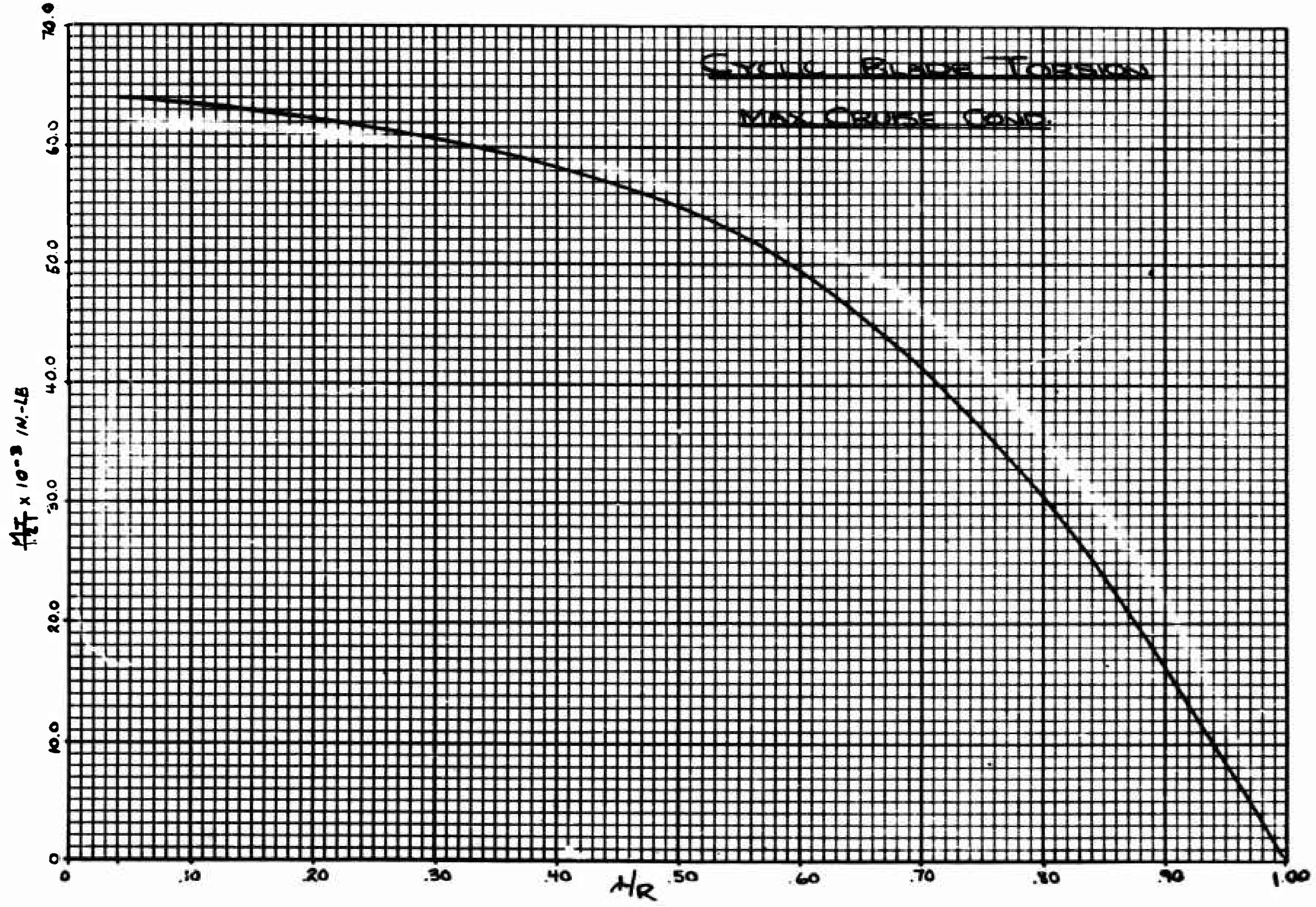
STATIC DESIGN CHORDWISE SHEAR
LIMIT VALUES
5.0
4.0
3.0
2.0
1.0
0
$V_{35} \times 10^{-3}$ – LB
0
.10
.20
.30
.40
.50
.60
.70
.80
.90
1.00
r/R

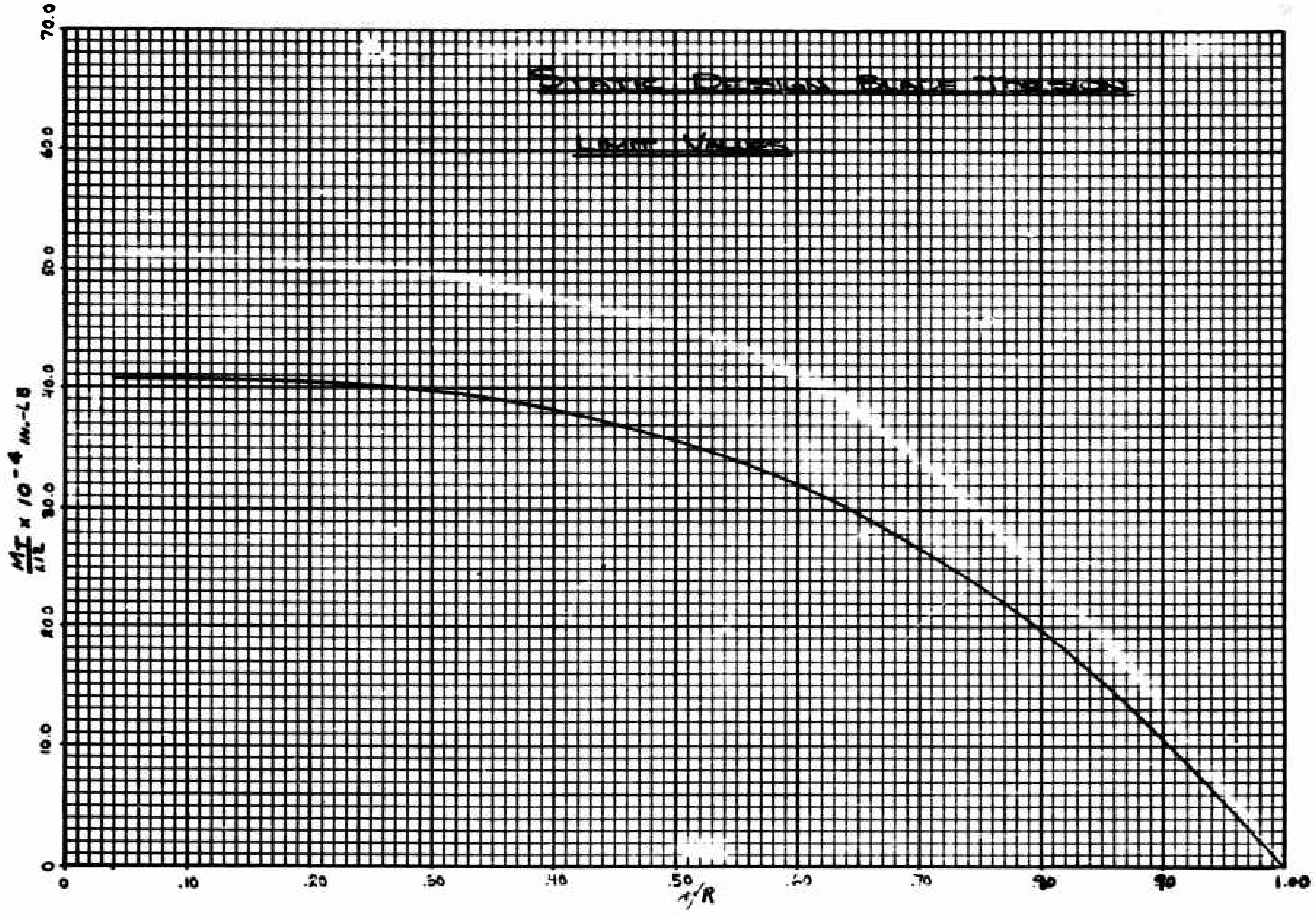


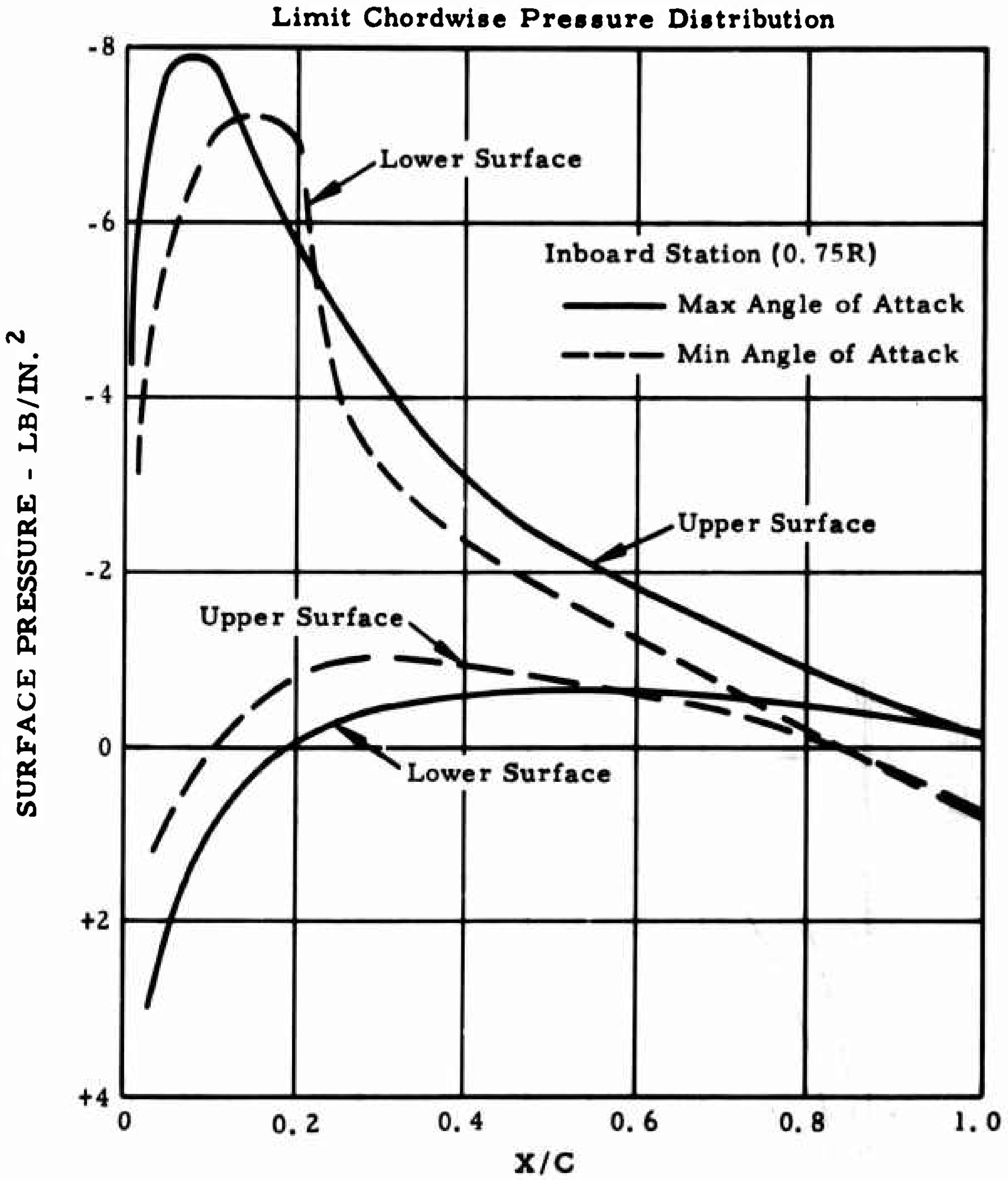
7.0
6.0
5.0
4.0
3.0
2.0
1.0
0
$V_{35} \times 10^{-2}$ – LB
r/
0
.05
.10
.20
.30
.40
.50
.60
.70
.80
.90
1.00
t/R


STATIC DESIGN CHORDWISE BENDING
LIMIT VALUES
$\frac{M_S}{.02} \times 10^{-3}$ IN-LB
0
200
400
600
800
1000
1200
1400
0
.10
.20
.30
.40
.50
.60
.70
.80
.90
1.00
r/R


CYCLIC CHORDWISE BENDING
MAX CRUISE COND.
$\frac{M_S}{.07} \times 10^{-3}$ IN-LB
0
50
100
150
200
r/R
0
.10
.20
.30
.40
.50
.60
.70
.80
.90
1.0

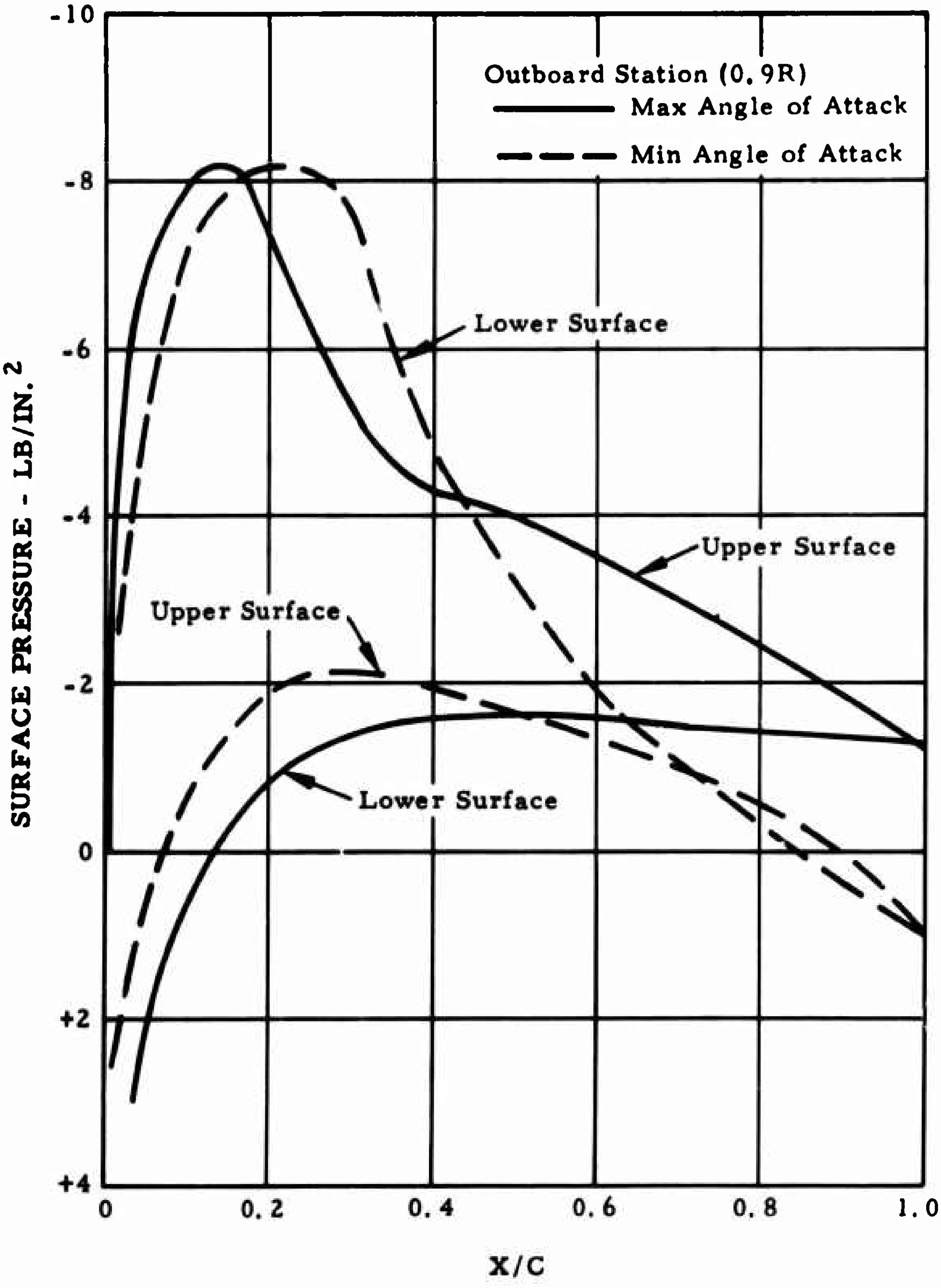

CYCLIC BLADE TORSION
MAX CRUISE COND.
$\frac{M_T}{\frac{1}{2}T}$ x $10^{-3}$ IN.-LB
70.0
60.0
50.0
40.0
30.0
20.0
10.0
0
0
.10
.20
.30
.40
r/R
.50
.60
.70
.80
.90
1.00

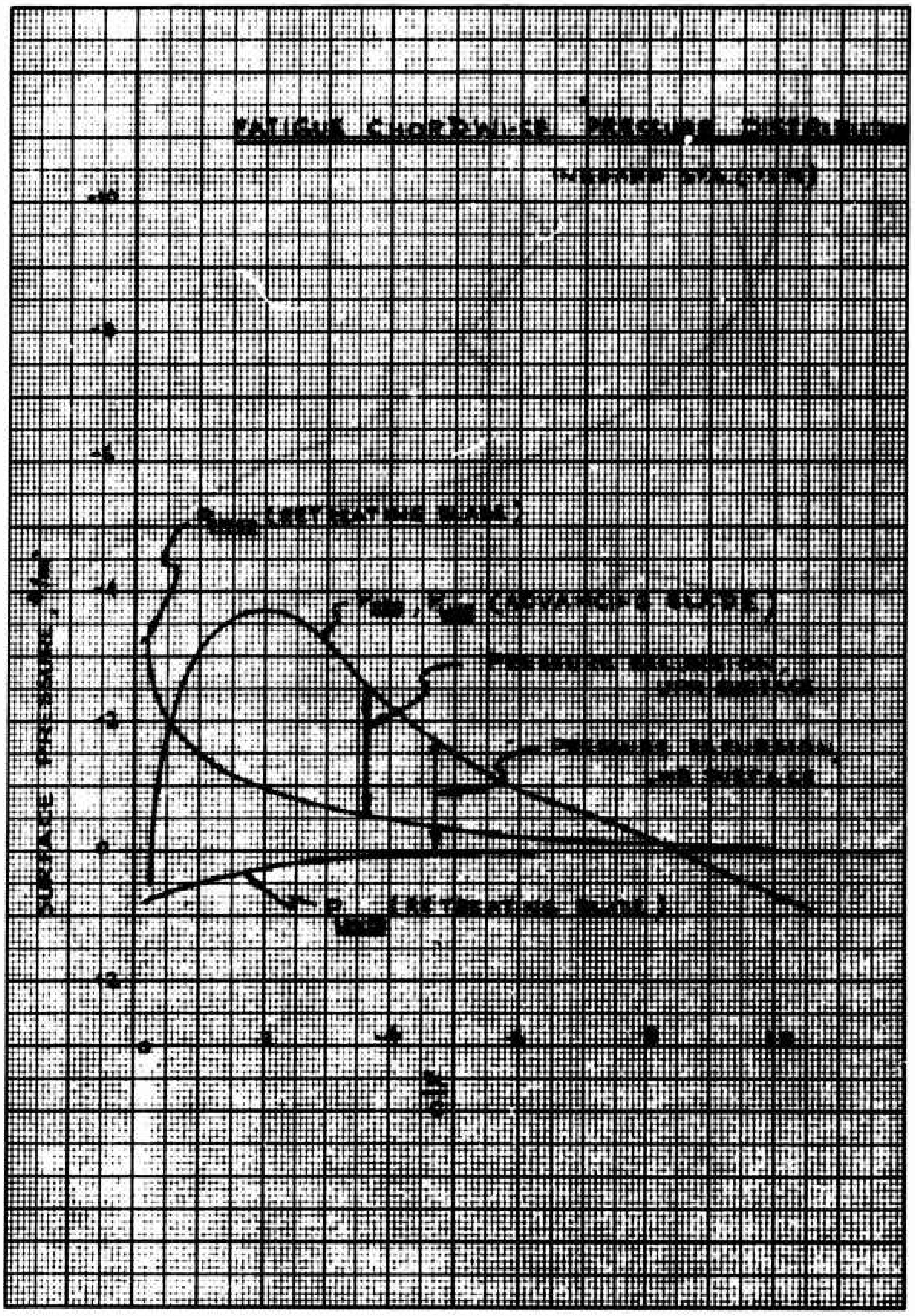
STATIC DESIGN BLADE TORSION
LIMIT VALUES
$\frac{M_T}{1/2} \times 10^{-4}$ IN.-LB
70.0
60.0
50.0
40.0
30.0
20.0
10.0
0
0
.10
.20
.30
.40
.50
.60
.70
.80
.90
1.00
r/R



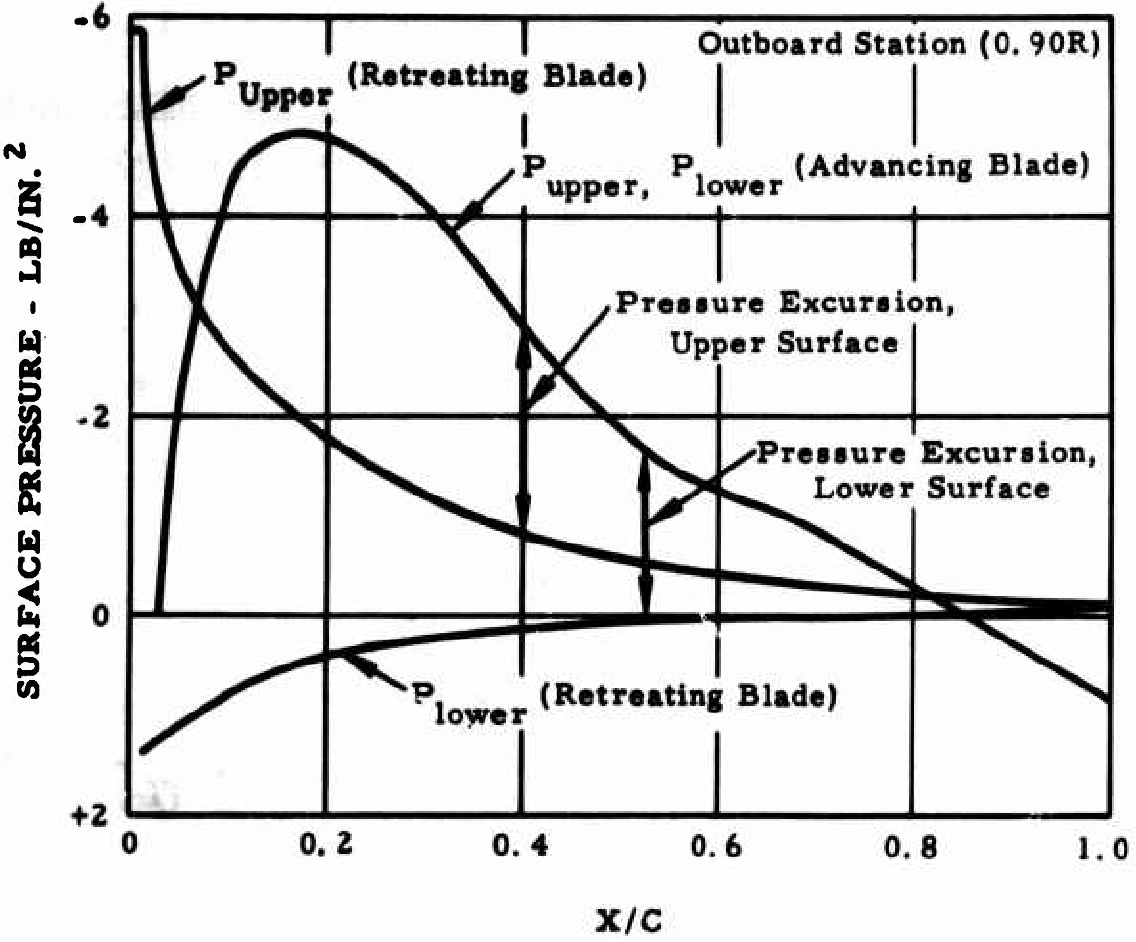
Limit Chordwise Pressure Distribution
SURFACE PRESSURE - LB/IN.$^2$
-8
-6
-4
-2
0
+2
+4
0
0.2
0.4
0.6
0.8
1.0
X/C
Lower Surface
Inboard Station (0.75R)
Max Angle of Attack
Min Angle of Attack
Upper Surface
Upper Surface
Lower Surface

Limit Chordwise Pressure Distribution

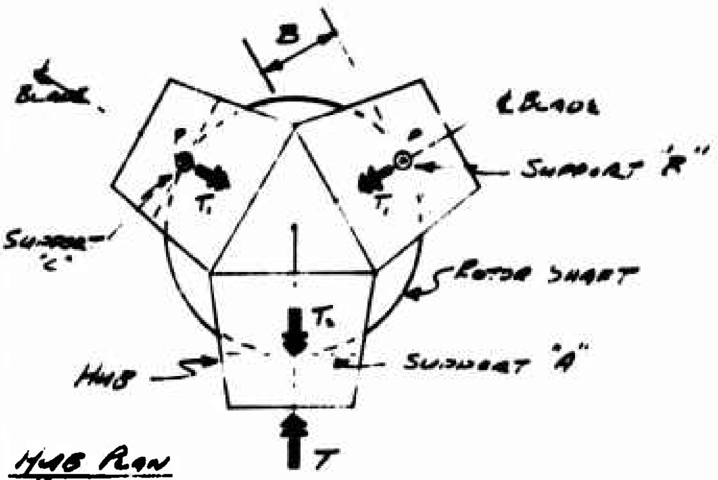



FATIGUE CHORDWISE PRESSURE DISTRIBUTION
-10
-8
-6
-4
-2
0
2
SURFACE PRESSURE
(RETREATING BLADE)
(ADVANCING BLADE)
(RETREATING BLADE)
0
.2
.4
.6
.8
1.0
x/c

Fatigue Chordwise Pressure Distribution



## ROTOR HUB LOADS



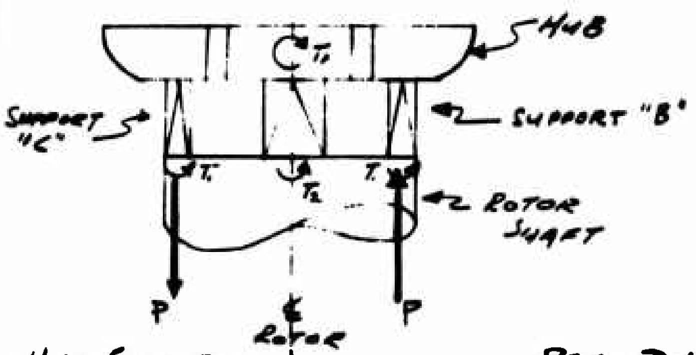
HUB PLAN VIEW

LEGEND—

T = UNBALANCED BLADE TORQUE ACTING ON THE HUB.

$T_1$ & $T_2$ = TORQUE REACTION DUE TO MOMENT STIFFNESS OF THE SUPPORT

P = COUPLE REACTION DUE TO AXIAL STIFFNESS OF THE SUPPORT



HUB ELEVATION VIEW

REF - DWG 395-0936

THE ROTOR HUB SUPPORT REACTIONS ARE REDUNDANT FOR ANY UNBALANCED BLADE TORQUE APPLIED TO THE HUB. THIS REDUNDANT ANALYSIS OF SUPPORT REACTIONS TAKES INTO ACCOUNT THE BENDING AND TORSIONAL STIFFNESS OF THE HUB AND THE AXIAL AND MOMENT STIFFNESS OF THE SUPPORTS.

- CONT. -

## ROTOR HUB LOADS

THE FOLLOWING EQUATIONS AND PARAMETERS ARE THE RESULT OF THE REDUNDANT ANALYSIS:

$$T_1 = T\left[\frac{1-\left(\frac{E+2D}{E+F+2D}\right)}{\frac{E+F+D}{D}-\frac{2D}{E+F+2D}}\right]$$

$$T_2 = T\left[1-\left\{\frac{1-\left(\frac{E+2D}{E+F+2D}\right)}{\left(\frac{E+F+D}{D}-\frac{2D}{E+F+2D}\right)}\right\}\left(\frac{E+F+D}{D}\right)\right]$$

$$P = \left(\frac{T-T_2-T_1}{1.732\,B}\right) \quad ; \quad E = \frac{D}{GJ} \quad ; \quad F = \frac{1}{K_T}$$

$$D = \frac{B}{9EI} + \frac{1}{3K_p B^2}$$

$B = 16$ IN.

$G = 11 \times 10^6$ PSI

$J = 94$ IN. (TORSIONAL STIFFNESS OR HUB CROSS-SECT.)

$F = \frac{1}{K_T} = .0222 \frac{\text{RAD}}{\text{IN.-\#}}$ (TORSIONAL STIFFNESS OF SUPPORT.)

$E = 30 \times 10^6$ PSI

$I = 43.4$ IN.$^4$

$K_p = 4.53 \times 10^6 \frac{\#}{\text{IN}}$ (AXIAL STIFFNESS OF SUPPORT.)

# ROTOR HUB LOADS:

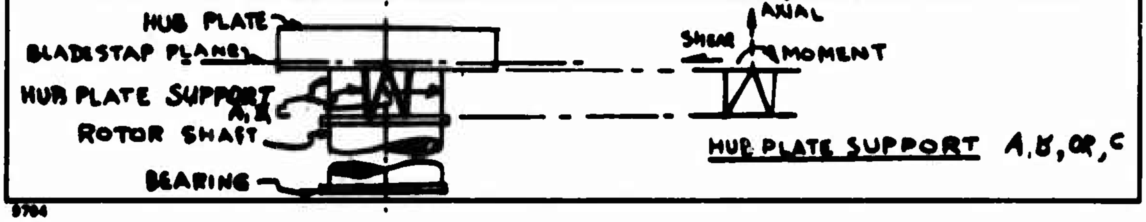
## • HUB PLATE SUPPORT LOADS:

### FATIGUE CONDITION:

| LOAD ON HUB PLATE | SUPPORT A | | | SUPPORT B | | | SUPPORT C | | |
|---|---|---|---|---|---|---|---|---|---|
| | MOM. | SHEAR | AXIAL | MOM. | SHEAR | AXIAL | MOM | SHEAR | AXIAL |
| LIFT | - | - | 35,000<br>±38,440 | - | - | 35,000<br>∓19,350 | - | - | 35,000<br>∓19,350 |
| C.F | - | 0 | - | - | ±2770 | - | - | ∓2770 | - |
| $V_x$ | - | ±740 | - | - | ∓310 | - | - | ∓370 | - |
| TORQUE | ±21,186 | - | 0 | ±5290 | - | ±1,070 | ±5290 | - | ∓1070 |
| $M_c$ | - | - | - | - | - | - | - | - | - |
| TOTAL | ±21,186 | ±740 | 35,000<br>±38,440 | ±5,290 | ±2400 | 35,000<br>∓18,280 | ±5,290 | ∓3,140 | 35,000<br>∓20,420 |

### LIMIT MANEUVER CONDITION (LIMIT VALUES SHOWN)

| LOAD ON HUB PLATE | SUPPORT A | | | SUPPORT B | | | SUPPORT C | | |
|---|---|---|---|---|---|---|---|---|---|
| | MOM. | SHEAR | AXIAL | MOM. | SHEAR | AXIAL | MOM. | SHEAR | AXIAL |
| LIFT | - | - | +192,500 | - | - | +37,500 | - | - | 37,500 |
| C.F | - | 0 | - | - | -11,080 | - | - | -11080 | - |
| $V_x$ | - | 3,700 | - | - | +1850 | - | - | +1850 | - |
| TORQUE | 193,920 | - | - | 38,790 | - | +9,490 | 38,790 | - | -9,490 |
| $M_c$ | - | - | - | - | - | - | - | - | - |
| TOTAL | 193,920 | 3,700 | 192,500 | 38,790 | 9,230 | 46,990 | 38,790 | 9,230 | 28,010 |



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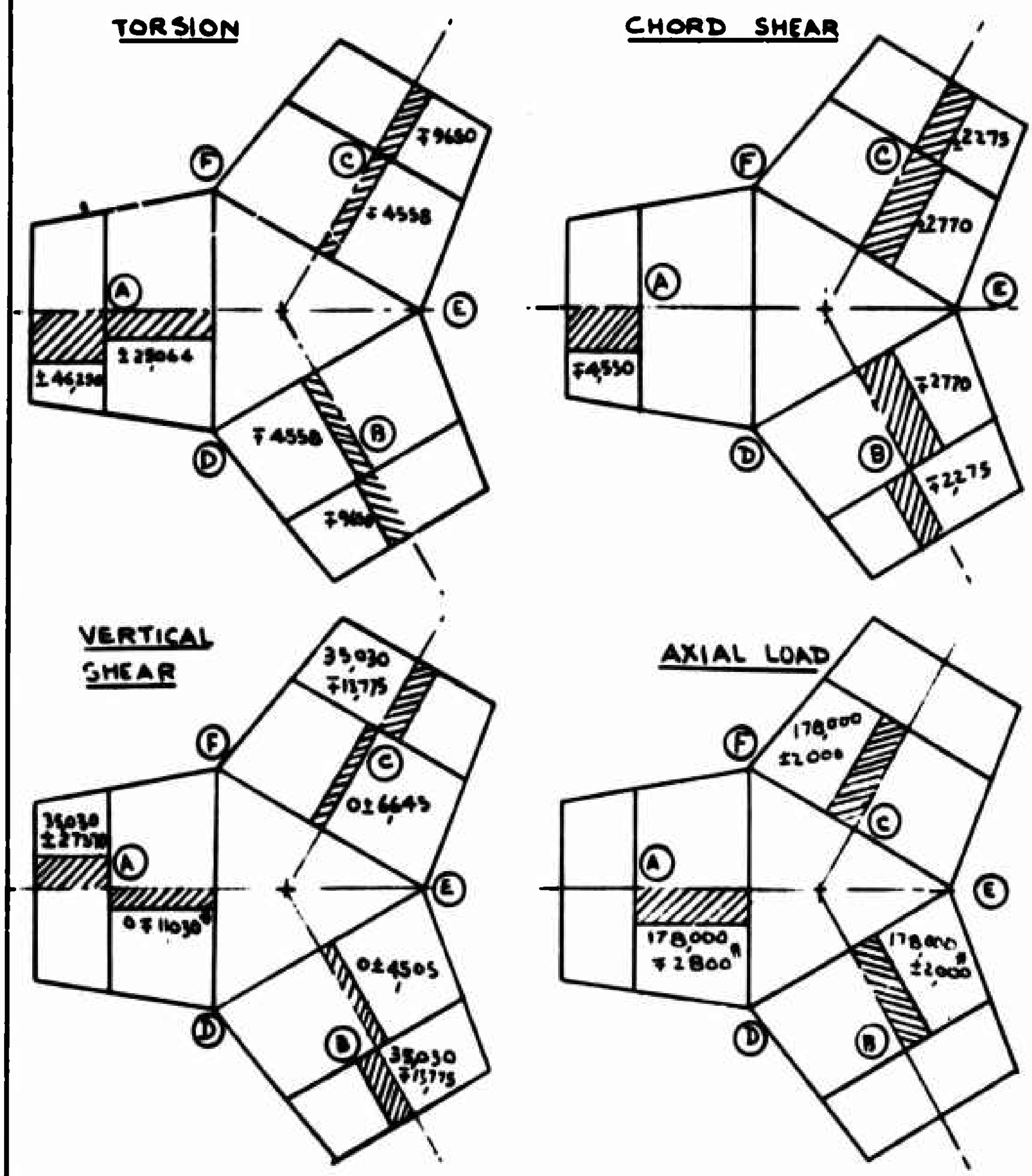
# ROTOR HUB LOADS FATIGUE CONDITI



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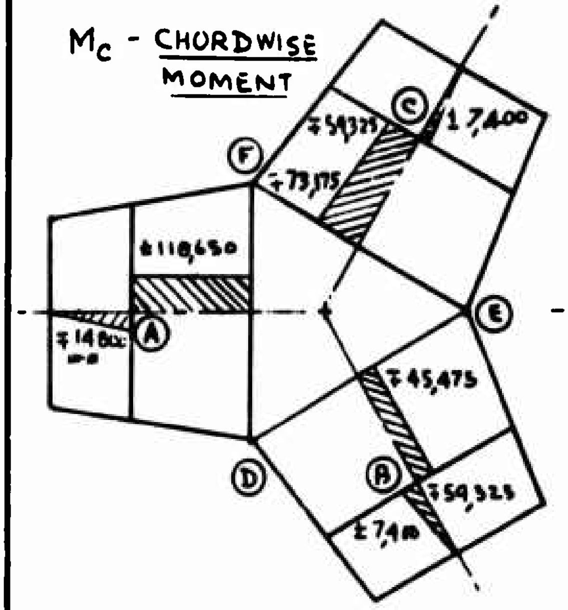
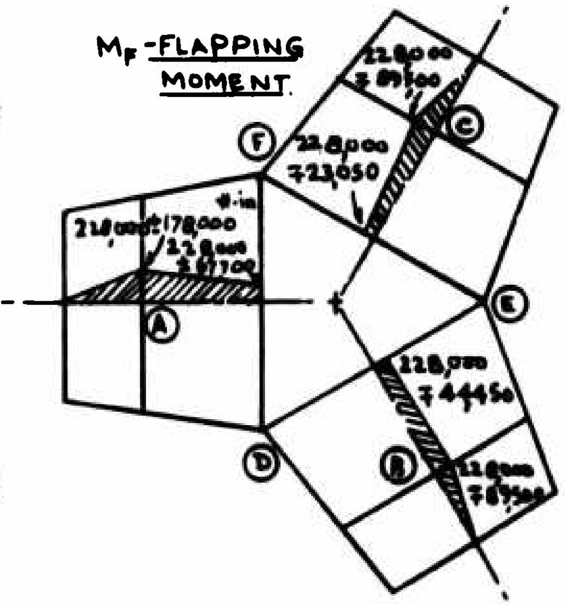
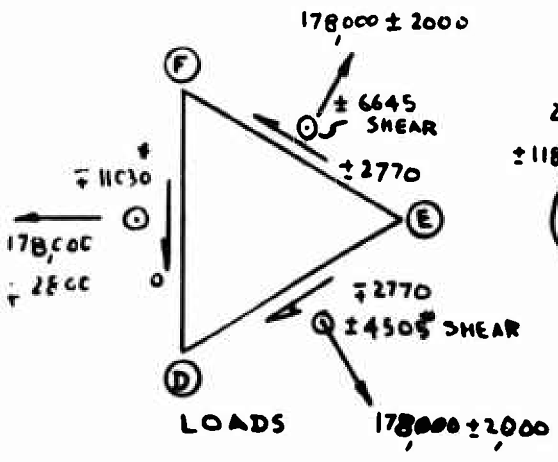
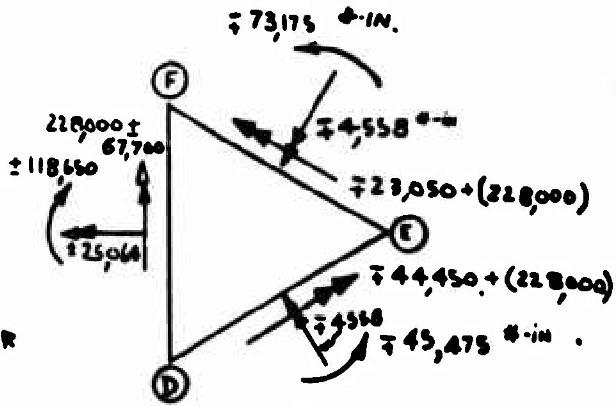
## ROTOR HUB LOADS: FATIGUE CONDITION





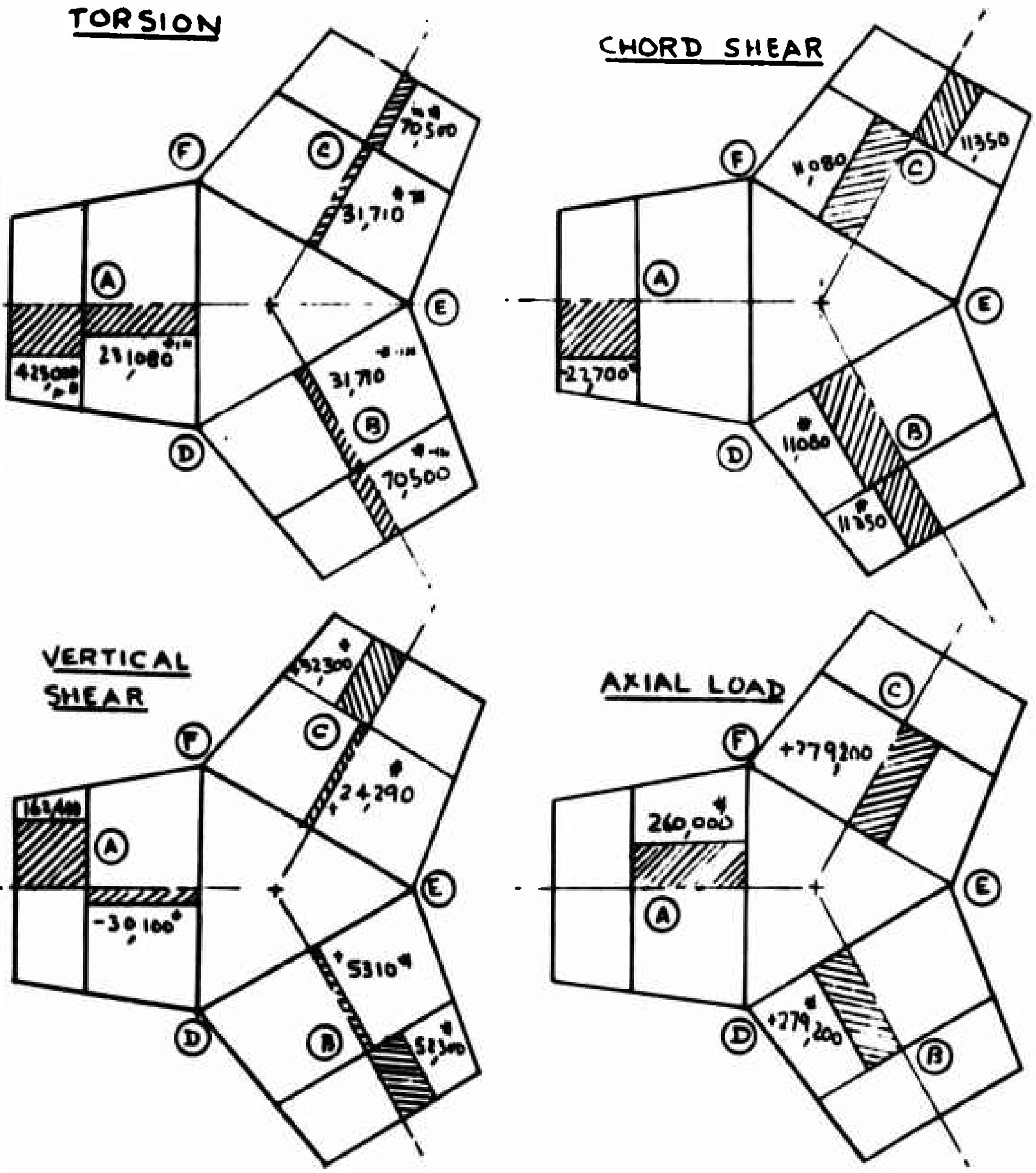


LOADS

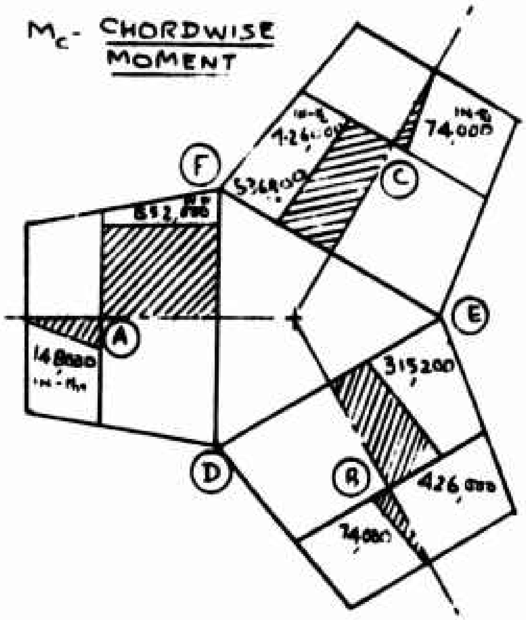


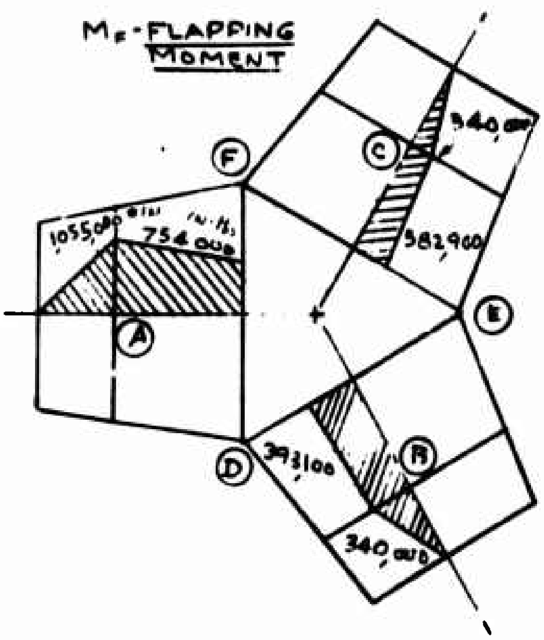
REACTIONS

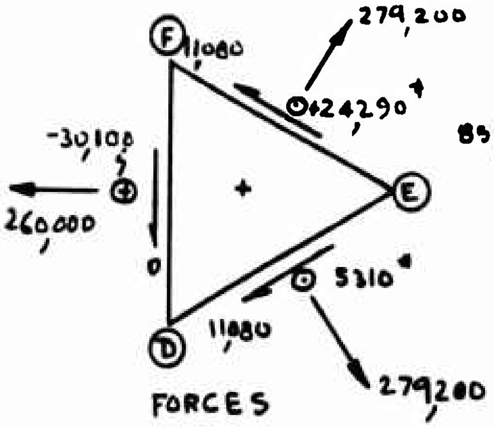
## ROTOR HUB LOADS: STATIC CONDITION



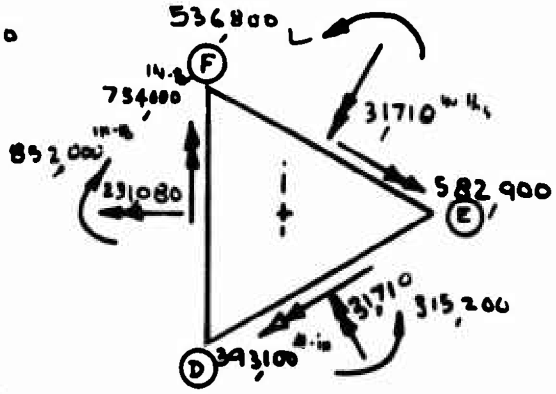
## ROTOR HUB LOADS: STATIC CONDITION







FORCES



MOMENTS

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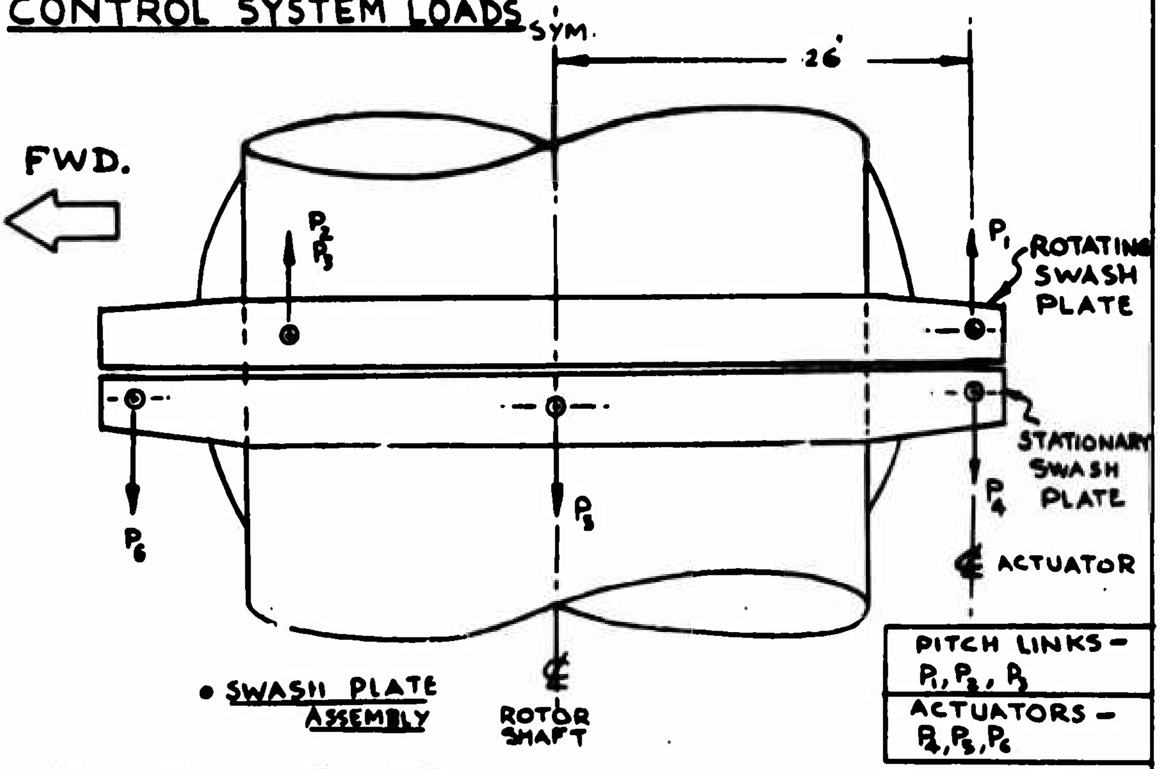
## CONTROL SYSTEM LOADS



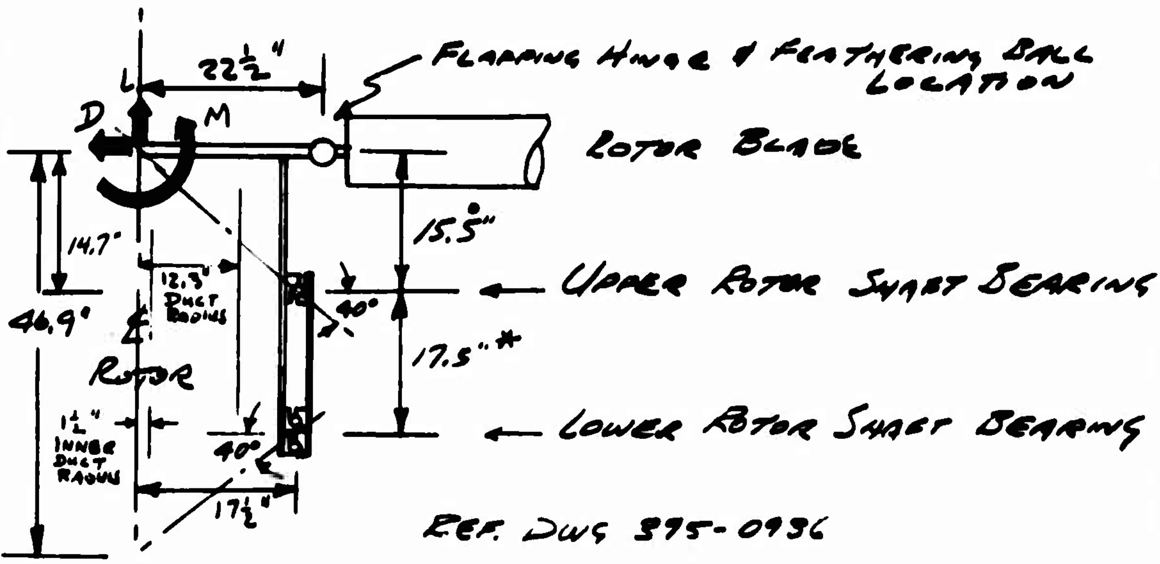
REF. DWG 395-0936

| DESCRIPTION OF LOADS | FATIGUE CONDITION | | | | LIMIT MAN. COND.* | | | |
|---|---|---|---|---|---|---|---|---|
| $M_{T_{ROOT}}$ - BLADE ROOT MOMENT | 64,000 ± 90,580 IN-LB. | | | | 242,250 ± 252,320 IN-# | | | |
| $P_1$ - PITCH LINK LOAD | 4,925 ± 6,970 | | | | 18,700 ± 19,400 | | | |
| SWASH PLATE - BEARING THRUST | 14,775 # | | | | 56,100 # | | | |
| SWASH PLATE - BEARING MOMENT | 271,000 #-IN | | | | 757,000 IN-LB | | | |
| ACTUATOR LOADS | LONGITUDINAL PITCHING | | LATERAL PITCHING | | LONGITUD PITCHING | | LATERAL PITCHING | |
| LONGITUDINAL ACTUATORS | MAX. | MIN. | MAX. | MIN. | MAX. | MIN. | MAX. | MIN. |
| ACTUATOR-$P_6$ (FWD.) | +12,600 | +2,200 | 12,600 | 2200 | 42,650 | 13,550 | 42,600 | 13,550 |
| ACTUATOR-$P_4$ (AFT.) | +2,200 | 12,600 | 12,600 | 2200 | 13,550 | 42,650 | 42,600 | 13,550 |
| LATERAL ACTUATOR – $P_5$ | 0 | 0 | -10,400 | +10,400 | 0 | 0 | -29,100 | +29,100 |

* LIMIT VALUES SHOWN

## HEAVY LIFT HOT CYCLE ARTICULATED ROTOR BLADE

### ROTOR SHAFT BEARING LOADS



REF. DWG 395-0936

ROTOR BEARING LOADS

| CONDITION | L # | D # | M IN # |
|---|---|---|---|
| FATIGUE COND. | 94,590 | 5840 | ±530,000 |
| LIMIT MANEUVER COND.** | 233,150 | 28.650 | 1,655,000 |

L = ROTOR LIFT − ROTOR INERTIA + GAS LOAD + LOAD FROM BLADE TORSION

| CONDITION | UPPER BEARING | | LOWER BEARING | |
|---|---|---|---|---|
| | RADIAL | AXIAL | RADIAL | AXIAL |
| FATIGUE COND. | 5950±1300 | 0 | 100±1,300 | 94,410# |
| LIMIT MANEUVER COND** | 6440 | 0 | 35,750 | 223,700# |

** LIMIT VALUES SHOWN (NOT CRITICAL)

* THIS DIMENSION IS NOW 27¾" GIVING 57.15" BETWEEN BEARING CENTERS. THE EQUIVALENT BEARING LOAD, P, USED IN ANALYSIS IS 2% GREATER THAN REQUIRED, IF THIS CORRECTION IS NEGLECTED

$$P = XF_R + YF_A = .5 \times 11400^{\#} \times \frac{46.9}{57.15} + .478(94590) = 49970^{\#} \quad \Omega \; \frac{50935^{\#}}{49970} - 1 = 2\%$$

## HEAVY LIFT HOT CYCLE ARTICULATED ROTOR BLADE

ROTOR SHAFT BEARING LOADS

LIMIT MANEUVER COND.

ROTOR RADIUS = 45'    TIP SPEED = 840 F.P.S.

$n = 18.65$    BLADE C.F. = 282000#

HUB MOMENT = $1.5(\text{BLADE C.F.})\, r \sin a_1$

$r = 22.5\text{IN}$    $a = 10°$

$M = 1.5(282000)22.5(\overset{.1736}{\sin 10°}) = 1,655,000$

$L = \text{LIFT} - \text{ROTOR WEIGHT} = (66000-5000)2\tfrac{1}{2}g = 152,500$

$D = 2\tfrac{1}{2}g(66000)\sin 10° = 28,650\#$

UPPER BEARING

$$P_R = \frac{28650\#}{} \times \frac{477}{46.9} + \frac{1655,000}{46.9} = 64400\#$$

$P_A = 0$

LOWER BEARING

$$P_R = 28650\# \times \frac{.8}{46.9} + \frac{1.655,000}{46.9} = 35750\#$$

$P_A = 152,500 + 18,550 + 52,650 = 223,700\#$

LOAD DUE TO GAS PRESSURE

$[12.3^2 - 1.5^2]\pi\ 39.6\ \text{psi} = 18550\#$

LOAD DUE TO BLADE TORSION

$\frac{228000\ \text{in}\#}{13''} = 17,550\#$    17,550 × 3 BLADES = 52,650#

# IV DESIGN LOADS

## HEAVY LIFT HOT CYCLE ARTICULATED ROTOR BLADE

ROTOR SHAFT BEARING LOADS

FATIGUE COND.

ROTOR RADIUS = 45'     TIP SPEED = 675 F.P.S.

$\Omega$ = 15. RADIANS / SEC.     BLADE C.F. = 180000#

HUB MOMENT = 1.5(BLADE C.F.) r sin $a_1$

$r = 22.5$ IN.     $a_1 = \pm 5^\circ$

$$M = 1.5(180000)\,22.5(\overset{.0872}{\sin 5^\circ}) = 530,000^{\text{in}\#}$$

L = LIFT − ROTOR WT = 66000 − 5000 = 61000#

D = 66000# sin 5° = 5840#

UPPER BEARING

$$P_R = 5840^{\#} \times \frac{47.7}{46.9} \pm \frac{530000}{46.9} = \underline{5940^{\#} \pm 11300^{\#}}$$

$$P_A = \frac{5840}{2} \tan 40^\circ = 2460\,\#$$

LOWER BEARING

$$P_R = 5840^{\#} \times \frac{.8}{46.9} \pm \frac{530000}{46.9} = 100 \pm 11300^{\#}$$

$$P_A = 61000^{\#} + 16400^{\#} + 14550^{\#} + 2460^{\#} = \underline{94,410\,\#}$$

LOAD DUE TO GAS PRESSURE

$$[12.3^2 - 1.5^2]\,\pi\ 35\ \text{PSI} = 16400^{\#}$$

LOAD DUE TO BLADE TORSION

$$\frac{63,000}{13}{}^{*\#} = {}^{*}4849^{\#} \qquad 4849 \times 3\ \text{BLADES} = 14550^{*}$$

*Revised 12 Aug 66

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## V. MATERIAL ALLOWABLES

The materials to be used for the Hot Cycle Heavy-Lift Helicopter have been selected on the basis of the greatest strength-to-density ratio suitable for the temperature environments and fatigue and static loads used in this design. The design conditions are very similar to the temperature and static and fatigue conditions successfully handled by Hughes on the XV-9A Hot Cycle research aircraft.

Design allowables for the following materials are presented in this section:

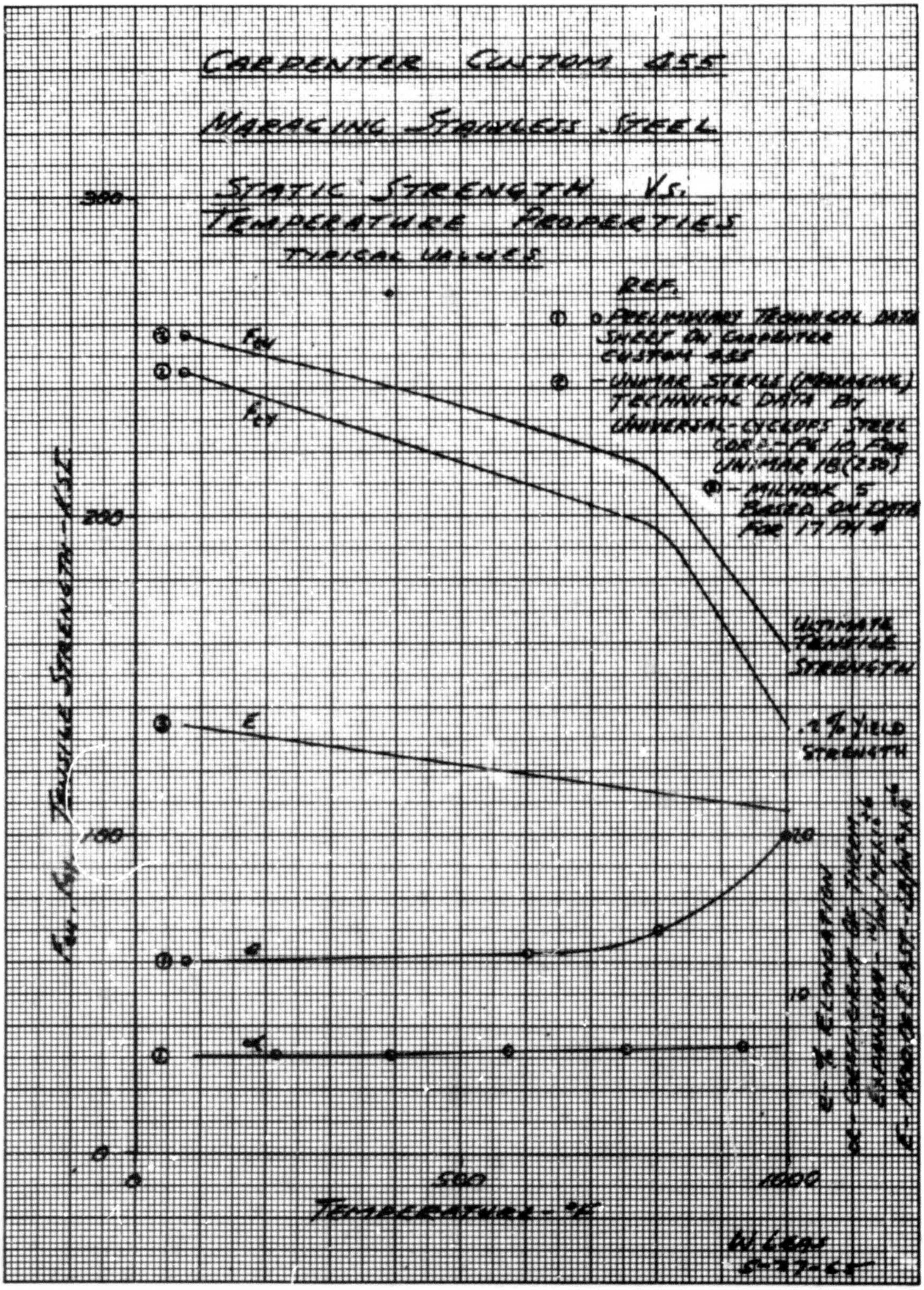
**Steel** - Carpenter 455 maraging steel is proposed for the spar material because of its high static strength-to-density ratio. It also exhibits exceptionally consistent fatigue properties for both smooth specimens and those with holes.

It shows only a slight dropoff in strength due to temperature at the expected 300° to 400°F environment. It also performs satisfactorily for short-time temperature conditions up to 1000°F.

**Titanium Alloys** - Titanium alloy sandwich panels are used for the blade covering from the leading edge to the 45 percent chord. Titanium alloy is used because of the slightly elevated temperature environment of 400°F, where aluminum cannot be used, and because of its high strength-to-density ratio.

**René 41** - René 41 is used as a ducting material because of its excellent static strength and creep and rupture properties in the 1200° to 1400°F range.

**Inconel 718** - Inconel 718 is used in the hot gas duct system and for applications where the temperature does not exceed 1200°F. It has excellent short-time strength properties and long-time rupture and creep properties up to 1200°F.



CARPENTER CUSTOM 455

MARAGING STAINLESS STEEL

STATIC STRENGTH VS. TEMPERATURE PROPERTIES

TYPICAL VALUES

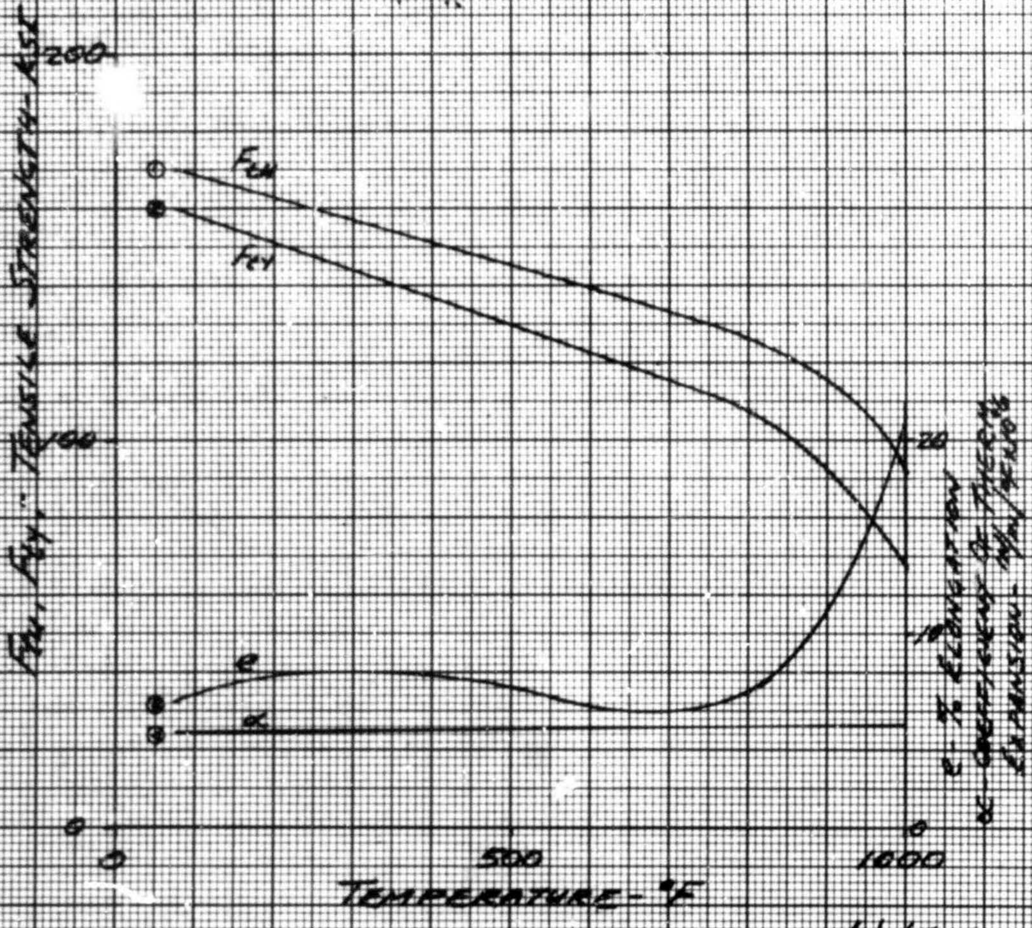
# TITANIUM - TI - 6AL - 6V - 2SN ALLOY

## SHEET PROPERTIES

REF - AIRCRAFT DESIGNERS HANDBOOK FOR
TITANIUM AND TITANIUM ALLOYS BY
BATTELLE MEMORIAL INSTITUTE

① TABLE 5-5.1-1, FIG. 5-5.3.1-1
② TABLE 5-5.1-1, FIG. 5-5.3.1-2
③ FIG. 5-4.3.1-8
④ FIG. 5-5.4-3

MINIMUM GUARANTEED VALUES



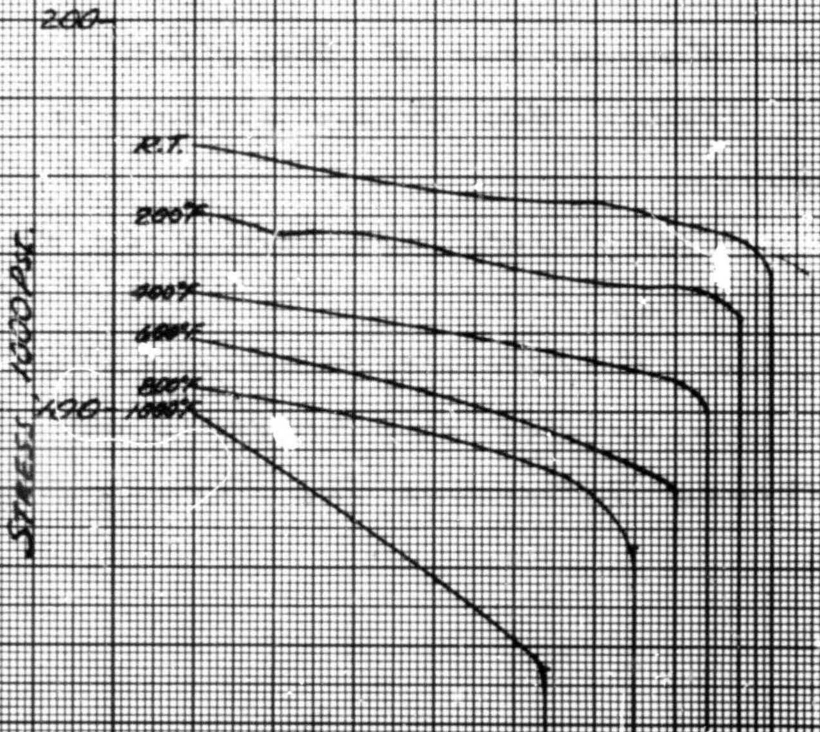
W. Leas
5-27-65

COMPRESSIVE TANGENT-MODULUS CURVES FOR SOLUTION-TREATED AND AGED Ti-6Al-4V ALLOY SHEET AT ROOM & ELEVATED TEMPERATURES

LONGITUDINAL PROPERTIES

REF - AIRCRAFT DESIGNERS HANDBOOK FOR
TITANIUM & TITANIUM ALLOYS BY
BATTELLE MEMORIAL INSTITUTE
FIG. 5-9.3.3-4

MINIMUM GUARANTEED VALUES



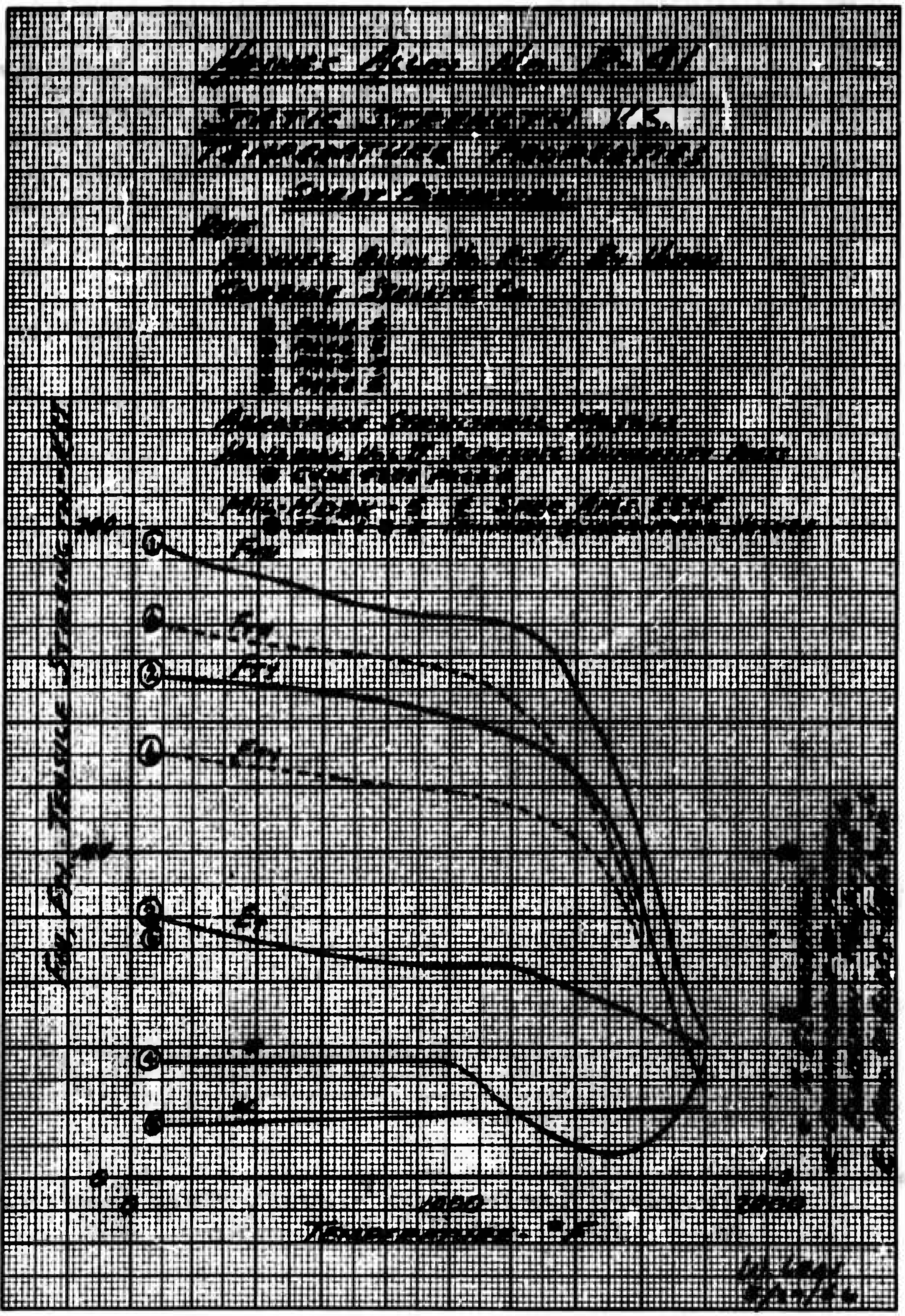
## Safe Endurance Limit Fatigue Allowables

| Material | Ftu PSI | Fcy or Fty PSI | Mean ± Cyclic Stress | | |
|---|---|---|---|---|---|
| | | | Smooth Areas | Moderate Stress Raisers | Severe Stress Raisers As Hol |
| Titanium | 170,000 | 160,000 | 25000 ± 14500 | 25000 ± 7500 | 25000 ± 5000 |
| Maraging Stainless Steel | 257,000 | 245,000 | 50,000 ± 25000 | 50,000 ± 18000 | 50,000 ± 10000 |

In Certain Applications A Trade Off Between Steady And Cyclic Allowable Is Permitted.

Moderate Stress Raisers Are Items Such As Unloaded Holes.

Severe Stress Raisers Are Loaded Holes. All Loaded Holes Are Pre-Stressed Or Shot Peened To Improve The Fatigue Life


Tensile Strength
1200
2000
Temperature - °F

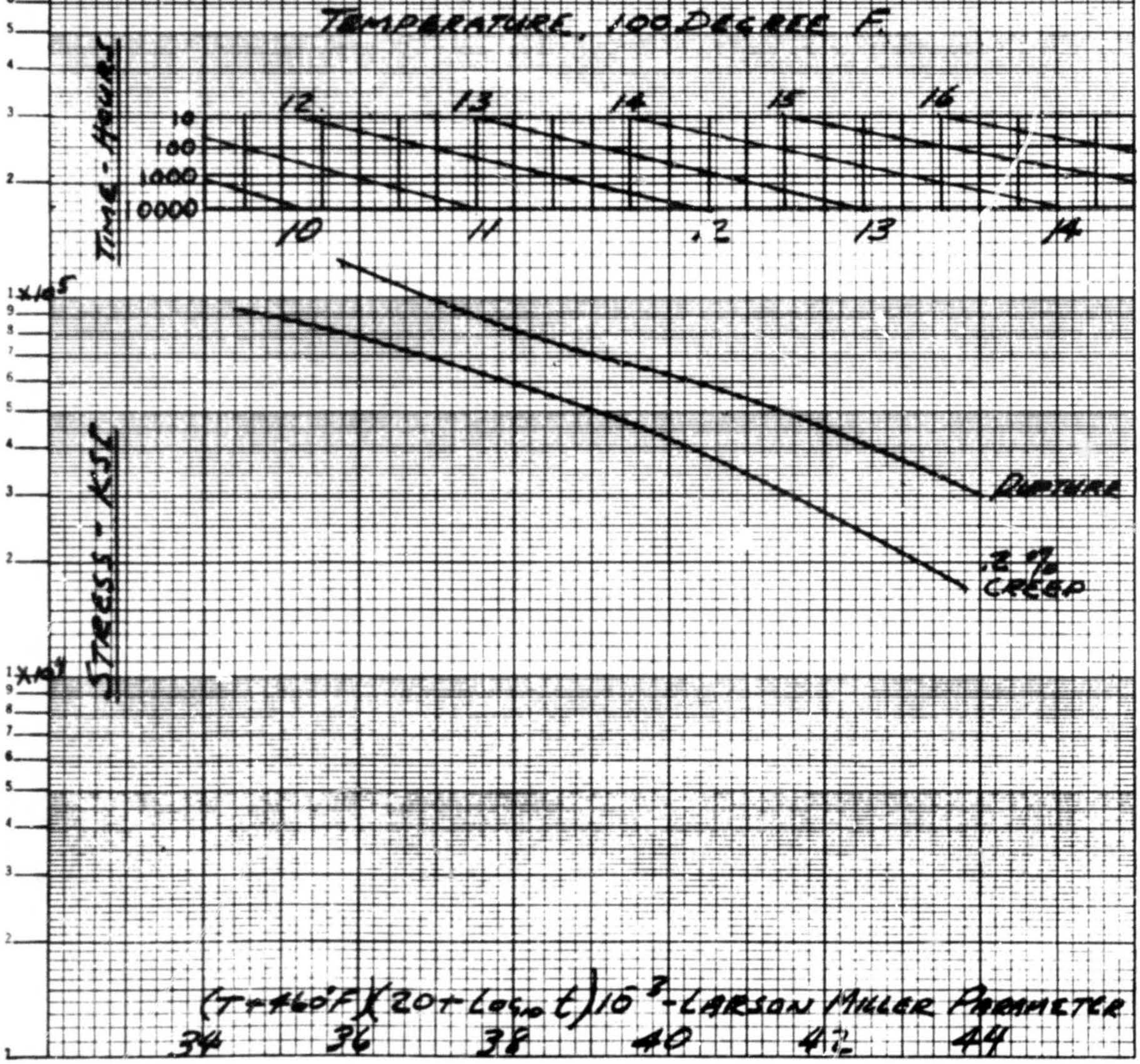
# HAYNES ALLOY NO. R-41

## .2% CREEP AND CREEP RUPTURE STRESS VERSUS LARSON MILLER PARAMETER

REF - HAYNES ALLOY NO. R-41 BY UNION CARBIDE
STELLITE CO.

## SHEET PROPERTIES

### TYPICAL VALUES

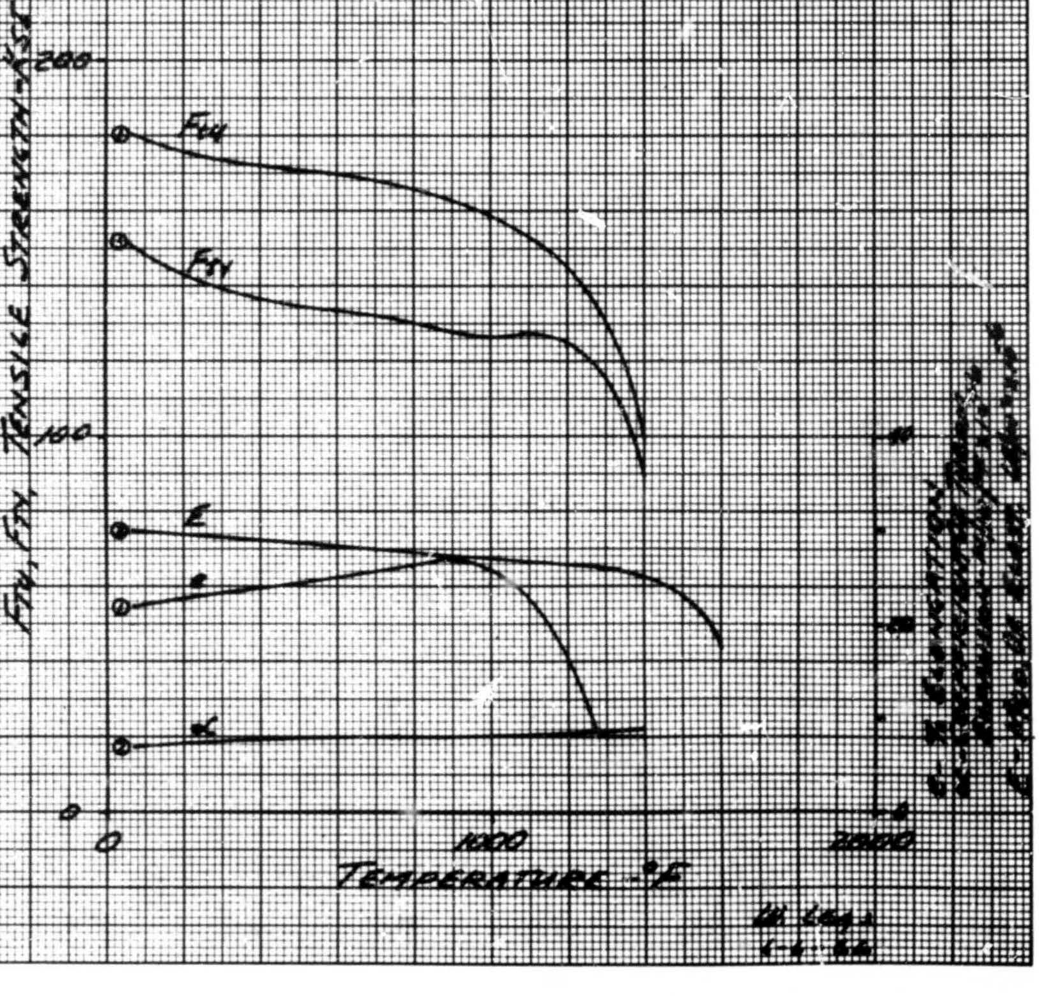


# INCONEL 718

## STATIC STRENGTH VS. TEMPERATURE PROPERTIES

SHEET PROPERTIES
GUARANTEED MINIMUM VALUES

REF.
AEROSPACE STRUCTURAL METALS
HANDBOOK VOL II SYRACUSE
UNIVERSITY PRESS
⊙ FIG. 3.0311
⊗ FIG. 3.014
● FIG. 3.061

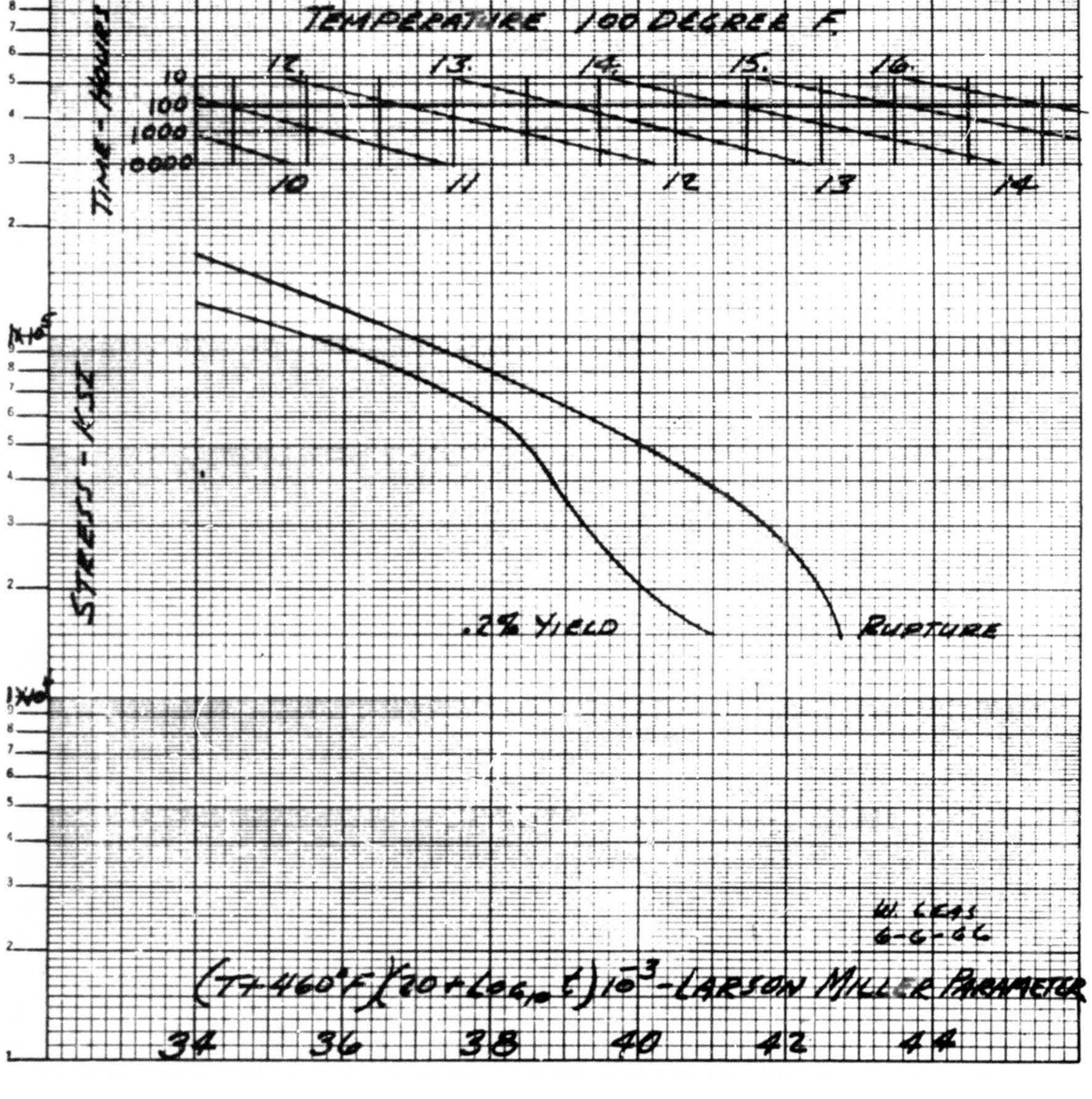


# INCONEL 718

## .2 % CREEP AND CREEP RUPTURE STRESS VERSUS LARSON MILLER PARAMETER

MINIMUM GUARANTEED VALUES

REF - AEROSPACE STRUCTURAL METALS HANDBOOK
VOL. II SYRACUSE UNIVERSITY PRESS, FIG. 3.0431



## Rotor Blade

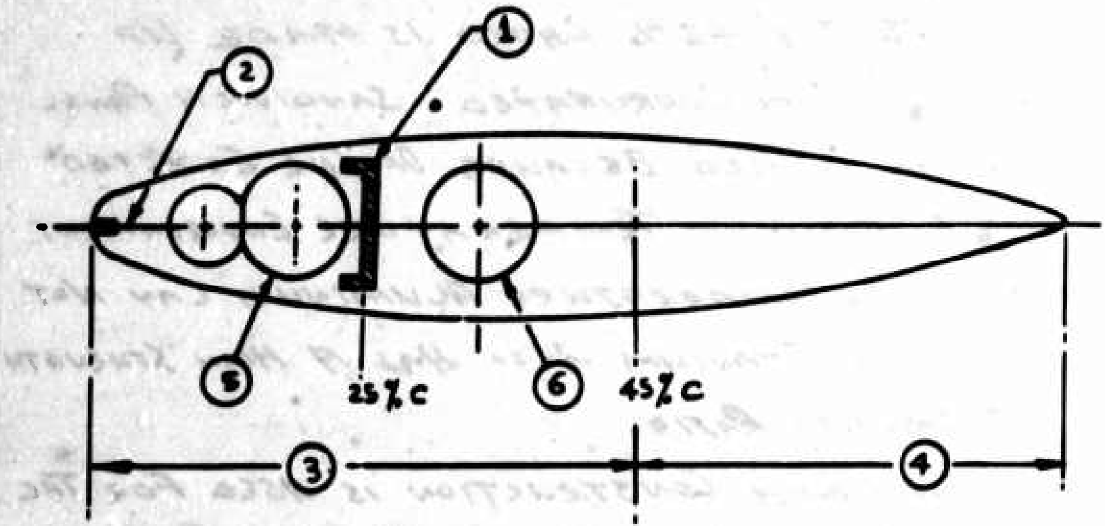
The Blade Skin From The Leading Edge To The 45% Chord Is Made Up Of Titanium Alloy Corrugated Sandwich Panel. Titanium Is Used Because Of The Expected 400°F. Maximum Temperature Environment At Which Temperature Aluminum Can Not Be Used. Titanium Also Has A High Strength To Density Ratio.

Sandwich Construction Is Used For The Skin To Minimize The Number Of Stiffeners Required To Carry Air Load And Blade Torsion.

The Trailing Edge Of The Blade Aft Of 45% Chord Is Made Of Light Gauge Aluminum Alloy As It Is Out Of The Elevated Temperature Environment And Is Lightly Loaded By Air Load.

The Blade Spar Is Made From Carpenter 455 Custom Maraging Steel. It Is Used Primarily For Its Excellent Fatigue And Static Properties. At The 400 Degree F. Environment, It Shows Only A Slight Reduction In Ultimate Strength.

## ROTOR BLADE (REF. DWG 395-0919)



- **LOAD CARRYING ELEMENTS IN ROTOR BLADE:**

① • THE ROTOR BLADE SPAR CARRIES THE ROTOR BLADE CENTRIFUGAL FORCE, THE FORCE FROM THE GAS PRESSURE IN THE DUCTS, VERTICAL SHEAR, ALL THE BLADE FLAPPING MOMENT AND ACTS WITH THE LEADING EDGE BALANCE WEIGHT TO CARRY THE COUPLE FORCE FROM CHORDWISE BENDING. IT PROVIDES ALL THE REQUIRED BLADE STIFFNESS NECESSARY TO LIMIT THE BLADE BENDING DEFLECTIONS DURING GROUND FLAPPING CONDITIONS.

② • THE LEADING EDGE BLADE BALANCE ACTING WITH THE SPAR (ITEM #1) CARRIES THE BLADE CHORDWISE MOMENT. ALSO IT CARRIES THE CHORDWISE SHEAR BETWEEN SEGMENTS.

## ROTOR BLADE (CONTINUED)

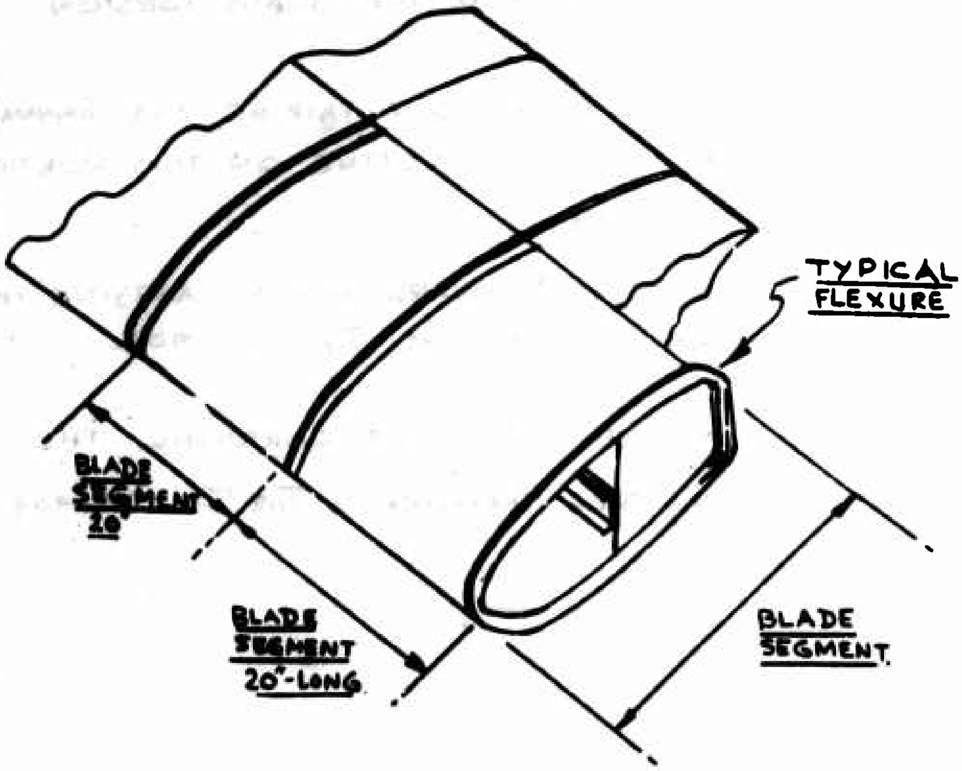
③ • THIS SECTION OF THE BLADE CARRIES THE LOCAL AIRLOAD AND THE BLADE TORSION.

④ • THE TRAILING EDGE IS A FAIRING AND SUPPORTS ONLY THE AIRLOAD ACTING ON THIS PORTION OF THE BLADE.

⑤ • THESE ARE THE FORWARD DUCTS CARRYING HOT GAS UNDER PRESSURE TO THE TIP CASCADE

⑥ • THIS IS THE AFT. DUCT CARRYING THE HOT GAS UNDER PRESSURE TO THE TIP CASCADE.

**BLADE SEGMENT:** (REF. DWG 395-0919)



THE ROTOR BLADE IN THE CONSTANT SECTION IS SEGMENTED INTO 20 INCH. LENGTHS, AND IN THE TRANSITION SECTION THE LENGTH OF SEGMENT VARIES SLIGHTLY FROM 20 INCHES.

THE SEGMENTS EXTEND FROM THE LEADING EDGE TO THE 45% CHORD. THE SEGMENTS ARE SEPARATED BY FLEXURES WHICH TRANSMIT TORSION FROM ONE SEGMENT TO THE NEXT, THUS PROVIDING TORSIONAL CONTINUITY TO THE BLADE. THE FLEXIBILITY OF THE

## BLADE SEGMENT (CONTINUED)

FLEXURES PREVENT THE SEGMENT FROM PICKING UP ANY STRAIN FROM FLAPWISE AND CHORDWISE BENDING AND CENTRIFUGAL FORCE.

AS A RESULT, THE SEGMENT CARRIES ONLY LOCAL AIRLOAD PLUS ALL THE BLADE TORSION AND CHORDWISE SHEAR

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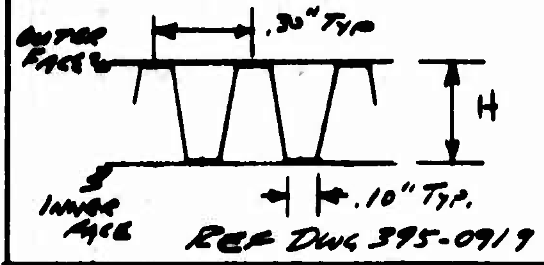
## BLADE SKIN PANELS



REF DWG 395-0919

SANDWICH SKIN - TITANIUM

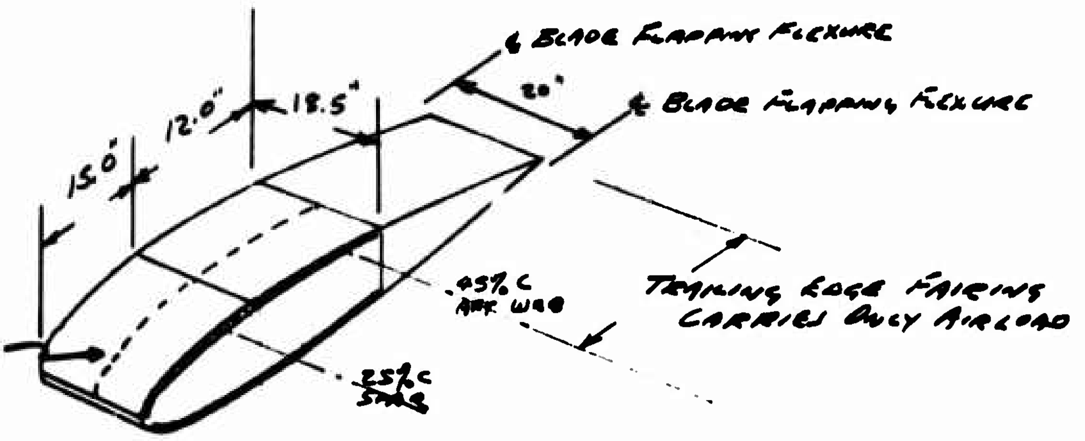
TYPE 1. FROM NOSE TO 25%C

TYPE 2. FROM 25%C TO AFT WEB

| TYPE | H. | CORE GAGE | INNER FACE GAGE | OUTTER FACE GAGE | | | |
|---|---|---|---|---|---|---|---|
| | | | | .2R TO .6R | .6R TO .8R | .8R TO .9R | .9R TO 1.0R |
| 1. | .310 | .007 | .012 (.2R-.6R)<br>.011 (.6R-1.0R) | .011 | .008 | .007 | .006 |
| 2. | .353 | .008 | .011 | .008 | .007 | .006 | .004 |

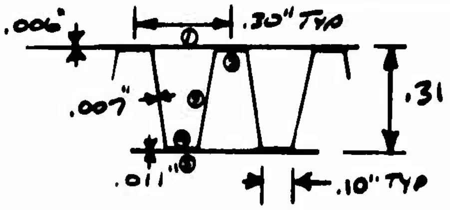
THE AIR LOADING IS HIGHEST ON THE SKIN PANELS AT THE .9R. THIS PANEL ALSO HAS THE LIGHTEST GAGE, THEREFORE A SAMPLE CALCULATION FOR AIRLOAD IS SHOWN FOR THIS STATION.

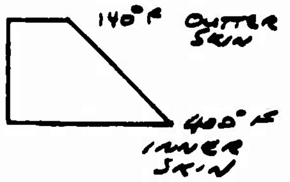
IN BOARD OF THE TIP, THE BLADE TORQUE BUILDS UP. SHEAR STABILITY OF THE SKIN PANEL DESIGNS THE PANEL RATHER THAN AIR LOADING



TYPICAL-30" BLADE SEGMENT

## BLADE SKIN PANELS





TYPE I CONSTRUCT

@ .9R.

MOMENT OF INERT

NOTE - TEMPERATURE GRADIENT BASED ON XV-9A DATA.

## THERMAL STRESSES

| EL | AREA | t - 70° | α × 10⁻⁶ | E × 10⁶ | f = tαE | f × A | f THERMAL STRESS |
|---|---|---|---|---|---|---|---|
| 1 | .3(.006) = .0018 | 70° | 4.9 | 16.0 | 5488 | 9.8784 | +11639 |
| 2 | .1(.007) = .0007 | 70° | 4.9 | 16.0 | 5488 | 3.8416 | +11539 |
| 3 | .297(.007)2 = .0042 | 200° | 4.9 | 15.25 | 14945 | 62.7690 | +2082 |
| 4 | .1(.007) = .0007 | 330° | 4.9 | 14.50 | 23446 | 10.4122 | -6419 |
| 5 | .3(.011) = .0033 | 330° | 4.9 | 14.50 | 23446 | 77.3718 | -6419 |
|  | .0100 |  |  |  |  | 170.2730 |  |

$$\frac{170.2730^{\#}}{.0100} = 17027 \text{ PSI}$$

## MOMENT OF INERTIA

FOR .3" WIDE STRIP

| EL | AREA | $E_c/E_{70°}$ | $\left(\frac{E_c}{E_{70°}}\right)A$ | Y | YA | Y²A | $I_o$ |
|---|---|---|---|---|---|---|---|
| 1. | .0018 | 1.0000 | .0018 | .3070 | .000 553 | .000 170 | — |
| 2. | .0007 | 1.0000 | .0007 | .3035 | .000 212 | .000 064 | — |
| 3. | .0042 | .9521 | .0040 | .1580 | .000 620 | .000 096 | .000 026 |
| 4. | .0007 | .9062 | .0006 | .0065 | .000 004 | — | — |
| 5. | .0033 | .9062 | .0030 | .0030 | .000 007 | — | — |
|  |  |  | .0101 | (.1384) | .001398 | .000330 |  |

$I = .000330 + .000026 - .0101(.1384)^2 =$

$= .000356 - .000193 = .000163 \text{ IN}^4$ FOR .3" WIDE STRIP

9704

## BLADE SKIN PANEL    BENDING STRESS

TYPE I CONSTRUCTION – MATERIAL TITANIUM

PANEL SIZE = 15" x 9.25"    $X = 1.62$    $\beta = .086$

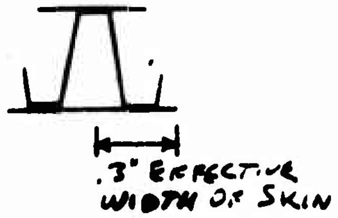
AIR PRESS = $8.2\ \frac{\#}{IN^2}$ LIMIT    FROM LOADS SECTION

$M = \beta p a^2 = .086\left(8.2\frac{\#}{IN^2}\right)(9.25)^2 = 60.3$ "# BENDING / 1" OF PANEL WIDTH

$$f_b = \frac{Mc}{I} = \frac{(60.3\times1.5)\frac{.3\ WIDE}{1.00\ WIDE}(.1384)}{1.63\times10^{-4}} = -23000\ PSI$$

$-6419$ PSI = $f_{TEMP}$

$-29419$ PSI



EFF SKIN WIDTH = $W_e = 1.9t\sqrt{\frac{E}{f_c}}$

$= 1.9(.011)\sqrt{\frac{14.5\times10^6}{29419}} = .465$

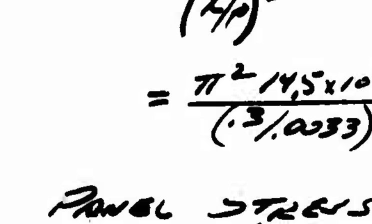
SINCE .465 > CORRUGATION WIDTH

ALL THE COMPRESSION SKIN IS EFFECTIVE = .3"

CHECK PANEL BENDING ACROSS CORRUGATIONS

$M = \beta_{MN}\,p a^2 = .049\left(8.2\frac{\#}{IN^2}\right)(9.25)^2$

$= 34.3$ "# BENDING / 1" OF WIDTH



$f_c = \frac{M}{W \times t} = \frac{34.3"\# \times \frac{1}{2}}{1" \times .011} = 4675$ PSI

6419 PSI = $f_{TEMP}$

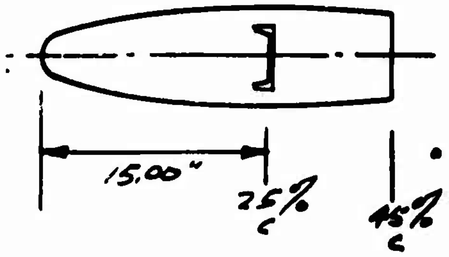
11 094 PSI

$F_c = \frac{\pi^2 E}{(L/\rho)^2}$    $\rho = .3t = .0033$

$= \frac{\pi^2\ 14.5\times10^6}{(.3/.0033)^2} = 17350$ PSI    $M.S. = \frac{17350}{11094} - 1 = +.57$

PANEL STRESSES ARE LOW FOR THE MINIMUM GAGES SHOWN AND MAXIMUM AIRLOADS AT THE BLADE TIP

## BLADE SKIN PANELS - SHEAR LOADS



|  | $A_0$ 45% c |
|---|---|
| .2R To .75R | 246 $IN^2$ |
| .75R To 1.0R | 191 $IN^2$ |

$$Q_T = \frac{M_T}{2A_0}, \quad Q_X = \frac{V_X}{2(15")}$$

### FATIGUE CONDITION

| r/R ROTOR RADIUS | $M_T$ IN# | $Q_T$ #/IN | $V_X$ # | $Q_X$ #/IN | $Q_T + Q_X$ #/IN | $Q_T - Q_X$ #/IN |
|---|---|---|---|---|---|---|
| 0 | ±77,440 | ±157.39 | ±374 | ±12.46 | ±169.85 | ±144.93 |
| .2 | ±75,383 | ±153.22 | ±374 | ±12.46 | ±165.68 | ±140.76 |
| .6 | ±59,890 | ±121.72 | ±187 | ±6.23 | ±127.95 | ±115.49 |
| .8 | ±37,080 | ±97.07 | ±93.5 | ±3.12 | ±100.19 | ±93.95 |
| .9 | ±19,720 | ±51.63 | ±46.75 | ±1.56 | ±53.19 | ±50.07 |
| 1.0 | 0 | — | 0 | — | - | - |

### LIMIT MANEUVER CONDITION

| r/R ROTOR RADIUS | $M_T$ IN# | $Q_T$ #/IN | $V_X$ | $Q_X$ #/IN | $Q_T + Q_X$ #/IN | $Q_T - Q_X$ #/IN |
|---|---|---|---|---|---|---|
| 0 | 453,600 | 922 | 1870 | 62.33 | 984.33 | 859.67 |
| ·2 | 453,600 | 922 | 1870 | 62.33 | 984.33 | 859.67 |
| ·6 | 358,400 | 728 | 935 | 31.16 | 759.16 | 697 |
| ·8 | 220,600 | 577.5 | 467 | 15.56 | 593 | 562 |
| ·9 | 119,800 | 313.61 | 231 | 7.70 | 321.31 | 306. |
| 1.0 | – | – | – | – | – | – |

9704

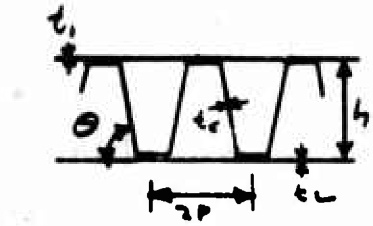
## BLADE SKIN PANELS

CHECK SHEAR STABILITY OF SKIN PANEL IN THE CONSTANT BLADE SECTION

REF.– ELASTIC CONSTANTS FOR CORRUGATE CORE SANDWICH PANELS NACA TN-2289

ANALYSIS & DESIGN OF FLIGHT VEHICLE STRUCTURES BY E.F. BRUHN

HIGHEST SKIN SHEAR OCCURS AT THE INBOARD END OF THE BLADE, USE VALUES AT .2 RADIUS



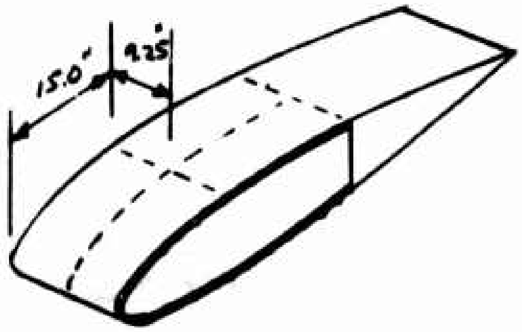
MATERIAL – TITANIUM

$t_1 = .011$ $p = .15$ $\theta = 80°$

$t_2 = .012$ $\mu = .31$ $t_c/t_1 = .637$

$t_c = .007$ $h_c = .288$ $p/h_c = .520$

$E_c = 14.7 \times 10^6$ @ 400°F $S = 7.0$ $h_c/t_c = 41.2$



$$D_{Qy} = Sh\left(\frac{E_c}{1-\mu^2}\right)\left(\frac{t_c}{h_c}\right)^3 = 505$$

$$D = \frac{E_c\, t\,(h+t)^2}{2(1-\mu^2)} = 9175$$

$$J = \frac{D_{Qy}\, b^2}{\pi^2 D} = .464$$

$$a/b = \frac{15}{9.25} = 1.6 \qquad K_s = 2.8$$

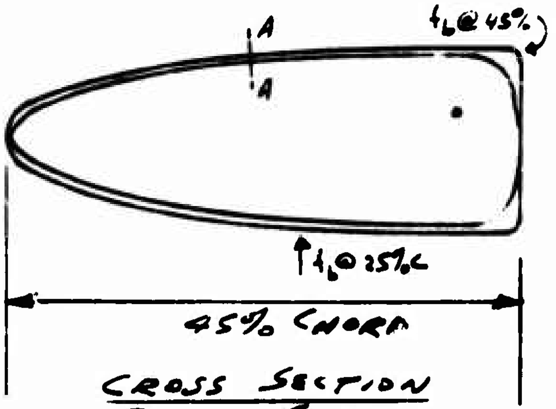
$$N_{xy} = \frac{K_s \pi^2 D}{2(1-\mu^2)}$$ = PANEL SHEAR BUCKLING STRESS

$N_{xy} = 2970$ #/IN

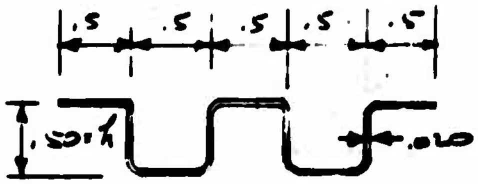
M.S. FOR PANEL BUCKLING $= \frac{2970}{984} - 1 = \underline{\underline{+2.00}}$

## BLADE FLEXURES (BLADE SEGMENT –

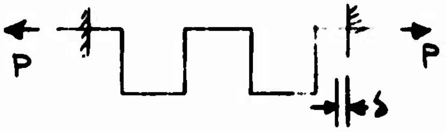
BLADE SEGMENT – MATERIAL – INCONEL 718



CROSS SECTION BLADE FLEXURE
DWG 395-0919



TYPICAL FLEXURE
CROSS SECTION AA



SEC, A.A

$$\delta = \frac{P}{EI}\left(\frac{h^3}{3} + \frac{h^2}{2}\right)$$

$$P = \frac{\delta EI}{\frac{h^3}{3} + \frac{h^2}{2}} \qquad h = .5 \qquad E = 29 \times 10^6$$

$$M = \frac{Ph}{2} \qquad f_b = \frac{Mc}{I} \qquad P = \frac{\delta EI}{.1667}$$

$$f_b = \frac{Ph}{2}\frac{c}{I} = \frac{\delta E\cancel{I}}{.1667}\frac{h}{2}\frac{c}{\cancel{I}} = \underline{3\delta E h c} \qquad c = t/2$$

$$f_b = .75Et\delta \qquad t = .020 \qquad f_b = 435 \times 10^3 \delta \text{ psi}$$

| r/R | δ | | $f_b = 435 \times 10^3 \delta$ | | |
|---|---|---|---|---|---|
| | 25%C | 45%C | 25%C | 45%C | |
| .135 | .0429±.0417 | .0400±.0468 | 18650±18100 | 17400±20400 | FATIGUE COND. |
| .735 | .0848±.0315 | .0347±.0355 | 15150±13700 | 15100±15450 | |
| .135 | .284 | —— | 123,000 | —— | GROUND FLAPP. 2.5g |
| .735 | .0663 | —— | 29,000 | —— | |

ULTIMATE STRENGTH CHECK

GROUND FLAPPING

$$M.S. = \frac{150000 \times 1.5}{123000 \times 1.5} - 1.0 = +.22$$

δ FROM BLADE STRESS CALCULATIONS

8704

## BLADE FLEXURES (BLADE SEGMENT TO BLADE SEGMENT - MATERIAL · INCONEL 718

CHECK SHEAR STRESS IN FLEXURE FROM BLADE TORSION.

$r/R = .735$ $T = 275000''\#$ LIMIT

$$q = \frac{T}{2A} = \frac{275000}{2(244)} = 560 \frac{\#}{IN}$$

$$q_{s_{cr}} = 5.3 E \frac{t^3}{b^2} = 5.3(29\times10^6)\frac{(.020)^3}{(.5)^2} = 4960\ \#/IN$$

FLEXURE WILL NOT BUCKLE FROM TORQUE SHEAR

$r/R = 135$ $T = 458000''\#$

$$q = \frac{T}{2A} = \frac{458000}{2(246)} = 933 \frac{\#}{IN}$$

$933 \frac{\#}{IN} < 4960 \frac{\#}{IN}$ FLEXURE WILL NOT BUCKLE

$$f_s = \frac{933\ \#/in}{.02''} = 46,500\ Psi$$

$F_{su} = 100,000$ Psi $\qquad M.S. = \frac{100,000}{46500 \times 1\frac{1}{2}} - 1 = +.43$

FATIGUE SHEAR $= \pm 75500''\#$ $\qquad f_s = \pm 7650$ PSI

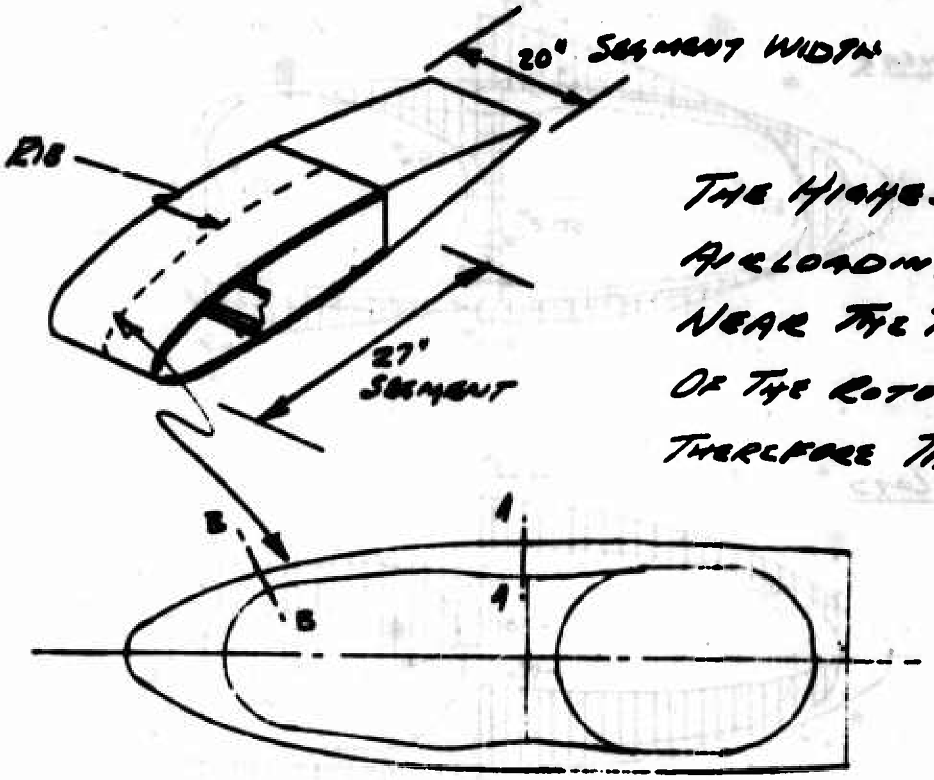
PRINCIPAL STRESS $= \frac{S_N}{2} \pm \sqrt{\frac{S_N}{2}^2 + S_S^2}$ $\qquad S_N = 20400$ Psi, $S_S = 7650$

$= 22800$ PSI

FATIGUE CHECK. $\qquad M.S. = \frac{25000}{22800} - 1 = +.10$

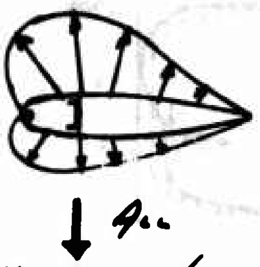
## BLADE RIB ANALYSIS

REF. DWG 395-0919



THE HIGHEST AIRLOADING IS NEAR THE TIP OF THE ROTOR BLADE, THEREFORE THE PRESSURE DISTRIBUTION AT .9 RADIUS IS USED TO ANALYSE THE RIB. (SEE LOADS SECTION).

### RIB LOADS & REACTIONS



VERTICAL LOAD IS REACTED AT THE SPAR

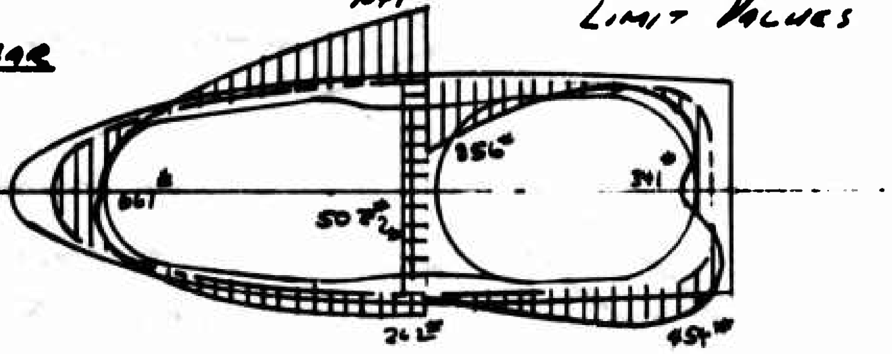
RIB SHEAR DISTRIBUTION FROM CHORDWISE COMPONENT OF AIR LOAD

RIB SHEAR DISTRIBUTION FROM AIR LOAD TORQUE

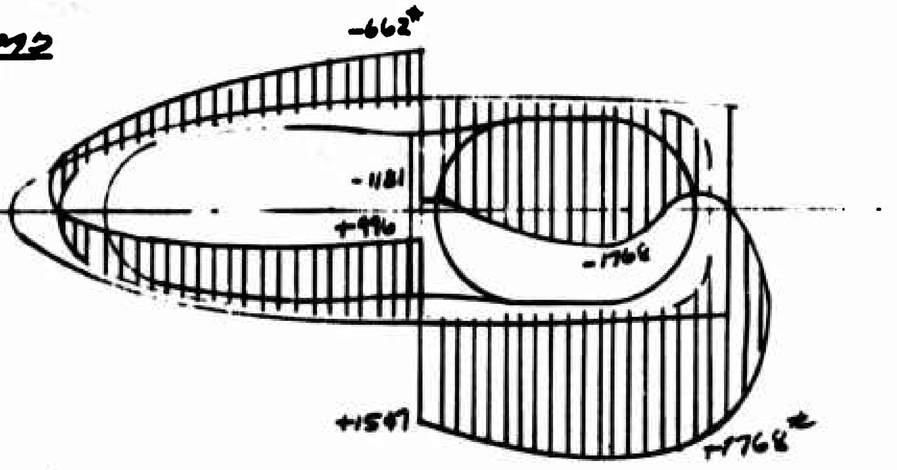
## BLADE RIB ANALYSIS

SHEAR, AND MOMENT CURVES BASED ON
AIR LOAD DISTRIBUTION AT .9 RADIUS
LIMIT VALUES

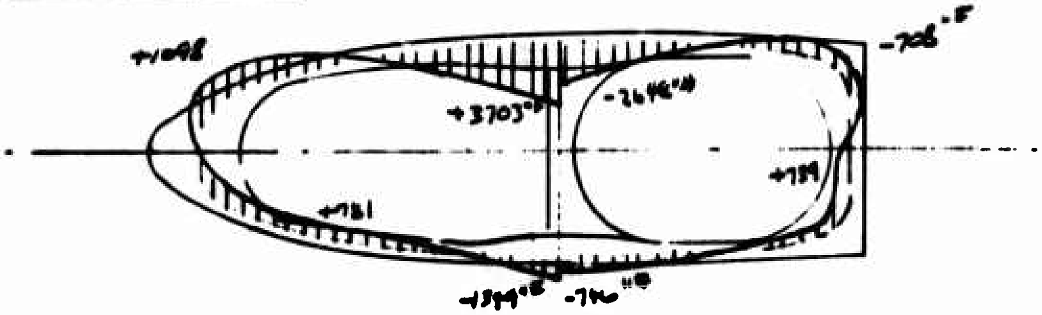
RIB SHEAR



RIB
AXIAL LOAD

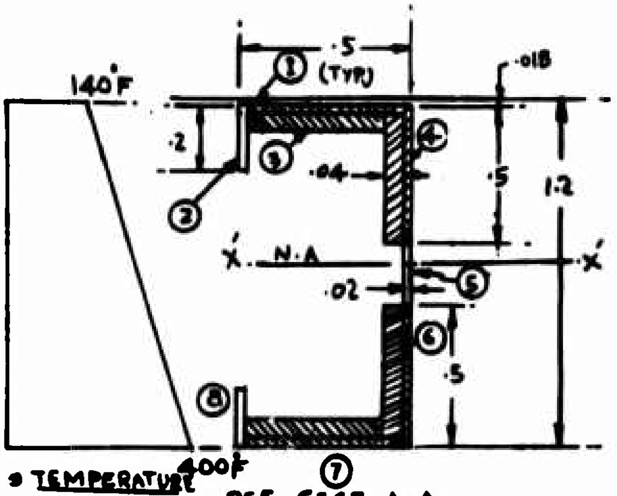


RIB
BENDING MOMENT



# VI STRESS ANALYSIS

## • BLADE RIB ANALYSIS



• TEMPERATURE GRADIENT

REF. SECT. A-A

$I_{x'x'} = 0.030696\ IN^4$ (N.A)

$\bar{y} = \frac{Ay}{A} = \frac{.07983}{.13684} = .5834\ in.$

$A = 0.13684\ in^2$

$B.M = 5,237$ lb-in

AXIAL LOAD(P) = 679 lbs

## SECT. B-B TEMPERATURE STRESSES

| ELEMEN | AREA | $\Delta t = T-70°$ | $\alpha \times 10^{-6}$ | $E \times 10^6$ | $f = \Delta t\,\alpha E$ | $\Delta P$ | $f_{RESULT.}$ |
|---|---|---|---|---|---|---|---|
| 1 | .0058 | 80° | 4.6 | 15.4 | - 5,670 | -32.88 | +37,248 * |
| 2 | .004 | 96 | 7.2 | 30.0 | - 20,700 | -82.80 | +22,210 |
| 3 | .02952 | 90° | 7.2 | 30.0 | - 19,400 | -573 | +23,518 |
| 4 | .0300 | 125 | 7.2 | 30.0 | - 27,000 | -810 | +15,918 |
| 5 | .004 | 200 | 7.9 | 29.0 | - 23,080 | -100.3 | +17,838 |
| 6 | .0300 | 275 | 7.9 | 29.0 | - 63,000 | -1,890 | -20,082 |
| 7 | .0295 | 323 | 7.9 | 28.0 | - 71,500 | -2,109 | -28,582 |
| 8 | .004 | 310 | 7.9 | 28.0 | - 68,500 | -274 | -25,918 |

= .1368

$\frac{5,872}{.1368} = (42,918)$

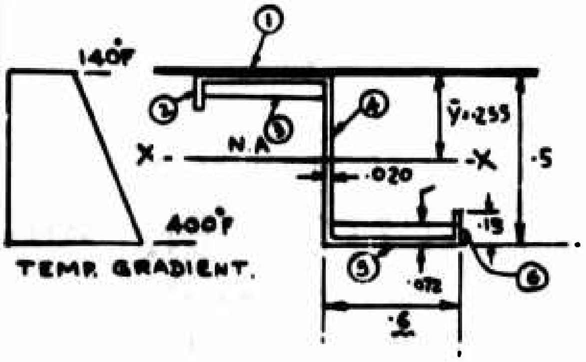
$$f_b + f_a = \left[\frac{Mc}{I} + \frac{P}{A}\right] = \left[\frac{5,237(.5834)}{.03069} + \frac{679}{.1368}\right] \times 1.5 = \underline{155,917\ psi}$$

RESULTANT STRESS (AT ITEM #1) = $(f_b + f_a) - f_t = (118,669)$ PSI *

RESULT. STRESS (AT ITEM #7) = $(f_b + f_t) - f_a = \underline{122,601}$ psi

$$M.S. = \frac{176000\ PSI}{155917\ PSI} - 1 = +.13$$

## • BLADE RIB ANALYSIS



$I_{xx}$ (N A) = .003983 in.$^4$

$\bar{Y} = \dfrac{.02234}{.0880} = .2538$ in.

A = .0880 in.$^2$

B.M = 1,150 (#-IN.)

AXIAL LOAD = 550 #

### SECT. C-C TEMPERATURE STRESSES

| ELEMENT | AREA - IN.$^2$ | Δt = T-70° | $\alpha \times 10^{-6}$ | $E \times 10^6$ | $f = \Delta T \times \alpha E$ | ΔP | $f_t$ = Σ RESULT. |
|---|---|---|---|---|---|---|---|
| 1 | .0048 | 80° | 4.6 | 15.4 | -5,660 | - 27.17 | + 46,494 |
| 2 | .0030 | 110 | 7.2 | 30.0 | -23,800 | - 71.4 | + 29,354 |
| 3 | .0336 | 90° | 7.2 | 30.0 | -19,450 | - 653.5 | + 32,704 |
| 4 | .0100 | 270° | 7.9 | 29.0 | 61,800 | - 618.0 | - 9,646 |
| 5 | .0336 | 400 | 7.9 | 28.0 | 88,500 | - 2973.6 | -36,346 |
| 6 | .0030 | 370 | 7.9 | 28.0 | 82,500 | -246.0 | - 29,846 |
| | .088 | | | | | | |

$\dfrac{4,589.6}{.088} = 52,154$

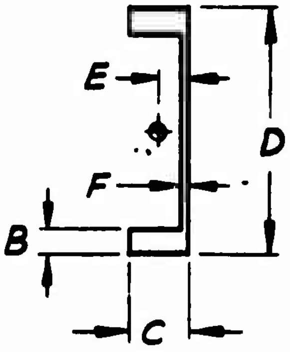
$$f_b + f_a = \left[\frac{Mc}{I} + \frac{P}{A}\right] = \left[\frac{1150(.2538)}{.003983} + \frac{550}{.088}\right] \times 1.5 = \underline{119,625} \text{ PSI (COMP.)}$$

RESUL. STRESS @ ITEM #1. = $(f_b + f_a) + f_t = (119,625) + 46,494 = \underline{166,119}$ PSI

RESUL. STRESS @ ITEM #5 = $(f_b + f_t) - f_a = \underline{103,590}$ PSI

$$M.S. = \frac{173,000 \text{ PSI}}{166,119 \text{ PSI}} - 1 = +.04$$

## BLADE SPAR DIMENSIONS AND SECTION PROPERTIES



| h/R | B | C | D | E | F | A | I | Z |
|---|---|---|---|---|---|---|---|---|
| .2 | .67 | 1.44 | 6.60 | .53 | .15 | 2.71 | 18.9 | 5.74 |
| .35 | .62 | 1.39 | 6.27 | .53 | .125 | 2.37 | 15.1 | 4.82 |
| .5 | .55 | 1.34 | 5.96 | .52 | .10 | 1.97 | 11.77 | 3.95 |
| .65 | .41 | 1.29 | 5.64 | .47 | .10 | 1.52 | 8.05 | 2.85 |
| .8 | .23 | 1.24 | 5.50 | .35 | .10 | 1.08 | 5.1 | 1.85 |
| .95 | .10 | .50 | 5.00 | .09 | .10 | .58 | 1.53 | .612 |

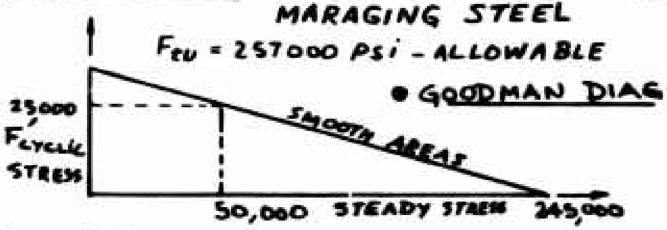
## BLADE SPAR LOADS AND STRESSES

| $r/R$ | FATIGUE LOADS | | | FATIGUE STRESSES (ALSO REFER TO NEXT PAGE) ▼ | | | | ULTIMATE MANEUV. LOADS | | | ULTIMATE MANEUVER STRESSES | | | | GROUND FLAPPING | | * |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | C.F. | $P_{Mc}$ | $\frac{M_f}{I}$ | $\frac{C.F.}{A}$ | $\frac{P_{Mc}}{A}$ | $f_b$ | $\Sigma$ | C.F. | $P_{Mc}$ | $\frac{M_f}{I}$ | $\frac{C.F.}{A}$ | $\frac{P_{Mc}}{A}$ | $f_b$ | $\Sigma$ | $M_F$ ULT. | $f_b$ | $L_{CR}$ |
| .2 | 163,000 | ± 7000 | 1960 ± 2350 | 60,000 | ± 2580 | 6460 ± 7750 | 68,238 ± 10,330 | 180,000 | ± 53,500 | 8550 ± 8550 | 149,000 | ± 19,700 | 28,200 ± 28,200 | 171,217 ± 47,900 | 660,000 | 115,000 | 20.1 |
| .35 | 140,000 | ± 4750 | 1080 ± 1300 | 59,000 | ± 2000 | 3390 ± 4070 | 64,424 ± 6070 | | | | | | | | 413,000 | 85,500 | 22.5 |
| .5 | 115,000 | ± 2770 | 2920 ± 3500 | 58,400 | ± 1400 | 8700 ± 10,400 | 69,547 ± 11,800 | | | | | | | | 240,000 | 60,800 | 26.0 |
| .65 | 85,500 | ± 1380 | 5350 ± 6430 | 56,300 | ± 910 | 15,200 ± 18,100 | 74,670 ± 19,100 | | | | | | | | 185,000 | 65,000 | 23.9 |
| .8 | 52,500 | ± 441 | 6600 ± 7900 | 48,600 | ± 413 | 18,200 ± 21,800 | 79,130 ± 22,210 | 121,000 | ± 4020 | 18,600 ± 18,600 | 112,000 | ± 3730 | 51,200 ± 51,200 | 168,855 ± 54,900 | 141,800 | 22,600 | 38.8 |
| .95 | 16,600 | 0 | 6060 ± 7280 | 28,700 | 0 | 15,200 ± 18,200 | 50,107 ± 18,200 | | | | | | | | 7000 | 11,400 | 22.1 |

- SPAR MATERIAL - CARPENTER CUSTOM 455 MARAGING STEEL

$F_{tu}$ = 257000 psi - ALLOWABLE

- GOODMAN DIAG.



* ALLOWABLE UNSUPPORTED LENGTH OF COMPRESSION FLANGE. L ≥ 20 REQUIRED.

### SPAR STRESSES DUE TO DUCT GAS PRESSURE:

| STA. ($r/R$) | DUCT CROSS AREA - $in^2$ | ULTIM. MANEU. @ 59.40 PSI (*) | FATIGUE @ 35 PSI ▼ |
|---|---|---|---|
| ·20 | 137.65 | 3,017 | 1,778 |
| ·35 | 137.65 | 3,449 | 2,034 |
| ·50 | 137.65 | 4,150 | 2,447 |
| ·65 | 137.65 | 5,379 | 3,170 |
| ·80 | 102.84 | 5655 | 3330 |
| ·95 | 102.84 | 10,531 | 6,207 |

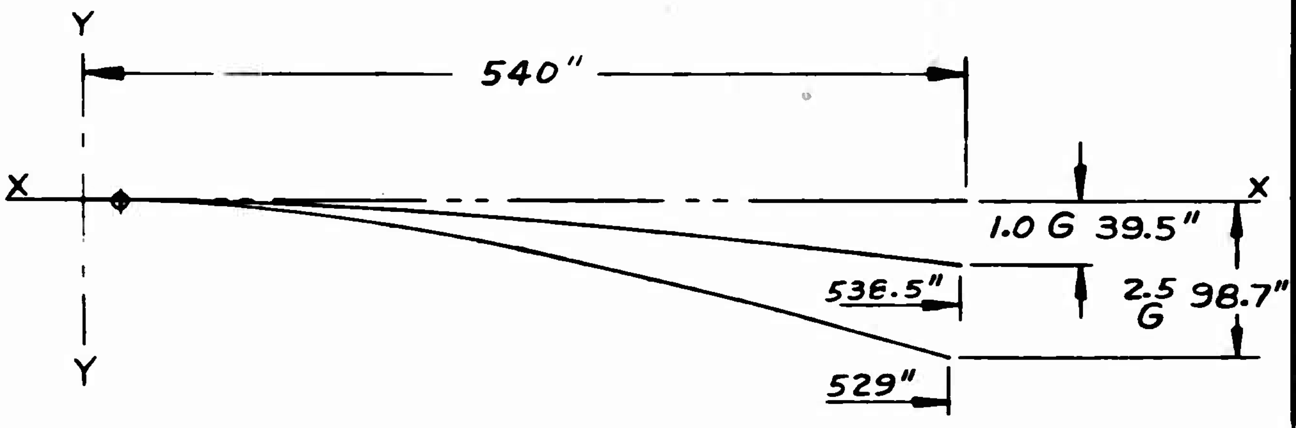
- ULTIMATE MANEUVER - GAS PRESS. = $1\frac{1}{2} \times 39.6 = 59.40$ PSI
- FATIGUE - GAS PRESS. = 35.0 PSI

## BLADE SPAR LOADS AND STRESSES

THE FOLLOWING CHART SHOWS THE ANALYSIS FOR THE FATIGUE CONDITION USING THE LOADS FROM THE COUPLED ANALYSIS. THE LOADS BETWEEN BLADE STA. 60 TO STA. 375 ARE SOME WHAT HIGHER THAN THE DESIGN VALUES, HOWEVER, THE BLADE AS DESIGNED HAS SUFFICIENT STRENGTH TO ACCOMODATE THESE HIGHER LOADS. AS SHOWN IN THE FOLLOWING CHART.

| BLADE STA. IN. | I IN⁴ | Z IN³ | A IN² | C.F. LB. | FLAPWISE MOMENT IN. LB | CHORDWISE MOMENT IN. LB. | FATIGUE STRESS | M. S. |
|---|---|---|---|---|---|---|---|---|
| 105 | 18.9 | 5.74 | 2.71 | 162 000 | 48000<br>±57500 | ±100,000 | 68140<br>±12540 | +.80 |
| 190 | 15.1 | 4.82 | 2.37 | 143 000 | 40750<br>±49,000 | ±55,000 | 68870<br>±12580 | +.75 |
| 330 | 8.0 | 3.16 | 1.63 | 95000 | 52000<br>±62,500 | ±23,000 | 74700<br>±20810 | +.03 |

NOTE - CHORDWISE COUPLE ARM IS 14"



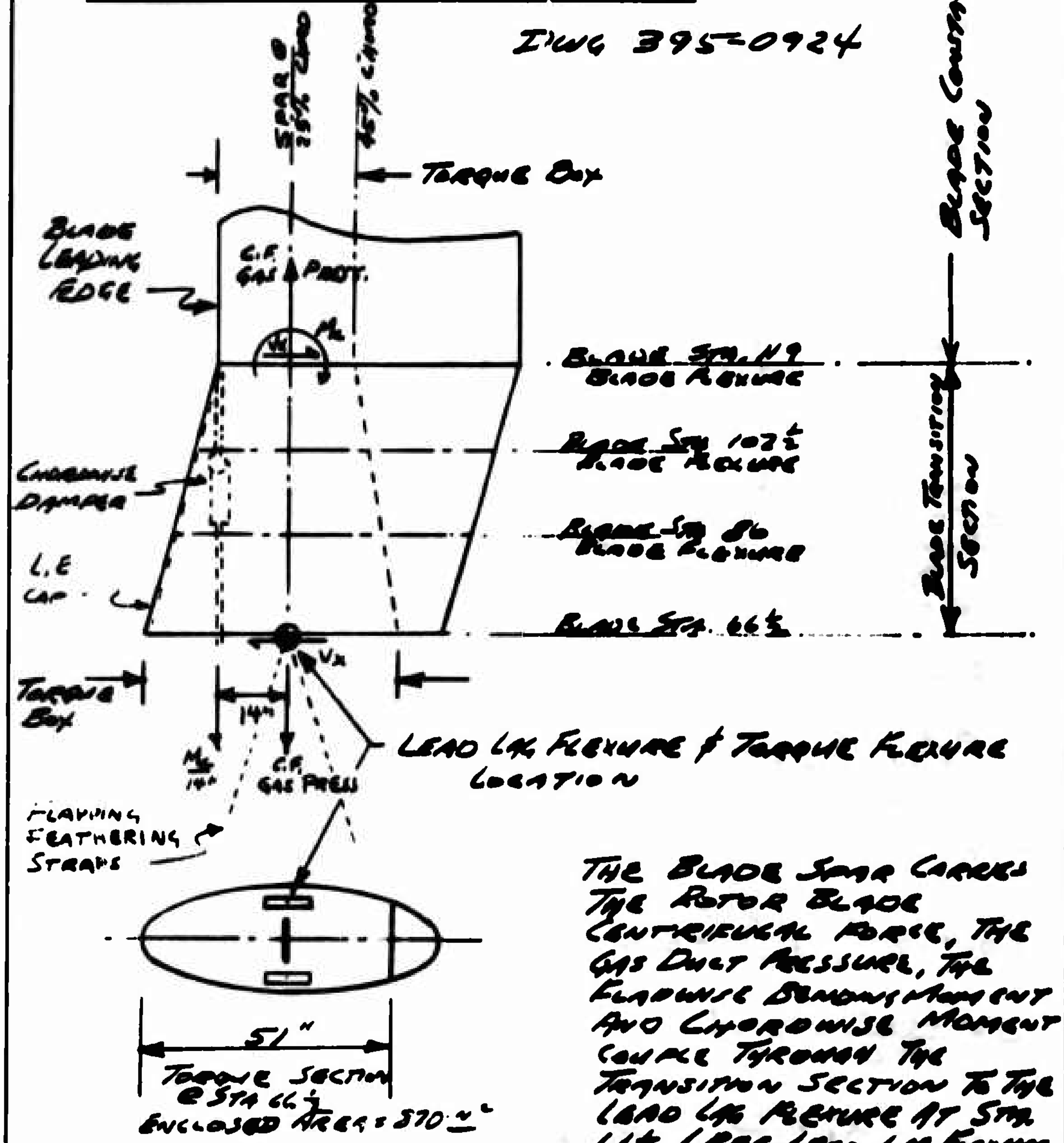
MAIN ROTOR BLADE

GROUND FLAPPING DEFLECTIONS

| | | | | | | | |
|---|---|---|---|---|---|---|---|
| 1.0 G | X | 108 | 189 | 269.8 | 350.5 | 431 | 538.5 |
| | Y | 2.1 | 6.3 | 12.2 | 19.6 | 27.9 | 39.5 |
| 2.5 G | X | 108 | 188.4 | 266 | 346.7 | 425 | 529 |
| | Y | 5.3 | 15.7 | 30.5 | 49 | 69.7 | 98.7 |

## BLADE TRANSITION SECTION

DWG 395-0924



THE BLADE SPAR CARRIES THE ROTOR BLADE CENTRIFUGAL FORCE, THE GAS DUCT PRESSURE, THE FLAPWISE BENDING MOMENT AND CHORDWISE MOMENT COUPLE THROUGH THE TRANSITION SECTION TO THE LEAD LAG FLEXURE AT STA. 66½ (REF. LEAD LAG FLEXURE ANALYSIS)

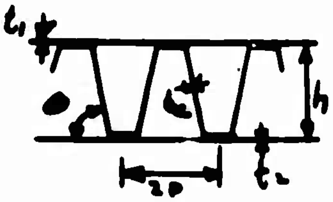
THE CHORDWISE BLADE SHEAR AND TORSION IS CARRIED BY THE BLADE TORQUE BOX AND REACTED BY THE TORQUE FLEXURES AT STA. 66½ (REF. TORQUE FLEXURE ANALYSIS)

## BLADE TRANSITION SECTION

ANALYSIS OF SKIN SHEAR FLOW LOADS

| | | FATIGUE | LIMIT MANEUVER |
|---|---|---|---|
| $V_x$ | REF LOADS SECTION | ±370# | ±1850# |
| $M_{T.R}$ | | 63000 ± 75,500 ''# | 228000 ± 228,000 ''# |
| $V_x/2\times14'' = \frac{V_x}{28}$ | | $\pm 13\ \frac{\#}{IN}$ | $\pm 66\ \frac{\#}{IN}$ |
| $M_{T.R}/2A = \frac{M_{T.R}}{1740}$ | | $35 \pm 43\ \frac{\#}{IN}$ | $131 \pm 131\ \frac{\#}{IN}$ |
| TOTAL SHEAR FLOW | | $35 \pm 56\ \frac{\#}{IN}$ | $\pm 328\ \frac{\#}{IN}$ |

CHECK SHEAR BUCKLING STABILITY OF PANEL-
REF - ELASTIC CONSTANTS FOR CORRUGATED CORE SANDWICH PANELS NACA TN-2289
ANALYSIS & DESIGN OF FLIGHT VEHICLE STRUCTURES BY E.F. BRUHN



MATERIAL TITANIUM

$t_1 = .011$
$t_2 = .012$
$t_c = .007$
$E_c = 14.7\times10^6$ @400°F
$\mu = .15$
$h = .31$
$h_c = .288$
$S = 7.0$
$\theta = 80°$
$t_c/t_1 = .637$
$p/h_c = .520$
$h_c/t_2 = 41.2$
$a = 15$
$b = 19.5$
$a/b = 1.54$

$$D_{Qy} = Sh\left(\frac{E_c}{1-\mu^2}\right)\left(\frac{t_c}{h_c}\right)^3 = 505. \qquad D = \frac{E_c t (h+t)^2}{2(1-\mu^2)} = 9175$$

$$J = \frac{D_{Qy} b^2}{\pi^2 D} = 2.06 \qquad K_s = 3.25$$

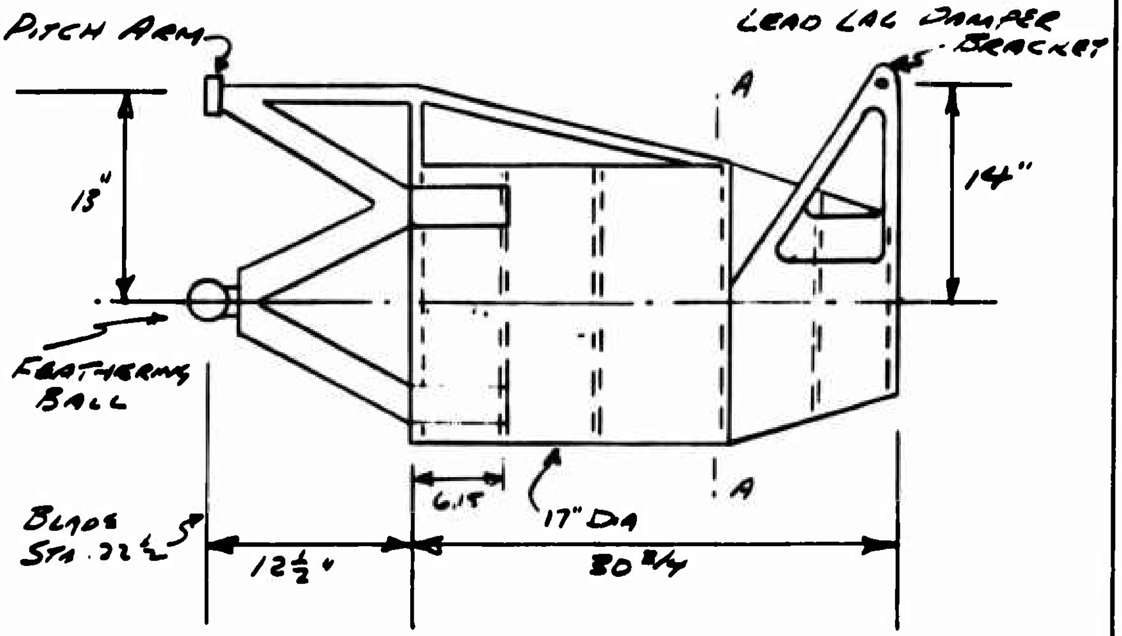
$$N_{xy} = \frac{K_s \pi^2 D}{2(1-\mu^2)} = \text{PANEL SHEAR BUCKLING STRESS}$$

$$N_{xy} = 777\ \frac{\#}{IN}$$

M.S. FOR PANEL BUCKLING $= \frac{777}{328} - 1 = \underline{\underline{1.37}}$

## BLADE TORQUE TUBE

(BETWEEN BLADE FEATHERING BALL AND LEAD LAG HINGE) REF DWG 395-0936



| LOADS | FATIGUE | LIMIT MANEUVER |
|---|---|---|
| $M_C$ | ± 100,000"# | ± 500,000"# |
| $M_F$ | 36,011 ± 43,213"# | 110,632 ± 110,632"# |
| $M_T$ | 63000 ± 75500 | 228000 ± 228000 |

CHECK SEC. AA. FOR STABILITY UNDER BENDING AND TORSIONAL SHEAR

$R = 8\frac{1}{2}''$ $I = \pi R^3 t = 272 \text{ IN}^4$ $A = \pi R^2 = 234$

$t = .14$ MATERIAL - TITANIUM

## BLADE TORQUE TUBE

SEC. AA CONTINUED

REF- ANALYSIS & DESIGN OF FLIGHT VEHICLE STRUCTURES BY BRUHN

$\ell = 6.15 \quad t = .11 \quad r = 8.5$

CHECK BUCKLING SHEAR ALLOWABLE

$Z_L = \frac{L^2}{\ell t}\sqrt{1-u^2} = 38.6 \qquad K_t = 12. \qquad$ FIG. C8.18

$\frac{K_t \pi^2 E}{12(1-u^2)F_{0.7}}\left(\frac{t}{L}\right)^2 = .437 \qquad F_{s_{cr}} = .44 \quad ) = 56000$ Psi

FIG. C8.19

$F_{0.7} = 127000$ Psi PAGE B1.11

BUCKLING STRESS OF CYLINDER UNDER COMPRESSION

$r/t = 77 \qquad Z = 38.6 \qquad K_c = 8.5 \qquad$ FIG. C8.7

$\frac{K_c \pi^2 E}{12(1-u^2)}\left[\frac{t}{r}\right]^2 \frac{1}{F_{0.7}} = .31 \qquad \frac{F_{cr}}{F_{0.7}} = .31 \qquad F_{cr} = 39000$ Psi

REF. FIG C5.8

FOR LIMIT MANEUVER CASE

$f_s = \frac{M_T}{2A} = \frac{2(278000)}{2\pi^2(8.5)^2(.11)} = 9200 \text{ Psi} \qquad \frac{9200 \times 1\frac{1}{2}}{56000} = .25$

$f_b = \frac{Mc}{I} = \frac{500000(8.5)}{214} = 20000 \text{ Psi} \qquad \frac{20000 \times 1\frac{1}{2}}{39000} = .77$

$$\left(\frac{f_b}{F_b}\right)^2 + \left(\frac{f_s}{F_s}\right)^2 = (.77)^2 + (.25)^2 = .65$$

$$M.S. = \frac{1}{\sqrt{.65}} - 1 = +.23$$

## BLADE TORQUE TUBE

### FATIGUE STRESSES

$$f_s = \frac{M_T}{2A} = \frac{63000 \pm 73500}{2(284)} = 134 \pm 161 \text{ \#/in.}$$

$$= 1220 \pm 1460 \text{ Psi}$$

$$f_b = \frac{Mc}{I} = \frac{\pm 100000(8.5)}{214} = \pm 3960 \text{ Psi}$$

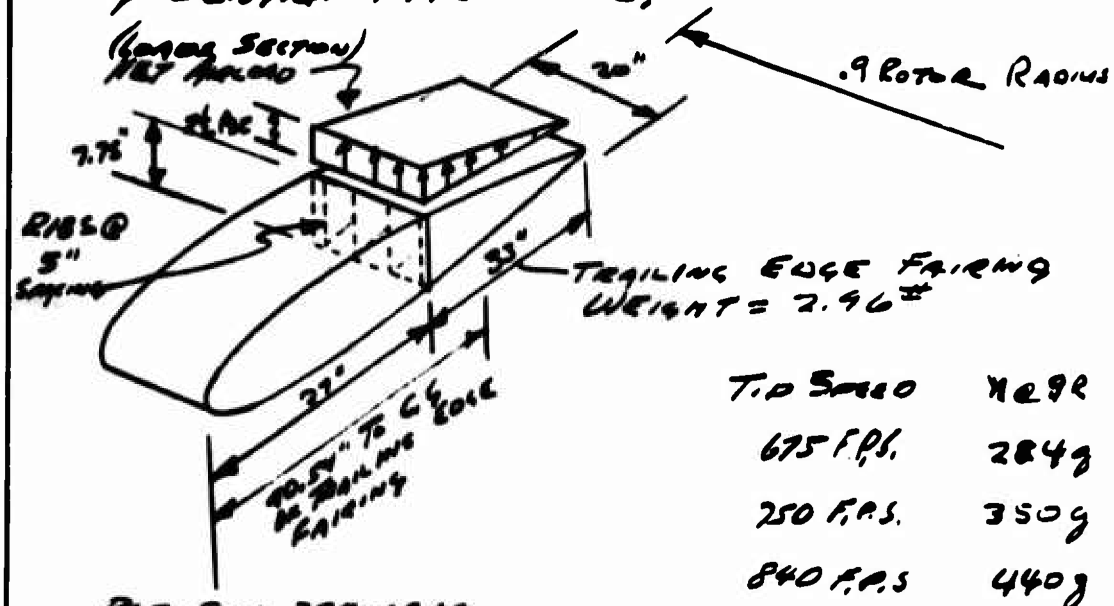
$$\text{MAX. STRESS} = \frac{S}{2} \pm \sqrt{\left(\frac{S}{2}\right)^2 + S_s^2} = \pm 4430 \text{ Psi}$$

$$M.S. = \frac{\pm 14500}{\pm 4430} - 1 = \text{LARGE}$$

# VI: STRESS ANALYSIS

## TRAILING EDGE FAIRING

THE TRAILING EDGE FAIRING AT THE .9 ROTOR RADIUS IS CHECKED FOR AIRLOADING & CENTRIFUGAL FORCE.



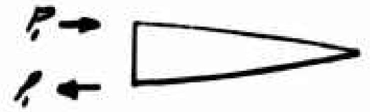
| TIP SPEED | N @ 9R |
|---|---|
| 675 F.P.S. | 284g |
| 750 F.P.S. | 350g |
| 840 F.P.S. | 440g |

REF DWG 395-0919

$M_{\text{OVR AIRLOAD}} = 2\tfrac{1}{2}\text{ PSF} \times \frac{33"}{2} \times \frac{33"}{3} = 453\ "\#/\text{IN}$

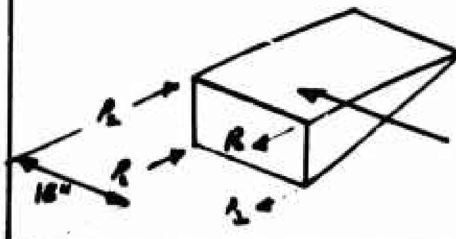
$\text{SHEAR} = 2\tfrac{1}{2}\text{ PSF} \times \frac{33"}{2} = 41.25\ \#/\text{IN}$

CHECK OUT BOARD RIB OF FAIRING



$M = 453\ "\#/\text{IN} \times \frac{5"}{2} = 1135\ "\#$

$P = \frac{1135}{2.75} = 147\#$



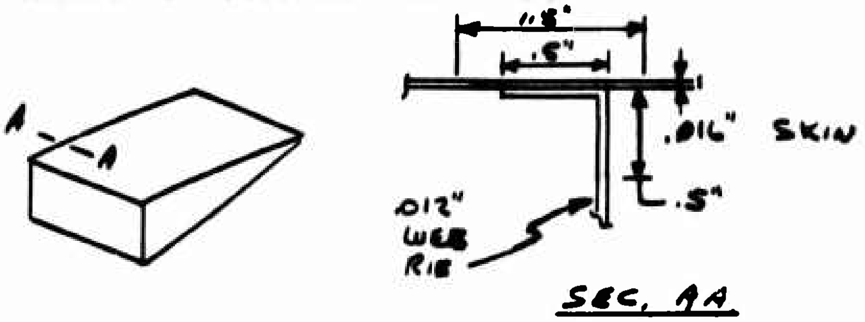
$440g \times 2.96\# = 1300$

$M = 1300\# (40.54 - 27) = 17600\ "\#$

$P_1 = \frac{17600}{2 \times 18"} = 490\#$

8704

## TRAILING EDGE FAIRING



SEC, AA

MATERIAL 2024 AL.

AREA = .036 IN$^2$

$f_{cu} = \frac{P_1+P_2}{A} = \left(\frac{147+490}{.036}\right)1.5 = 26600$ PSI

$F_{cc} = 27000$ PSI

$M.S. = \frac{27000}{26600} - 1 = 0$

SHEAR IN OUTSIDE SKIN DUE TO C.F. FORCE

$q = \frac{2.96\#\times 350g}{2\times 20''} = 26\ \#/IN = 1625$ PSI

$F_{s_{cr}} = 5.3\,E\,\frac{t^2}{b^2}$ $\qquad b^2 = \frac{5.3\,E\,(.016)^2}{1625} = 8.35$

$b = 2.9$

FOR NO BUCKLING OR SKIN AT HOVER.

THE TRAILING EDGE FAIRING DESIGN IS QUITE SIMILAR TO THAT USED ON XV-9A RESEARCH HELICOPTER.

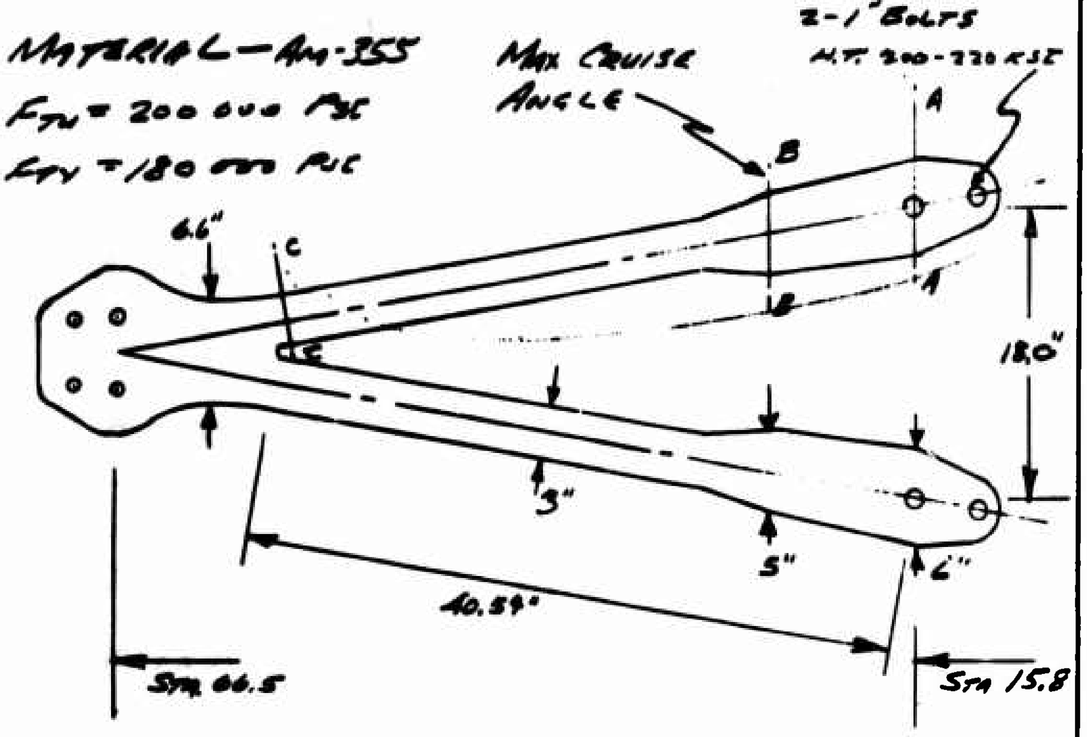
## FLAPPING - FEATHERING BLADE STRAPS

REF. 395-0986

MATERIAL — AM-355

$F_{TU}$ = 200 000 PSI

$F_{TY}$ = 180 000 PSI



LAMINATE THICKNESS = .020"

TOTAL NUMBER OF LAMINATES = 25

LOADS AND MOTIONS (NOTE LIMIT COND = 105% x 750 FPS TIP SPEED)

| LOADS | FATIGUE COND | LIMIT MANEUVER COND |
|---|---|---|
| $V_{X,2R}$ | ± 370# | ± 1850# |
| $M_{C,2R}$ | ± 100,000"# | ± 375,000"# |
| C. F. | + 180,000# | + 248 000# |
| MOTIONS | | |
| FLAPPING | ± 5° (NOTE-STRAP INSTALLED AT 8° PRE-CONING) | +25° (NOTE. 8° CONING BUILT IN) |
| LEAD-LAG | ± 1.25° | ± 3° |
| FEATHERING | ± 12° (STRAPS UN-TWISTED AT 7° [illegible] R) | +28° (NOTE. +7° 18 BUILT IN) |

## FLAPPING-FEATHERING BLADE STRAPS

## SUMMARY OF STRESSES

| | FATIGUE COND. | | | LIMIT MAN. COND | | |
|---|---|---|---|---|---|---|
| SECTION | AA | BB | CC | AA | BB | CC |
| $f_1$ - C.F. STRESS | +36600 | +36600 | +61000 | +50500 | +50500 | +84000 |
| $f_2$ - PACK BENDING | ±10150 | ±10150 | ±16900 | ±35000 | +35000 | +57250 |
| $f_3$ - LAMINATE BEND | — | 7250±7250 | — | — | +15000 | — |
| $f_4$ - STRAP WIND UP | — | ±1200 | ±5300 | — | +7370 | -36400 |
| $f_5$ - LEAD LAG SHEAR | ±4950 | ±4950 | ±6250 | +17000 | +17000 | +28400 |
| $f_6$ - CHORD MOMENT | ±2625 | ±2625 | ±4375 | +9800 | +9800 | +16300 |

$$f_{TOTAL} = f_1 + \left[f_2 + f_3\right] \cos(\psi \cdot 30^\circ) + \left[f_4 + f_5 + f_6\right] \sin \psi$$

$$f_{TOTAL\ MAX} \text{ WHEN } \tan \psi = \frac{2(f_4 + f_5 + f_6) - (f_2 + f_3)}{\sqrt{3}\ (f_2 + f_3)}$$

MAX. STRESS FATIGUE CONDITION

SEC. AA @ BOLT HOLE = 36,600 ± 13425 PSI

SEC. BB IN SHOE = 43850 ± 21780 PSI

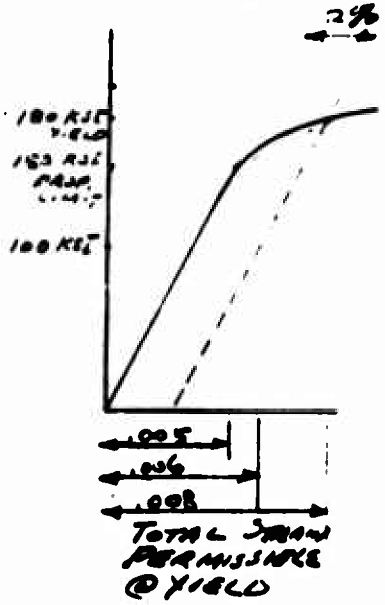
SEC CC = 61000 ± 26550 PSI

THESE STRESSES ARE REASONABLE IN VIEW OF THE FAIL SAFE FEATURE OF THIS DESIGN. AT THE BOLT HOLE, SEC. AA, THE LAMINATES ARE TIGHTLY CLAMPED TOGETHER SO THERE IS NO WORKING IN THE BOLT HOLE.

THIS STRAP DESIGN IS SIMILIAR AND IS BASED ON THE PROVEN OH-6A HELICOPTER BLADE RETENSION STRAP CONFIGUERATION.

## FLAPPING-FEATHERING BLADE STRAPS

SEE - C.C. ANALYSIS LIMIT MANEUVER COND.



CHECK FOR WELD AT LIMIT LOAD

$$\frac{\Sigma f}{E} = \frac{128700 \text{ PSI}}{29. \times 10^6} = .0044$$

DUE TO DIRECT LOADS

$$\frac{\Sigma f}{E} = \frac{93650 \text{ PSI}}{29 \times 10^6} = .0032$$

STRAIN DUE TO MOTIONS

$$\underset{@ \text{ YIELD}}{M.S.} = \frac{.008}{.0044 + .0032} - 1 = +.05$$

CHECK FOR ULTIMATE STRENGTH.

$\Sigma$ STRESS FROM LOADS $= \Sigma f_1 + f_5 + f_6$

$= 128700$ Psi

$$\underset{@ \text{ Ult.}}{M.S.} = \frac{200000}{128700 \times 1\frac{1}{2}} - 1 = .04$$

60000 ± 30000 PSI FOR SMOOTH AREAS FREE OF CONCENTRATION EFFECTS AND 40000±15000 PSI IN THE AREA OF A BOLTED ATTACHMENT WHICH IS CLAMPED SO TIGHTLY THAT THE LOAD IS CARRIED BY FRICTION ENTIRELY. THE STRAP SYSTEM IS MULTIPLY FAIL SAFE.

## FLAPPING - FEATHERING BLADE STRAPS

### FATIGUE CONDITION

$f_1$ = C.F. STRESS = $\frac{C.F}{AREA}$

| SEC AA | SEC BB | SEC CC |
|---|---|---|
| +36600 PSI | +36600 PSI | + 61000 PSI |

$f_2$ = PACK BENDING STRESS (STEADY CONING ANGLE BUILT IN.)

$\beta = 5° = .0872$ RAD  PACK THICK. = $25(.020'') + 24(.004) = .54$

$\Delta = \frac{.546''}{2}(.0872) = .026'' = \frac{PL}{AE} = \frac{PL}{wtE}$  $E = 29 \times 10^6$

$\frac{10.5P}{8.3tE} + \frac{27P}{3.0tE} + \frac{4.4P}{4tE} + \frac{8.5P}{5.5tE} = .026$  $P = 1015^{\#}$

SEC AA = $\frac{1015}{.02} \times \frac{1}{5} = \pm 10150$ Psi

SEC BB = $\frac{1015}{.02} \times \frac{1}{5} = \pm 10150$ Psi

SEC CC = $\frac{1015}{.02} \times \frac{1}{3} = \pm 16900$ Psi

$f_3$ = LAMINATE BENDING (OCCURS ONLY AT SHOE SEC BB)

$f_3 = \frac{Et}{2R}$  R = SHOE RADIUS

$f_3 = \frac{29 \times 10^6 \times .02}{2 \times 20} = 14500$ Psi  SET SHOE FOR $7250 \pm 7250$

$f_4$ = STRAP WIND UP

$I = \frac{.50 \times 3^3}{12} = 1.125\ IN^4$  $\frac{EI}{P} = \frac{29 \times 10^6 \times 1.125}{91500} = .357 \times 10^3$

$J = \sqrt{\frac{EI}{P}} = 18.9$  $\frac{L}{J} = \frac{40.54}{18.9} = 2.15$, $\sinh \frac{L}{J} = 4.2342$, $\cosh \frac{L}{J} = 4.35$

$\gamma_2 = \frac{R_1(1-\cos\theta)}{L}$, $R_1 = \frac{18''}{2} = 9''$  $L = 40.54$  $\theta = 12°$  $\gamma_2 = .00465$

$\gamma_1 = \frac{R_2(1-\cos\theta)}{L}$, $R_2 = 2''$  $\gamma_1 = .001080$

## FLAPPING - FEATHERING BLADE STRAPS

### FATIGUE CONDITION

$f_4$ = STRAP WIND UP (CONTINUED)

$$P\gamma_1 = \frac{1}{J}\left[\frac{M_2 - M_1 \cosh \frac{L}{J}}{\sinh \frac{L}{J}}\right] - \frac{1}{L}\left[M_1 - M_2\right]$$

$$99 = ,0372 M_2 - .0792 M_1$$

$$P\gamma_2 = \frac{1}{J}\left[\frac{M_1 - M_2 \cosh \frac{L}{J}}{\sinh \frac{L}{J}}\right] + \frac{1}{L}\left[M_1 - M_2\right]$$

$$444 = ,0372 M_1 -,0792 M_2$$

$M_1 = -4980''^{\#}$ $M_2 = -7950''^{\#}$

SEC. AA @ BOLT HOLE $f_4 = 0$

SEC. BB = $\frac{MC}{I} = \frac{4980(2.5)}{5.2} = 2400$ PSI $\times \frac{1}{2} = \pm 1200$ Psi

SEC, CC = $\frac{MC}{I} = \frac{2950(1.5)}{1.125} = 10600$ PSI $\times \frac{1}{2} = \pm 5300$ PSI

$f_5$ - LEAD LAG SHEAR

$$\frac{\pm 1.25^\circ}{57.3^\circ}(180,000''^{\#}) \pm 370 = \pm 4295^{\#}$$

$$f_5 = \frac{\frac{1}{2}(4295^{\#})}{\sin 10^\circ} \times \frac{1}{AREA}$$ = ± 8250 PSI @ SEC CC

= ± 4950 PSI @ SEC AA & BB

$f_6$ - CHORD MOMENT

$$\frac{M_{C.R.}}{\text{DISTANCE FEATHER BALL TO LEAD LAG ANGLE}} = \frac{\pm 100\ 000}{44''} = \pm 2270^{\#}$$

$$f_6 = \frac{\frac{1}{2}(2270)}{\sin 10^\circ} \times \frac{1}{AREA}$$ = ± 4375 PSI @ SEC. CC

= ± 2625 PSI @ SEC AA & BB

## FLAPPING - FEATHERING BLADE STRAPS ATTACHMENTS

BOLTS AT HUB END

$$\text{MAX. C.F.} = \frac{282000}{2 \cos 10^\circ} = +143{,}800^{\#}$$

$$\text{DAMPER MOM.} \ \frac{500\,000''^{\#}}{44''} \times \frac{1}{2} \times \frac{1}{\sin 10^\circ} = +32{,}800^{\#}$$

$$\text{L.L. SHEAR} = [282000 \sin 3^\circ + 475] \times \frac{1}{2} \times \frac{1}{\sin 10^\circ} = +43{,}700^{\#}$$

$$\underline{\phantom{000000}} \ 220{,}000^{\#}$$

USE 2 - 1.0" BOLTS @ 200 - 220 KSI H.T.

D.S. ALLOW - 2 BOLTS = 332000#

$$\text{M.S.} = \frac{332000}{220000 \times 1\frac{1}{2}} - 1 = 0$$

BOLT CLAMP UP REQUIRED FOR CYCLIC LOADS

$$\left.\begin{array}{l} f_2 \text{ - PACK BENDING} = \pm 10150 \text{ PSI} \\ f_5 \text{ - LEAD LAG SHEAR} = \pm 4950 \text{ PSI} \\ f_6 \text{ - CHORD MOMENT} = \pm 2625 \text{ PSI} \end{array}\right\} \text{@ SEC A A.}$$

$$\left[\pm \frac{10150}{2} \pm \frac{4950}{2} \pm \frac{2625}{2}\right] 5.00 \times .50 = \pm 18070^{\#}$$

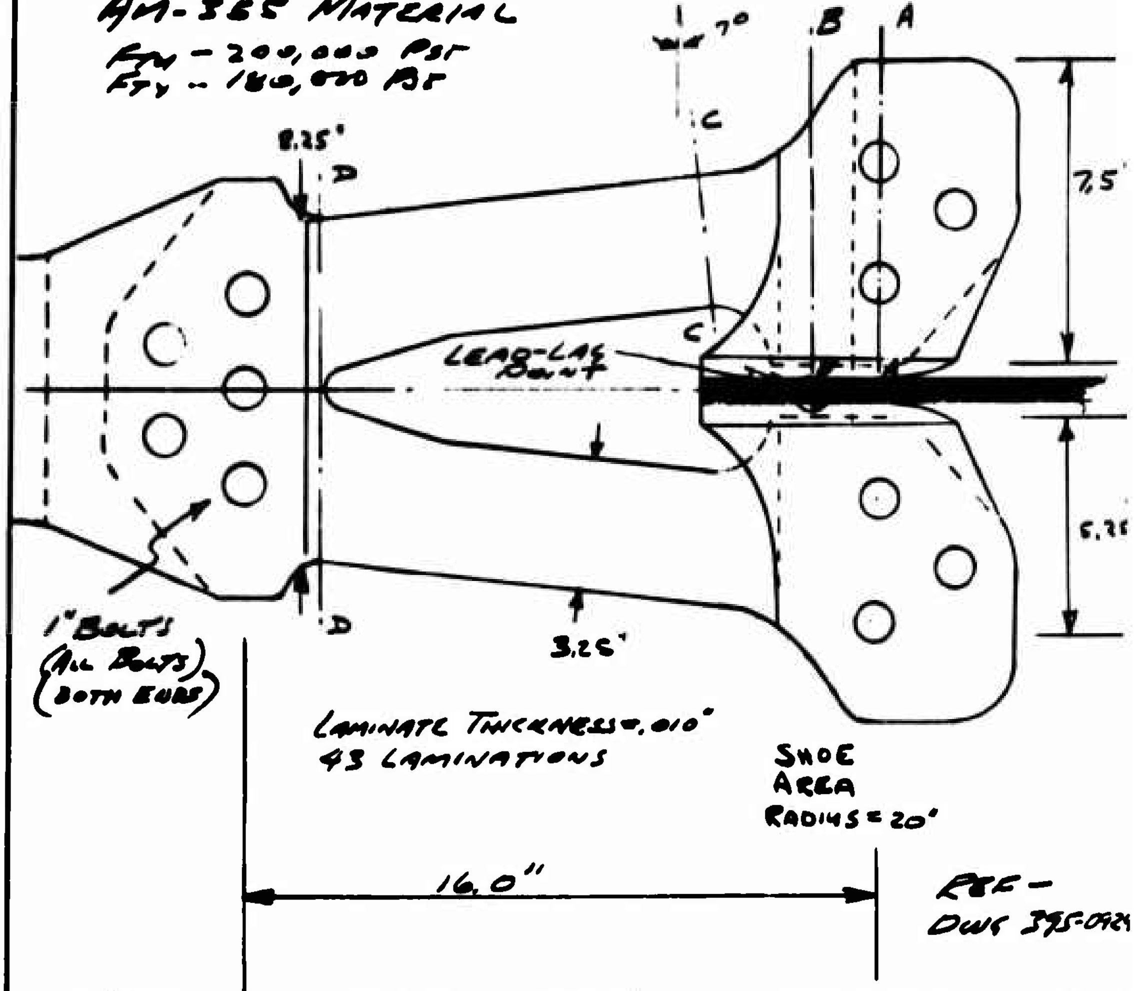
ULT. TENSILE LOAD FOR 1.0" BOLT H.T. 200-220 = 141000#

$$P = \mu N = .3(.6)(141000) = 25{,}400 > 18070^{\#}$$

NOTE - ATTACHMENTS ARE CHECKED FOR C.F. @ 840 F.P.S. TIP SPEED (1.25 × 673) AND MAXIMUM DAMPER MOMENT AND LEAD LAG SHEAR.

## LEAD-LAG BLADE FLEXURE

AM-355 MATERIAL
FTU - 200,000 PSI
FTY - 180,000 PSI



| LOADS | FATIGUE COND. | LIMIT MANEUVER COND |
|---|---|---|
| $V_Z$ | 4200 ± 5100 # | 5100 ± 5100 # |
| $M_{C.R}$ | ± 100,000 "# | ± 375,000 "# |
| $M_F$ | 43,250 ± 52000 | 134000 ± 134000 |
| C.F | +180,000 # | 248 000 #<br>@ 105% OF 750 F.P.S |
| MOTIONS | | |
| LEAD LAG | ± 1.25° | ± 3° |

## LEAD-LAG BLADE FLEXURE

$I_{DO}$ MUST EQUAL $20^{IN^4}$ TO EQUAL I OF BLADE SPAR

REQUIRED AREA $= \frac{C.F}{60000-\underset{\text{DUE TO BEND.}}{9000}} = \frac{180000}{51000} = 3.53^{IN^2}$

$I = \frac{BH^3}{12} = \frac{3.53H^2}{12} = 20$ $H = 8.25"$ $\frac{3.53^{IN^2}}{8.25"} = .43"$

LAMINATE THICKNESS = .010"

PACK THICKNESS = .43" + 42(.004) = .60"

### FATIGUE CONDITION

$f_1$ = STEADY C.F STRESS = $\frac{C/F}{AREA}$

| SEC | AA | BB | CC | DD |
|---|---|---|---|---|
| AREA | $2.37^{IN^2}$ | $2.26^{IN^2}$ | $1.52^{IN^2}$ | $3.53^{IN^2}$ |
| $f_1$ | 38000 PSI | 40000 PSI | 60000 PSI | 51,000 PSI |

$f_2$ = PACK BENDING STRESS

$$\Delta = \frac{b}{2}\beta = \frac{.60}{2}\left(\frac{1.25}{57.3}\right) = .0065 = \varepsilon\frac{PL}{AE}$$

$$\frac{P(2)}{4.13} + \frac{P(12)}{3.25} + \frac{P(2.5)}{5.25} = .0065Et = 4.66P, \quad P = 40300t$$

| SEC. | AA | BB | CC | DD |
|---|---|---|---|---|
| WIDTH | (7.5-2.0) | 5.25 | 3.25 | 4.13 |
| $f_2$ | ±7350 | ±7700 | ±12400 | ±9750 |

$f_3$ = LAMINATE BENDING STRESS

OCCURS IN SHOE ONLY SEC B.B.

$$f_3 = \frac{Et}{2R} = \frac{29 \times 10^6 \times .01}{2 \times 20} = 7500 \text{ PSI} = 3750 \pm 3750 \text{ PSI}$$

9704

## LEAD-LAG BLADE FLEXURE

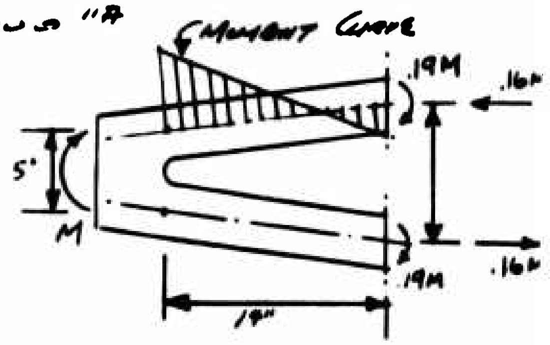
## FATIGUE CONDITION

### $f_L$ - FLAPPING BENDING STRESSES.

$M_F = 43250 \pm 52000$ "#

$.19 M_F = 8225 \pm 9900$ "#

$.16 M_F = 6920 \pm 8320$ #



| SEC. | At Bolt Hole<br>AA | BB | CC | DD |
|---|---|---|---|---|
| AREA | 2.37 IN² | 2.26 IN² | 1.52 IN² | 3.53 IN² |
| C/I | 2.00/14.10 | 2.63/5.19 | 1.63/1.23 | 4.13/20 |
| $.16 M_F/A$ | 2910 ± 3500 | 3060 ± 3700 | 4550 ± 5500 | — |
| $.19 \frac{M_F C}{I}$ | 1170 ± 1400 | 4320 ± 5220 | MOMENT = 0 | 8950 ± 107 |
| | 4080 ± 4910 | 7380 ± 8920 | 4550 ± 5500 | 8950 ± 107 |

### $f_S$ - CYCLIC AXIAL STRESS DUE TO DAMPER MO

$M_C = \frac{\pm 100\,000 \text{"\#}}{14"} = \pm 7150$ # HALF GOES TO EAC LEG

| SEC. | AA | BB | CC | DD |
|---|---|---|---|---|
| P/A | ± 1500 Psi | ± 1580 Psi | ± 2350 Psi | ± 2000 Ps |

9704

## LEAD-LAG BLADE FLEXURE

### SUMMARY OF STRESSES

| | FATIGUE COND. | | | | LIMIT COND. | | | |
|---|---|---|---|---|---|---|---|---|
| SECT. | AA | BB | CC | DD | AA | BB | CC | DD |
| | @ Bolt Hole | | | | @ Bolt Hole | | | |
| $f_1$ CF Load | +38000 | +40000 | +60000 | +51000 | +52500 | +55000 | +82500 | +70000 |
| $f_2$ [illegible] Bend | ± 7350 | ± 7700 | ±12400 | ± 9750 | +17700 | +18500 | +29800 | +23400 |
| $f_3$ Lag Bend | —— | 5250±3750 | —— | —— | —— | +7500 | —— | —— |
| $f_4$ Flap Bend | [illegible] | 7380±8920 | 4550±5500 | 8950±10750 | +25450 | +44900 | + 2700 | +10500 |
| $f_5$ [illegible] Mom | ±1500 | ±1600 | ±2350 | ±2000 | +5650 | +5900 | +7800 | +7500 |
| | | | | | 101,800 | 121,800 | 122800 | 111400 |

TOTAL FATIGUE STRESSES.

$$f_{TOTAL} = f_1 + [f_2 + f_3 + f_5] \sin\psi + f_4 \cos(\psi - 30^\circ)$$

$$\text{MAX WHEN } \tan\psi = \frac{2(f_2 + f_3 + f_5) - f_4}{\sqrt{3}\, f_4}$$

MAX. STRESS FATIGUE CONDITION

SEC AA @ BOLT HOLE = +43080 ± 11790 Psi

SEC BB @ SHOE = +51,130 ± 18200 Psi

SEC CC = +64,550 ± 18020 Psi

SEC DD = +60,750 ± 17350 Psi

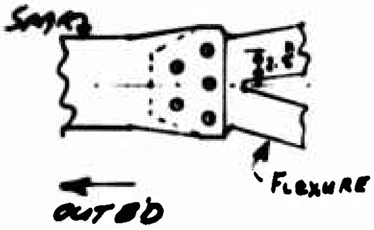
CHECK FOR ULTIMATE STRENGTH

$$M.S. = \frac{200\,000}{122800 \times 1.5} - 1 = +.08$$

## LEAD-LAG BLADE FLEXURE

## BOLT ATTACHMENT

THE OUTBOARD ATTACHMENT IS SHOWN AS THIS ATTACHMENT IS MADE UP OF 5 BOLTS WHERE AS THE INBOARD ATTACHMENT OF THE FLEXURE HAS SIX BOLTS



$I_{BOLT\ GROUP} = 18.9^{IN^4}$

$MAX.\ C.F. = \frac{282000^{\#}}{5\ BOLTS} = 56400^{\#}$

AT 840 FPS [illegible] REV

$DAMPER\ MOM = \frac{500000^{"\#}}{14" \times 5\ BOLTS} = 7150^{\#}$

$FLAP\ MOM,\ \frac{MC}{I} = \frac{268000(2.5)}{18.9} = \frac{35500^{\#}}{99050^{\#}}$

USE 5-1.0" BOLTS @ 200-220 KSI. HT.

D.S. ALLOWABLE = 166000#

$M.S. = \frac{166000}{99050 \times 1\frac{1}{2}} - 1 = +.11$

BOLT CLAMP UP REQUIRED FOR CYCLIC LOADS

$f_2$ - PACK BENDING = ±9750 PSI } @ SEC DD PREVIOUS PAGE
$f_5$ - DAMPER MOM. = ±2000 PSI }

$$\left[\frac{\pm 9750\ PSI}{2} \pm 2000\ PSI\right] \underbrace{8.25" \times .43"}_{DIMENSIONS\ OF\ SEC.\ DD} \times \frac{1}{5\ BOLTS} = \pm 4870^{\#}$$

$$CYCLIC\ FLAP\ MOM = \frac{MC}{I} = \pm\frac{82000(2.5)}{18.9} = \frac{\pm 6400^{\#}}{\pm 11770^{\#}}$$

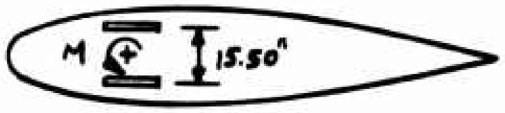
$F_T(ULT)$ OF 1" BOLT = 141000#

$P = \mu N = .3(60\%\ OF\ 141000^{\#}) = 25400^{\#}$

$M.S. = \frac{25400}{11770} - 1 = +1.16$

NOTE - ATTACHMENTS ARE CHECKED FOR C.F. @ 840 F.P.S. TIP SPEED (1.25 x 675 FPS) AND MAXIMUM DAMPER MOMENT & FLAPPING MOMENT

## BLADE TORSION FLEXURES AT LEAD LAG HINGE



FLEXURES RESIST
BLADE TORSION AND
MUST FLEX WITH
THE BLADE LEAD LAG
ANGLE.

FLEXURE 2.0" WIDE BY 7.5" LONG
.20" THICK

### FATIGUE CONDITION

$M_T = 63{,}000 \pm 75{,}500''\#$ $\delta = \pm 1.25^\circ$ LEAD LAG

$$P = \frac{63000 \pm 75500}{15.50''} = 4070\# \pm 4870\#$$

$$\delta = \frac{ML}{EI}, \quad f_b = \frac{Mc}{I} \qquad f_b = \frac{\delta E t}{2L} = \frac{\pm 1.25^\circ}{57.3} \times \frac{29 \times 10^6 t}{15} = \pm .422 \times 10^5 t$$

$$\frac{P}{A} + f_b = \frac{4070 \pm 4870}{2'' \times .200} \pm .422 \times 10^5 (.200) = 10150 \pm 20590$$

$$M.S. = \frac{\pm 25000}{\pm 20590} - 1 = +.21$$

### LIMIT CONDITION

$M_T = 228000''\# \pm 228000''\#$ $\delta = \pm 4^\circ$ LEAD LAG

$$P = \frac{228000 \pm 228000}{15.50} = 14700 \pm 14700 = 29400\#_{MAX}$$

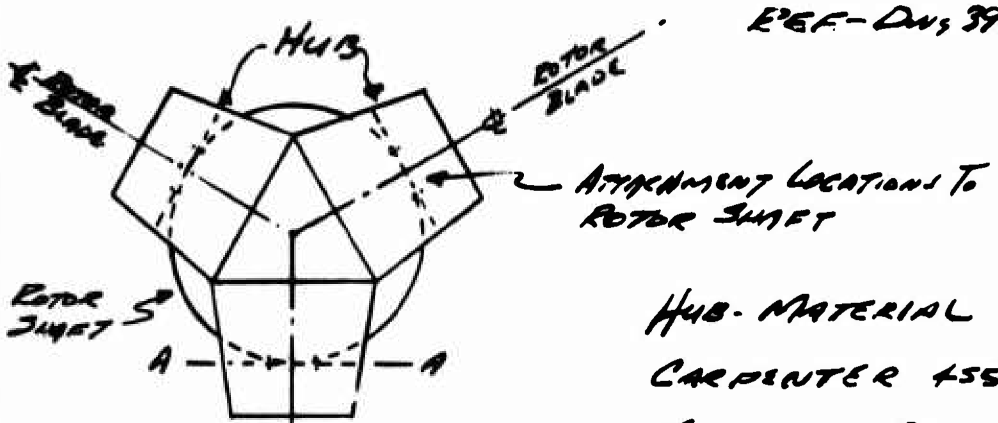
$$\frac{P}{A} + f_b = \frac{29400\#}{2 \times .20} + .422 \times 10^5 \times .125 \times \frac{4^\circ}{1.25^\circ}$$

$$= 73500 + 27000 = 100500 \text{ PSI}$$

$$M.S. = \frac{257000}{100500 \times 1\frac{1}{2}} - 1 = +.70$$

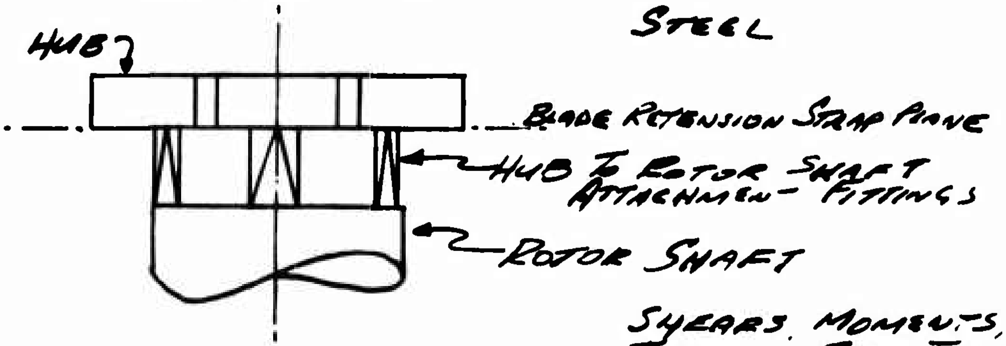
## ROTOR HUB ANALYSIS

REF-DWG 395-09X



PLAN VIEW OF HUB

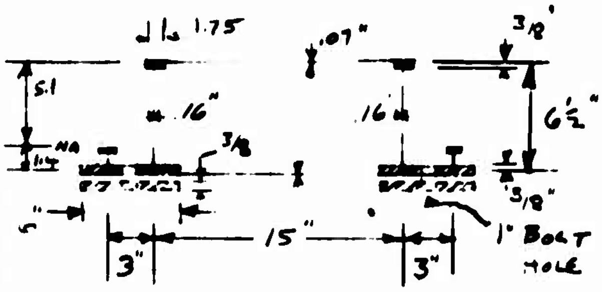
HUB-MATERIAL
CARPENTER 455
MARAGING STAINLESS
STEEL



ELEVATION VIEW OF HUB & ATTACHMENT FITTINGS CONNECTING THE HUB TO THE ROTOR SHAFT

SHEARS, MOMENTS, AND TORQUES FROM THE ROTOR BLADES ARE REACTED IN THE HUB THUS REDUCING CONSIDERABLY THE LOADS TRANSMITTED TO THE ROTOR SHAFT. THE ROTOR SHAFT EXPERIENCES ONLY THE UNBALANCED PORTION OF THESE LOADS. THE NET LOADS ARE TRANSMITTED TO THE ROTOR SHAFT BY THE ATTACHMENT FITTINGS WHICH PROVIDE A SHORT DIRECT LOAD PATH. THE LOADS ENTER THE ROTOR SHAFT IN THE PLANE OF THE ROTOR SHAFT WALL TAKING ADVANTAGE OF THE AVAILABLE STRENGTH AND STIFFNESS IN THIS DIRECTION OF THE SHAFT. RADIAL STIFFENING REQUIREMENTS ARE GREATLY REDUCED. AS A RESULT OF THESE SHORT LOAD PATHS AND EFFICIENT LOADING OF THE HUB AND ROTOR SHAFT, THEY ARE OPTIMIZED FOR MINIMUM WEIGHT.

## ROTOR HUB ANALYSIS



| | LOADS @ SECT. A-A | |
|---|---|---|
| | FATIGUE | LIMIT |
| ⊗ $M_c$ | ±118,650 | 852,000 in-# |
| $M_F$ | 228,000±178,000 | 1,055,000 in-# |
| $M_-$ | ±44,250 | 425,000 in-# |
| ⊗ $V_x$ | ∓4550 | 22,700 # |
| ⊗ $V_z$ | 35,730 ± 27370 | 102,000 # |
| Axial-$V_y$ | 178,000 ± 2800 | 260,000 # |

REF-LOADS FROM LOADS SECTION

⊗ THESE LOADS ARE APPLIED AND REACTED IN THE LOWER CAP MEMBERS

BENDING SECTION OF HUB AT A-A

$I_{xy} = 43.4^{in^4}$ (TOP SKIN INEFFECTIVE)

**TOP CAP MEMBERS**

$f = \frac{Mc}{I} = 20800 \pm 21,000$ Psi

$= 124000$ Psi

NOTE - "THIS SECTION IS FREE OF BOLT HOLES & OTHER STRESS CONCENTRATIONS

FATIGUE LOADING

$M.S. = \frac{25000}{21000} - 1 = +.19$

$M.S. = \frac{257000}{124000\ \text{Psi}} - 1 = +.38$

ULTIMATE CONDITION

**LOWER CAP MEMBER**

THROUGH BOLT HOLE ATTACHING BLADE RETENSION STRAPS

| | FATIGUE | LIMIT |
|---|---|---|
| $\frac{M_c}{18"} =$ | ±660 # | 4725 # |
| $\frac{\text{AXIAL}}{2} =$ | 8900 # ± 140 | 13000 # |

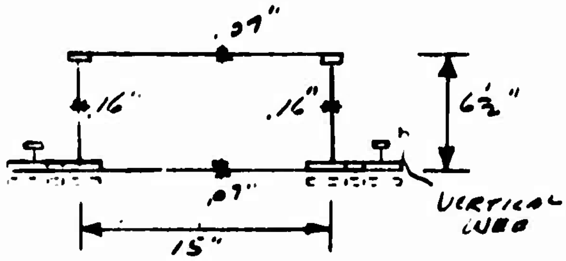
NET AREA = $[6'-1"]\,3/8 \times 2 = .375^{in^2}$

| | | |
|---|---|---|
| P/A DUE TO $M_c$ = | ±1760 PSI | 12,600 PSI |
| P/A DUE TO AXIAL = | 23700 ± 373 Psi | 34,700 Psi |
| $\frac{Mc}{I}$ DUE TO $M_F$ = | 7400 ± 5750 Psi | 34,100 Psi |
| | 31100 ± 7885 Psi | 81,400 Psi |

$M.S. = \frac{\pm 10000}{\pm 7883} - 1 = +.27$ FATIGUE @ BOLT HOLE

$M.S. = \frac{257000}{81400\ \text{Psi}} - 1 = +1.10$ ULTIMATE

# ROTOR HUB ANALYSIS



ENCLOSED AREA =
$6\frac{1}{2} \times 21 = 136.5 \text{ IN}^2$

NOTE - THE VERTICAL WEBS INBOARD OF THE HUB SYMMETRY CAN BE REDUCED TO .08" AS THE SHEAR IS REDUCED TO HALF

CHECK VERTICAL WEB

| | FATIGUE | LIMIT MANEUVER |
|---|---|---|
| TORQUE SHEAR $= \frac{M_z}{2At}$ | ± 1485 | 13600 |
| VERTICAL SHEAR $= \frac{V_z}{2(6\frac{1}{2}).16}$ | 16800 ± 13000 | 78,000 PSI |
| | 16800 ± 14485 PSI | 91,600 PSI |

$F_{S_{ALLOWABLE}}$ FATIGUE = 60% TENSION FATIGUE ALLOWABLE = 60% × 25000 PSI = 15000 PSI

$$M.S. = \frac{15000}{14485} - 1 = 0$$

CHECK STIFFENER SPACING FOR LIMIT MANEUVER CASE FOR VERTICAL WEB.

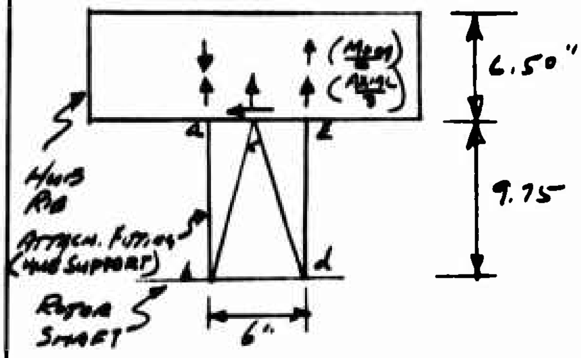
$$f_s = 5.3 E \frac{t^2}{b^2} = 5.3 (27 \times 10^6) \frac{.16^2}{b^2} = 91600 \text{ PSI}$$

$b = 6.3$" STIFF REQUIRED AT 6.3" SPACING

9704

## ROTOR HUB ANALYSIS

### RIB & ATTACHMENT FITTING TO ROTOR SHAFT



FITTING LOADS FROM LOADS SECTION

| | LIMIT MAN. | FATIGUE |
|---|---|---|
| AXIAL | 192500# | 35100±384 |
| MOM | 193920 | ±2,186 |
| SHEAR | 3700 | ±740 |

| MEMBER → | ab | bc | cd | de |
|---|---|---|---|---|
| LIMIT MAN. | +31,880# | +39,900# | +27,300 | +96520# |
| FATIGUE | 11700±9668 | 6120±5440 | 6120±7160 | 11,700±16332 |
| CROSS SECT AREA | .9 IN² | .39 IN² | .39 IN² | 9 IN² |

$$M.S._{ULT.} = \frac{257000}{\frac{96520 \times 1\frac{1}{2}}{.9}} - 1 = \underline{\underline{+.60}}$$

$$M.S._{FATIGUE} = \frac{25000}{\frac{7160^{\#}}{.39}} - 1 = \underline{\underline{+.24}}$$

NOTE - CROSS SECTION AREA OF THESE MEMBERS IS DETERMINED BY THE STIFFNESS ANALYSIS OF THE HUB PLATE AND ATTACHMENT FITTING AND NOT BY STRENGTH REQUIREMENTS AS EXPLAINED IN THE LOADS SECTION

ALLOWABLES FOR CARPENTER 455 MARAGING STAINLESS STEEL ARE TAKEN FROM THE MATERIALS ALLOWABLE SECTION.

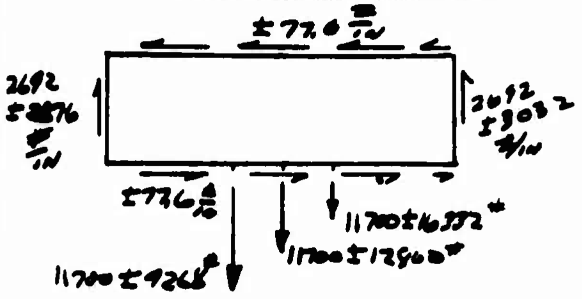
THE ANALYSIS PERTAINS TO TYPICAL MEMBER SECTIONS AND THE LOCAL AREAS AROUND ATTACHMENT BOLTS ARE REINFORCED TO REDUCE STRESSES

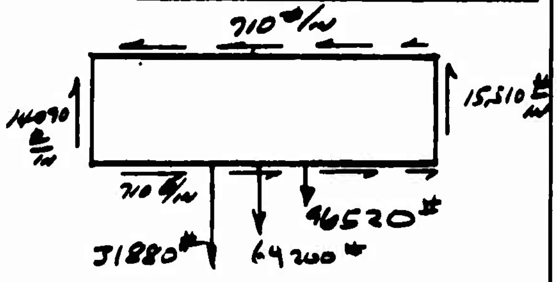
# ROTOR HUB ANALYSIS

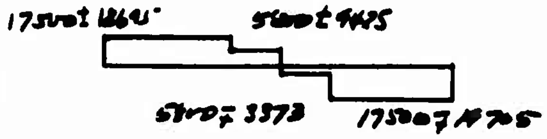
## ANALYSIS OF HUB RIB AT SEC. A-A

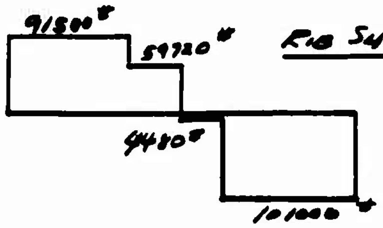
MATERIAL CARPENTER 455 MARAGING STEEL

**FATIGUE COND.** **LIMIT MANEUVER COND**

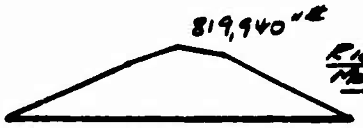




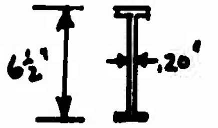




RIB SHEAR



RIB BEND. MOMENT



TYPICAL RIB CROSS SECTION AT MAX MOM.

$I = 22.0\ in^4$ CAP AREA $= .98\ in^2$

WEB SHEAR STRESS (FATIGUE)

$$f_s = \frac{2692 \pm 3032\ \#/in}{.2} = 13,460 \pm 15000\ psi$$

$$M.S = \frac{15000}{15000} - 1 = 0$$

FOR FATIGUE SHEAR USE 60% OF FATIGUE BEND ALLOW
$= .60 \times \pm 25000 = 15000$

CAP STRESS

$$\frac{Mc}{I} = \frac{148400 \pm 163449\left(3\frac{1}{4}\right)}{22} = 22000 \pm 24200\ psi$$

WEB SHEAR STRESS ULT.

$$f_s = \frac{15510\ \#/in}{.20} \times 1\frac{1}{2} = 113000\ psi$$

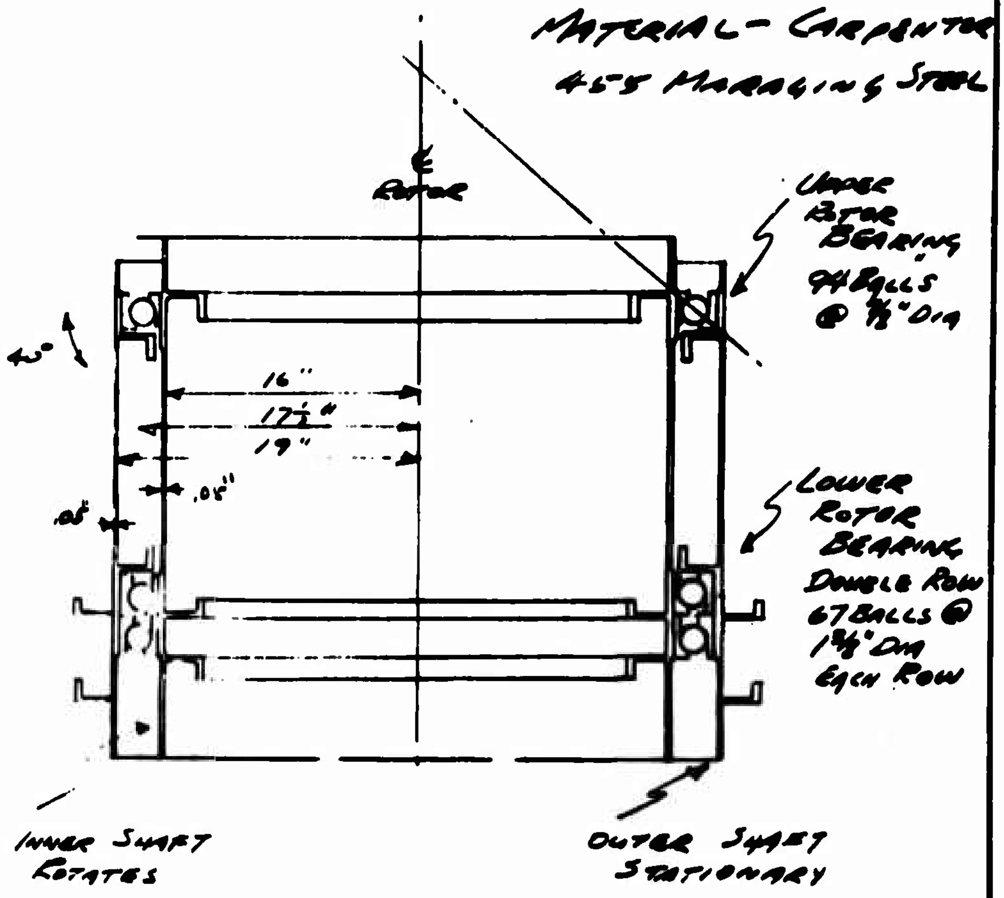
CAP STRESS ULT

$$\frac{Mc}{I} = \frac{819,940\left(3\frac{1}{4}\right) \times 1\frac{1}{2}}{22} = 182000\ psi$$

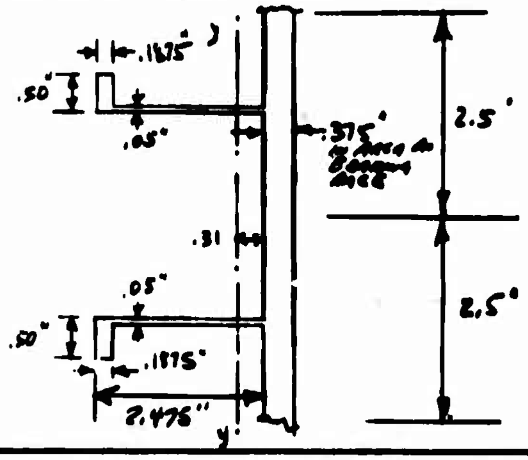
CRITICAL RIB STRESSES ARE FOR THE FATIGUE CONDITION

$$M.S. = \frac{\pm 25000}{\pm 24200} - 1 = +.03$$

## ROTOR SHAFT & BEARING



ALL LIFT LOADS ARE REACTED AT THE LOWER BEARING



TYPICAL STIFFENING AT THE BEARINGS

$I_{YY} = 1.22 \text{ IN}^4$

$A = 1.72 \text{ IN}^2$

## ROTOR SHAFT & BEARING — CONT'D

REFERENCE:- "BALL & ROLLER BRG. ENGINEERING"
(See pages 73 and 90) BY ARVID PALMGREN

**LIFE BASED ON FATIGUE LOADING — SEE LOADS SECTION**

a) **LOWER BRG.** $F_R = 11,400$ #  $F_a = 94,635$ #  $\alpha = 40°$

**EQUIV. LOAD** $P = XF_R + YF_a$  $X = .50$  $Y = .4 \cot 40°$

$$= (.5 \times 11,400) + (.478 \times 94,635)$$

$$= \underline{50,935 \ \#}$$

$f_c = 4.0$

**SPECIFIC CAPACITY** $C = f_c \dfrac{i D_w^2 z^{2/3} \cos\alpha}{1 + .02 D_w}$

$i = 2.0$

$D_w = 34.9$ m/m (1.375")

$z = 67$

$$= \frac{4 \times 2 \times 34.9^2 \times 67^{2/3} \times .766}{1 + (34.9 \times .02)}$$

$$= 72,511 \text{ Kg.} = \underline{159,887 \ \#}$$ (For 1,000,000 cycles B-10 LIFE)

THEN $L_N = \left(\dfrac{159,887}{50,935}\right)^3 \times 10^6$ CYCLES

$$= \underline{30,929,571 \text{ CYCLES}}$$

$N = 143$ RPM

$= 8580$ Rev/HOUR

$\therefore B_{10}$ LIFE IN HOURS $= \dfrac{30,929,571}{8,580} = \underline{3605 \text{ HOURS}}$

b) **UPPER BRG** $F_R = 17,240$ #  $F_a = 2460$ #  $\alpha = 40°$

**EQUIV. LOAD** $P = XF_R + YF_a$  $X = .50$  $Y = .4 \cot 40° = .478$

$$= 8620 + 1176 = \underline{9796 \ \#}$$

$f_c = 4.5$

**SPECIFIC CAPACITY** $C = f_c \dfrac{i D_w^2 z^{2/3} \cos\alpha}{1 + .02 D_w}$

$i = 1.0$

$D_w = 7/8" = 22.2$ m/m

$z = 94$

$$= \frac{4.5 \times 492.8 \times 20.675 \times .766}{1 + .44} = 24,389 \text{ Kg}$$

**UPPER BRG. — CONT'D**

$\therefore \; C = 53{,}777 \; \#$ for $10^6$ cycles

$$L_N = \left(\frac{53{,}777}{9796}\right)^3 = 165{,}000{,}000 \text{ CYCLES}$$

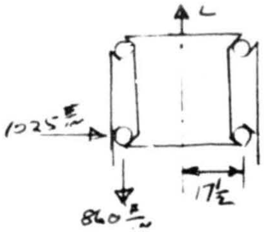
THEN $B_{10}$ LIFE $= \dfrac{165{,}000{,}000}{8580} = \underline{\underline{19{,}250 \text{ hours}}}$

## ROTOR SHAFT & BEARING

CHECK STRESSES IN ROTOR SHAFT AND STIFF. RINGS FOR BEARING HOUSINGS.

### FATIGUE CONDITION

$L = 94{,}590$ REF - ROTOR SHAFT BEARING LOAD TABLE

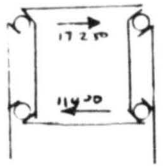


$$W_{vertical} = \frac{94590^{\#}}{2(17\tfrac{1}{2})\pi} = \underline{860\ \tfrac{\#}{IN}}$$

$$W_{radial} = 860^{\#/IN} \times \frac{1}{\tan 40^{\circ}} = 1025\ \tfrac{\#}{IN}$$

$$\text{HOOP TENSION (COMPRESSION)} = 1025\,(17\tfrac{1}{2}) = \underline{18000^{\#}}$$

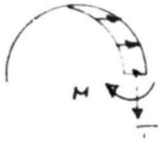
$P_R = 5950 \pm 11300$ (UPPER BEARING) $= 17250$ $P_R = 150 \mp 11300$ (LOWER BEAR) $= 11400$



MOMENT IN STIFFENER RING DUE TO LATERAL BEARING LOAD

$M = .0683\,K R^3$ , $\frac{K \pi R^2}{2} = 17250^{\#}$

$K = 35.9\ \tfrac{\#}{IN^3}$



$M = \underline{13300''^{\#}}$ STEADY MOM. IN UPPER OUTTER STIFF RING

$M = \underline{6650 \pm 6650''^{\#}}$ IN UPPER INNER STIFF RING (ROTATING

$T = .75 K R^2$

$T = \underline{8250^{\#}}$ STEADY IN UPPER OUTER STIFF. RING (NON ROTATING)

$T = \underline{4125 \mp 4125^{\#}}$ IN UPPER INNER STIFF. RING

$W_{vert} = 17\tfrac{1}{2}(35.9)\frac{1}{\tan 40^{\circ}} = 750\ \#/IN$

$W_{vert} = +375 \mp 375\ \#/IN$ LOAD IN PLANE OF INNER SHAFT SKIN

$W_{vert} = 750\ \tfrac{\#}{IN}$ LOAD IN PLANE OF OUTER SHAFT SKIN

FOR VALUES AT LOWER BEARING MULTIPLY BY $11400/17250 = .66$

$M = 8850''^{\#}$ OUTER RING
$M = 4400 \pm 4400$ INNER RING

$T = 5450^{\#}$ OUTER RING
$T = 2725 \mp 2725^{\#}$ INNER RING

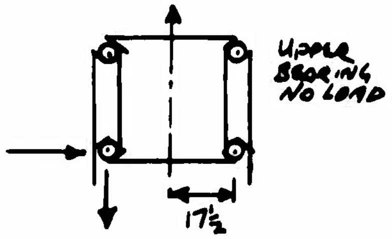
REF - FORMULAS FOR STRESS & STRAIN BY ROARK

## ROTOR SHAFT & BEARING

CHECK STRESSES IN ROTOR SHAFT & STIFFENER RINGS FOR BEARING HOUSINGS

LIMIT MANEUVER CONDITION

$L = 233,150$ REF.- ROTOR SHAFT BEARING LOAD TABLE

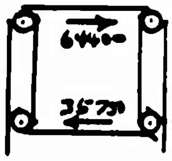


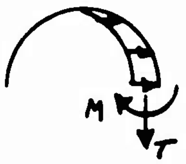
$W_{VERTICAL} = \frac{233,150^{\#}}{2(17\frac{1}{2})\pi} = \underline{2120 \frac{\#}{IN}}$

$W_{RADIAL} = 2120 \frac{\#}{IN} \times \frac{1}{\tan 40^\circ} = 2530 \frac{\#}{IN}$

HOOP TENSION (COMPRESSION) $= 2530 (17\frac{1}{2}) = \underline{44250^{\#}}$

$P_R = 64400^{\#}$ (UPPER BEARING), $P_R = 35,750^{\#}$ (LOWER BEARING)





MOMENT IN STIFFENER RING DUE TO LATERAL BEARING LOAD (UPPER)

$M = .0683\ KR^3$, $\frac{K\pi R^2}{2} = 64400$

$K = 134.0 \frac{\#}{IN^3}$

$M = 49700''^{\#}$ PEAK IN BOTH INNER & OUTER RING

$T = .75 KR^2$

$T = 30800^{\#}$ TENSION IN UPPER OUTER RING COMPRESSION IN UPPER INNER RING

$W_{VERTICAL} = 17\frac{1}{2}(134.0 \frac{\#}{IN^3}) \frac{1}{\tan 40} = 2800 \#/IN$

$W_{VERTICAL} = 2800 \frac{\#}{IN}$ PEAK TENSION IN INNER SHAFT SKIN

$W_{VERTICAL} = 2800 \frac{\#}{IN}$ PEAK COMPRESSION OUTER SHAFT SKIN

FOR VALUES AT LOWER BEARING MULTIPLY BY 35750/64400 = .555

$M = 27600''^{\#}$ PEAK IN LOWER INNER & OUTER RING

$T = 17100^{\#}$ TENSION IN LOWER OUTER RING & COMPRESSION IN LOWER OUTER RING

REF - FORMULAS FOR STRESS & STRAIN BY ROARK

# VI STRESS ANALYSIS

## ROTOR SHAFT & BEARINGS

SUMMATION OF LOADS IN ROTOR SHAFT

|  |  | SHAFT AT UPPER BEARING |  |  |  |
|---|---|---|---|---|---|
|  |  | INNER STIFF RING (ROTATING) |  | OUTER STIFF RING (STATIONARY) |  |
|  |  | T | M | T | M |
| FATIGUE COND. | FROM LIFT-L<br>FROM MOMENT | —<br>-4125∓4125# | —<br>6650±6650"# | —<br>+8250# | —<br>13300"# |
|  | TOTAL | -4125∓4125 | 6650±6650 | +8250# | 13300"# |
| LIMIT COND. | FROM LIFT-L<br>FROM MOMENT-M | —<br>-30,800# | —<br>49700"# | —<br>+30800# | —<br>49700 |
|  | TOTAL | -30,800# | 49700"# | +30800# | 49700"# |

|  |  | SHAFT AT LOWER BEARING |  |  |  |
|---|---|---|---|---|---|
|  |  | INNER STIFF RING |  | OUTTER STIFF RING |  |
|  |  | T | M | T | M |
| FATIGUE COND. | FROM LIFT L<br>FROM MOMENT-M | -18000#<br>-2725∓2725 | —<br>4400±4400 | +18000#<br>+5450# | —<br>8800"# |
|  | TOTAL | -20725∓2725# | 4400±4400 | +23450# | 8800"# |
| LIMIT COND. | FROM LIFT-L<br>FROM MOM. M | -44250#<br>-17100# | —<br>27600"# | +44250#<br>+17150# | —<br>27600"# |
|  | TOTAL | -61,350# | 27600"# | +61,350# | 27600"# |

CRITICAL LOADING FOR FATIGUE T=-4125∓4125# M=6650±6650

CRITICAL LOADING FOR LIMIT T=-30800# M=49700"#

9704

## ROTOR SHAFT & BEARING

SUMMATION OF LOADS IN SHAFT WALLS

| INNER SHAFT WALL | FATIGUE | LIMIT |
|---|---|---|
| DUE TO LIFT | +860 #/IN | +2120 #/IN |
| DUE TO MOMENT | +375 ± 375 #/IN | +2800 #/IN |
| | +1235 ± 325 #/IN | +4920 #/IN |
| $f^*$ = (FOR .05") | 24700 ± 7500 PSI | 98,400 PSI LIMIT |

| OUTER SHAFT WALL | | |
|---|---|---|
| DUE TO LIFT | 0 | 0 |
| DUE TO MOMENT | 750 #/IN | -2800 #/IN |
| $f^*$ = (FOR .05" | 15,000 PSI STEADY | 56000 PSI |

CHECK STRESSES IN STIFFENING RINGS ON SHAFT FOR BEARINGS.

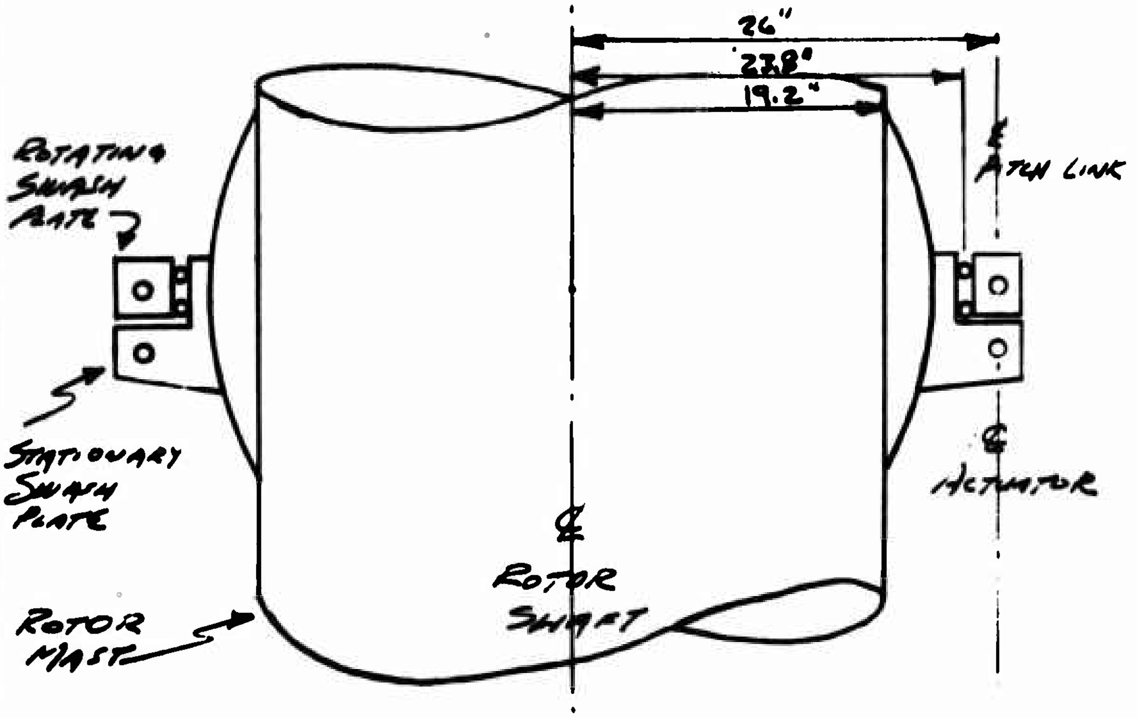
| LOAD | FATIGUE | LIMIT |
|---|---|---|
| T - HOOP LOAD | 4125 ∓ 4425 | 30800 |
| M - BEND. MOM. | 6650 ∓ 6650 | 49700 |

| STRESS | FATIGUE | LIMIT |
|---|---|---|
| $T/A = T/1.72W^2$ | 2400 ∓ 2400 PSI | 18000 PSI |
| $f_b = \frac{MC}{I} = \frac{M(2.16)}{1.22}$ | 11800 ∓ 11800 PSI | 78000 PSI |
| | 14200 ∓ 14200 PSI | 96000 PSI |

*$F_{ALLOWABLE}$ = 50000 ± 25000 PSI 257000 PSI

## FLIGHT CONTROLS

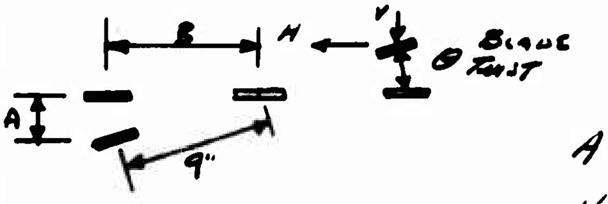
REF. DWG - 395-0936



DISTANCE FROM PITCH ARM TO ℄ FEATHERING BALL = 13"

LENGTH OF PITCH ARM = 17.8"

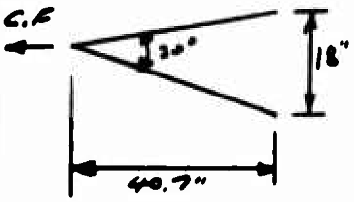
TORQUE DUE TO BLADE RETENSION STRAP TWIST



$$\text{TORQUE} = 2BV - 2AH$$

$$A = .9'' \sin\theta \qquad B = 9'' \cos\theta$$

$$H = \frac{C.F.}{2} \times \tan 10^\circ \qquad V = \frac{C.F.}{2} \cdot \frac{A}{40.7} = \frac{C.F}{2} \frac{9'' \sin\theta}{40.7}$$



$$\text{TORQUE} = 9'' C.F \left[ \frac{9 \sin\theta \cos\theta}{40.7} - .1736 \sin\theta \right]$$

| | FATIGUE | LIMIT |
|---|---|---|
| C. F. | 120,000# | 248,000 |
| θ | ±12° | (14°-7°) ± 14° |
| TORQUE | ±14,580"# | 12250 ± 22320 |

NOTE 7° OF BLADE FEATHER IS BUILT IN

## FLIGHT CONTROLS

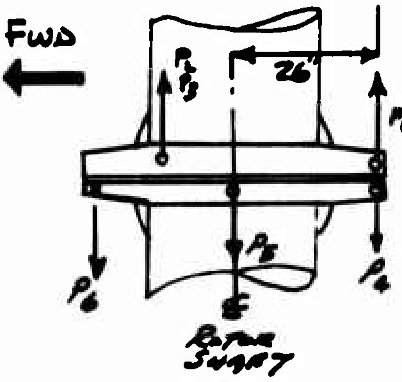
SWASHPLATE BEARING AND ACTUATOR LOADS

LIMIT MANEUVER CONDITION

BLADE TORSION AT ROOT $= M_{T_{ROOT}} = 230,000 \pm 230,000 + M_{STRAP}$

$M_{T_{ROOT\,MAX}} = 230000 \pm 230,000 + (12250 \pm 22320)$

$= 242,250 \pm 252,320$ "#



$P_1 = \frac{242,250 \pm 252,320}{13"} = 18700 \pm 19400$#

PITCH LINK LOAD

MOMENT TO SWASHPLATE $= \pm 19400(\frac{1}{2})26" = 757,000$"# LIMIT

LONGITUDINAL PITCHING

LOAD TO LONGITUDINAL ACTUATORS $P_4$, $P_6$

$$= \frac{18700\# \times 3}{2} + \frac{757000}{2 \times 26"} = \left.\begin{matrix} 42650\# & MAX \\ 13550\# & MIN \end{matrix}\right\} LIMIT$$

LOAD TO LATERAL ACTUATOR, $P_5 = 0$

LATERAL PITCHING

LOAD TO LONGITUDINAL ACTUATORS $P_4$, $P_6$

$$= \frac{18700\# \times 3}{2} + \frac{757000}{2 \times 26"} = \left.\begin{matrix} +42650\# & MAX \\ +13550\# & MIN \end{matrix}\right\}$$

LOAD TO LATERAL ACTUATOR, $P_5$

$$= \frac{75700"\#}{26"} = 29100\# \text{ EITHER DIRECTION}$$

MAXIMUM THRUST ON SWASHPLATE BEARINGS

$= 3(18,700\#) = 56100\#$ LIMIT

8704

## FLIGHT CONTROLS

SWASHPLATE BEARING AND ACTUATOR LOADS

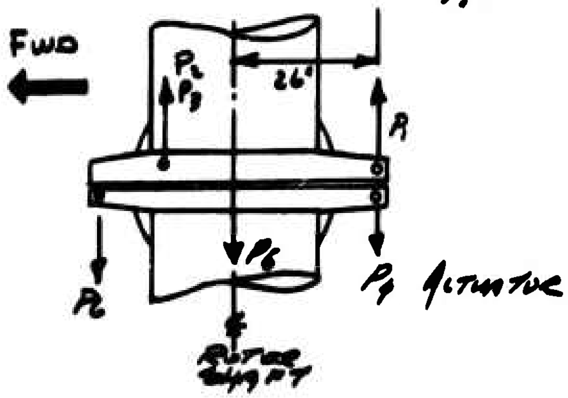
FATIGUE CONDITION

BLADE TORSION AT ROOT $= M_{T_{ROOT}} = 64000 \pm 76000 + M_{STRAPS}$

$M_{T_{ROOT}} = 64000 \pm 76000 + (\pm 14580) = 64000 \pm 90580''^{\#}$

$P_1 = \frac{64000 \pm 90580}{13''} = 4925 \pm 6970^{\#}$

PITCH LINK LOAD



MOMENT TO SWASHPLATE

$= \pm 6970 (\frac{1}{2}) 26'' = 271,000''^{\#}$

MAX THRUST ON SWASHPLATE

$= 3(4925) = 14,775^{\#}$

### LONGITUDINAL PITCHING

LOAD TO LONGITUDINAL ACTUATORS, $P_4$, $P_6$

$$= \frac{4926^{\#} \times 3}{2} + \frac{271000}{2 \times 26'} = \begin{matrix} +12600^{\#} \text{ MAX} \\ +2200^{\#} \text{ MIN} \end{matrix}$$

LOAD TO LATERAL ACTUATOR, $P_5 = 0$

### LATERAL PITCHING

LOAD TO LONGITUDINAL ACTUATORS, $P_4$, $P_6$

$$= \frac{4925^{\#} \times 3}{2} + \frac{271,000}{2 \times 26''} = \begin{matrix} +12600^{\#} \\ +2200^{\#} \end{matrix}$$

LOAD TO LATERAL ACTUATOR

$= \frac{271000}{26''} = 10,400^{\#}$ EITHER DIRECTION

## FLIGHT CONTROLS

**PITCH LINK ANALYSIS** – FATIGUE LOAD = 4925 ± 6970$^{\#}$

LIMIT LOAD = 38,100$^{\#}$ LIMIT

LENGTH = 17.8"

2" × .083", A = .4999 $\rho$ = .6784

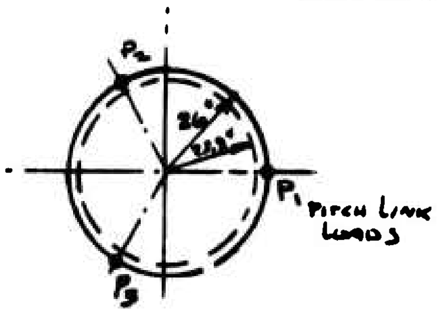
$L/\rho = 26$  $F_c = 120000$ FOR 150000 H.T.

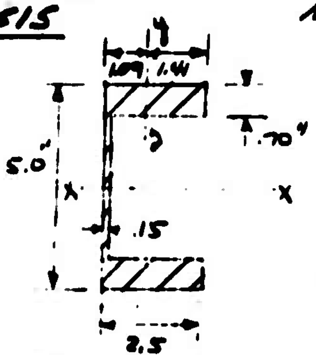
$f_c = \frac{38100^{\#}}{.4999} = 76200$  $M.S. = \frac{120000}{76200 \times 1.5} - 1 = +.05$

$f_{FATIGUE} = \frac{\pm 6970^{\#}}{.4999} = \pm 13950$ Psi

$F_{FATIGUE}$ ALLOWABLE = ±25000 Psi

**ROTATING SWASHPLATE ANALYSIS** (MAT'L CARPENTER 455 MARAGING STEEL)





TYPICAL CROSS SECTION

$I_{XX} = 16.20^{IN^4}$

$A = 4.04^{IN^2}$

$I_{YY\,FLANGE} = 1.23^{IN^4}$

$A_{FLANGE+WEB} = 2.02^{IN^2}$

| CRITICAL LOADS | FATIGUE | LIMIT |
|---|---|---|
| $M_{XX}$ | 33000 ± 92960 "# | 152450 "# |
| $M_{YY}$ FLANGE | 0 ± 7400 "# | 20700 "# |
| $T_1$ HOOP TENSION | 2800# ± 0 | 10650# |
| $T_1$ FLANGE ONLY | 0 ± 2710# | 7510# |
| $f_{TOTAL}$ | 5800 ± 24190 Psi | 53670 Psi LIMIT |
| $F_{ALLOWABLE}$ | ± 25000 Psi | 237000 Psi |
| M.S. = | +.03 | +2.10 |

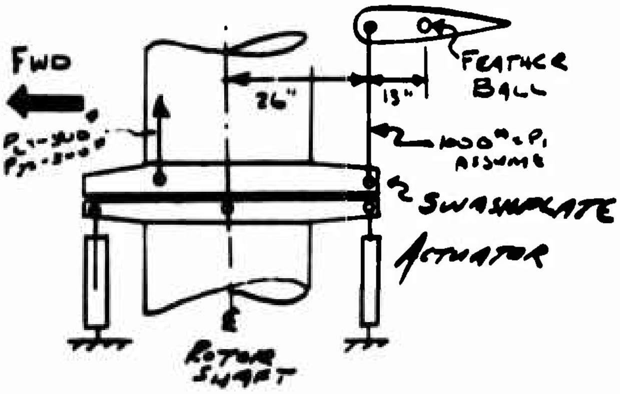
## FLIGHT CONTROLS

### CONTROL SYSTEM STIFFNESS

ASSUMPTIONS

1.- SWASHPLATE IS RIGID

2.- USE FOR ACTUATOR DEFLECTION $1.145\times10^{-5}\frac{in}{\#}$ FROM XV-9A HELICOPTER



$M = 1000^\#(1.5\times26") = 39000^{in\#}$

ACTUATOR LOAD $= \frac{39000}{52} = 750^\#$

REF DWG 395-0836

$U = \frac{\Sigma PL^2}{2AE}$ FOR PITCH LINKS $= \frac{(500^2+500^2+1000^2)/8}{2(.50\,in^2)30\times10^6} = .900$ in#

$U$ FOR SWASHPLATE ASSUMED $= 0$

$U$ FOR ACTUATORS $= \frac{P\Delta}{2} = 2\left[\frac{750}{2}(750\times1.145\times10^{-5})\right] = 4.90$ in#

$5.82$ in#

$\frac{M\theta}{2} = \Sigma U, \quad \frac{39000}{2}\theta = 5.82 \qquad \theta = 3.0\times10^{-4}$ RADIANS

$$\frac{\text{IN. \# OF BLADE ROOT TORQUE}}{\text{CHANGE IN BLADE PITCH (RADIANS)}} = \frac{1000^\#\times13"}{3.0\times10^{-4}\,\text{RADIANS}\times\frac{26"}{13"}} = \underline{\underline{2.17\times10^{7}}}$$

NOTE - VALUE OF STIFFNESS USED IN DYNAMIC ANALYSIS IS $.376\times10^{7}$ IN# BLADE TORQUE/RADIAN OF BLADE PITCH

## FLIGHT CONTROLS

## SWASHPLATE BEARING

REFERENCE 1 – "BALL & ROLLER BEARING ENGINEERING"
BY ARVID PALMGREN

LIFE BASED ON FATIGUE LOADING (REF. LOADS SECTION)

$F_a = 14775$ # $\qquad F_R = 6570$ #

EQUIV. LOAD $P = .5F_R + .478 F_a$

$= 3285 + 7388 = 10,673$ #

SPECIFIC CAPACITY $C = f_c \dfrac{D_w^2\, i\, Z^{2/3} \cos\alpha}{1 + .02 D_w}$

$f_c = 4.5$
$i = 1.0$
$D_w = \frac{1}{2}''$ (12.7 m/m)
$\alpha = 40°$
$Z = 224$

$C = \dfrac{4.5 \times 161.29 \times 224^{2/3} \times .766}{1 + .254}$

$= 16,350$ Kg. (36,050 #)

$\therefore L = \left(\dfrac{36,050}{10,673}\right)^3 \times 10^6 = 38,545,966$ CYCLES.

$N = 143$ RPM $= 8580$ REVS/HR.

$\therefore B_{10}$ LIFE IN HOURS $= \dfrac{38,545,966}{8580} = \underline{4493 \text{ HOURS}}$

## HOT GAS DUCTING (REF-DWG 395-0909)

DUCT MATERIAL IS RENE' 41

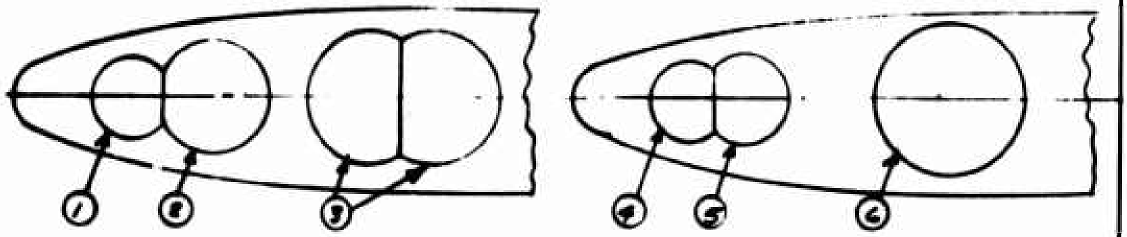
STRENGTH ALLOWABLES

$F_{tu}$ = 150,000 PSI @ 1395° PEAK TEMPERATURE

$E_t$ = 25.3 × 10$^6$ PSI

$F_{t\ .2\%\ CREEP}$ = 38000 PSI @ 1260°F EQUIVALENT TEMPERATURE FOR 3600 HO

DUCT DIMENSIONS IN THE BLADE



BLADE SECTION OUTBOARD OF .75% R

BLADE SECTION INBOARD OF 75% R.

| DUCT | DIAMETER IN. | ENCLOSED DUCT AREA IN² | PERIMETER OF DUCT IN. | THICKNESS OF DUCT IN. | $f_t$ @ BURST PRESS F/IN² | $f_t$ @ FOR 3600 HR OPERATION #/IN² |
|---|---|---|---|---|---|---|
| ① | 5.13 | 19.39 | 15.84 | .007 | 29300 | 12850 |
| ② | 6.46 | 31.85 | 20.14 | MINIMUM THICKNESS MANUFACTURING REQUIREMENT | 36800 | 16150 |
| ③ | 6.72 | 25.80 | 20.3 | | 38300 | 16800 |
| ④ | 6.81 | 23.77 | 19.3 | | 39800 | 17050 |
| ⑤ | 8.26 | 45.95 | 24.9 | | 47000 | 20700 |
| ⑥ | 9.30 | 67.93 | 29.22 | .007 | 53000 | 23700 |

$$f_t \text{ @ BURST PRESS.} = \frac{P \times 1.33 \times 1.5\ R}{t} = \frac{39.6\ PSI \times 1.33 \times 1.5 R}{.007} = 11.38 R \times 10^3$$

$$f_t \text{ FOR 3600 HOUR (.2\% CREEP CRITERIA)} = \frac{PR}{t} = \frac{35\ PSI}{.007} = 5.00 R \times 10^3$$

## HOT GAS DUCTING

BLADE DUCTS MUST WITHSTAND AN INTERNAL 5 PSI NEGATIVE PRESSURE. THEREFORE EXTERNAL STIFFENING RINGS MUST BE ADDED.

$$p' = .807 \frac{Et^2}{lr} \sqrt[4]{\left(\frac{1}{1-\nu^2}\right)^3 \frac{t^2}{r^2}}$$

REF.- FORMULAS FOR STRESS AND STRAIN BY ROARK TABLE VII - Q - 31

$p'$ = 5 PSI = ELASTIC BUCKLING LIMIT PRESS.
$l$ = RING STIFFENER SPACING - INCHES

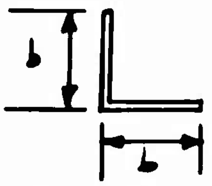
| DUCT → | ① | ② | ③ | ④ | ⑤ | ⑥ |
|---|---|---|---|---|---|---|
| $l$ → | 4.4 | 3.1 | 2.9 | 2.9 | 2.25 | 1.80 |

EXAMPLE OF RING STIFFENER REQUIRED FOR DUCT ①

$$p' = \frac{3EI}{r^3} \qquad p' = 5 \text{psi} \times l = 2.20 \text{ \#/in}$$

$$I = \frac{(2.20)(2.57)^3}{3(25.3\times10^6)} = .493\times10^{-6}$$

REF - FORMULAS FOR STRESS AND STRAIN BY ROARK TABLE XV - 12



RING STIFFENER CROSS SECTION

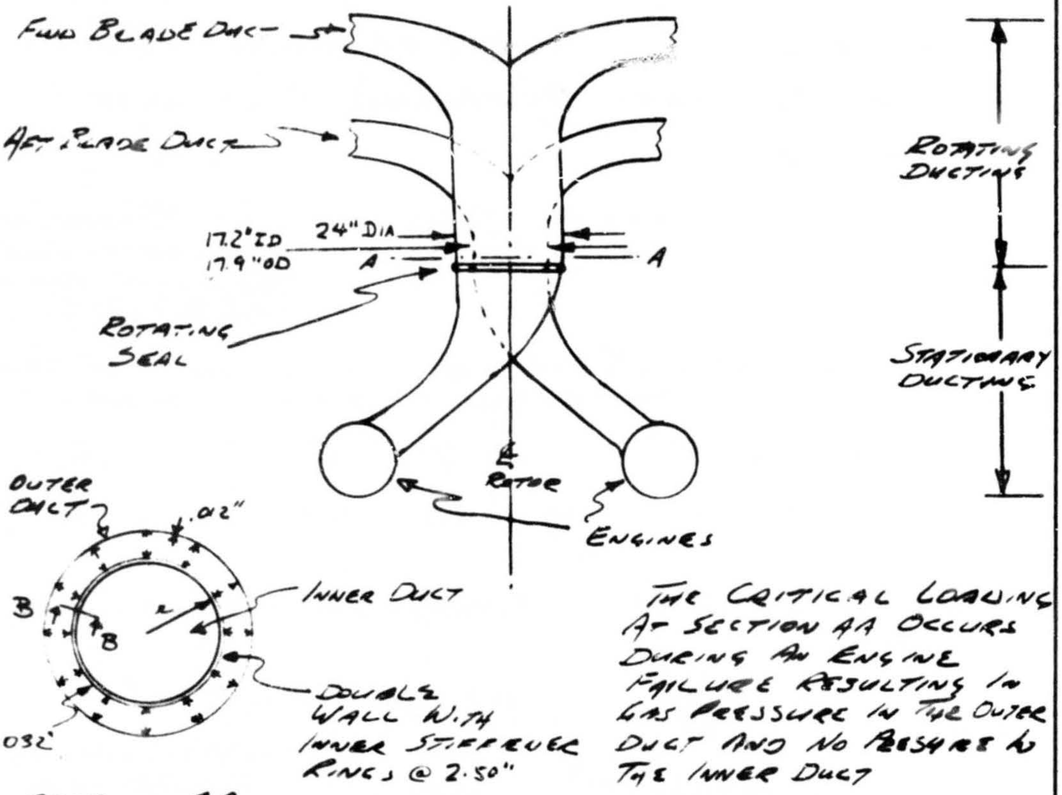
$$I = \frac{5tb^3}{24} \qquad b = \left(\frac{24I}{5t}\right)^{1/3}$$

$$b = .07''$$

## HOT GAS DUCTING THROUGH HUB AREA

REF- DWG 395-0925

DUCT MATERIAL - RENE 41



THE CRITICAL LOADING AT SECTION AA OCCURS DURING AN ENGINE FAILURE RESULTING IN GAS PRESSURE IN THE OUTER DUCT AND NO PRESSURE IN THE INNER DUCT

SECTION AA

| CONDITION ⟶ | PEAK | LONG TIME CONDITIONS |
|---|---|---|
| PRESSURE #/in² | 39.6 | 35.0 |
| DUCT WALL TEMPERATURE °F | 1390 | 1260. |

OUTER WALL OF INNER DUCT = .032" AS DETERMINED BY STABILITY FROM "FORMULAS FOR STRESS AND STRAIN" BY ROARK TABLE XIII - Q - 31

$$p' = .607 \frac{Et^2}{lr} \sqrt[4]{\left(\frac{1}{1-\nu^2}\right)^3 \frac{t^2}{r^2}}$$

INNER SKIN OF INNER DUCT = .008"

$$f_{t_{PEAK}} = \frac{39.6\text{psi} \times .33 \times 1.5 \times 8.6"}{.008} = 85000\text{ PSI}$$

ALLOWABLE = 150000 psi

$$f_{t_{LONG\ TIME}} = \frac{35 \times 8.6}{.008} = 37500\text{ PSI}$$

$$M.S. = \frac{38000}{37500} - 1 = +.01$$

9704

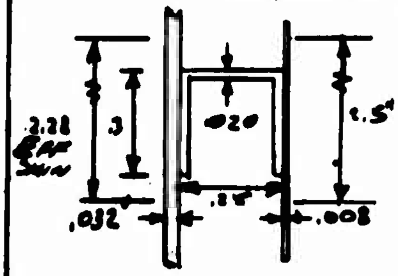
## HOT GAS DUCTING THROUGH HUB AREA

STIFFENER RING REQUIRED IN INNER DUCT

$I_{REQ}$ FOR STABILITY $= \frac{p'r^3}{3E}$ REF "FORMULAS FOR STRESS AND STRAIN" BY ROARK TABLE XV-12

$I = \frac{99.0(8.9)^3}{3(25.3\times10^6)} = .912\times10^{-3}$   $p' = 39.6\frac{\#}{in^2}\times2\frac{1}{2}'' = 99\frac{\#}{in}$

$I_{AVAILABLE} = .92\times10^{-3}$



SECTION B-B

CROSS SECTION OF INNER STIFFENER BOX

NOTE - SANDWICH CONSTRUCTION WOULD SHOW A LIGHTER WEIGHT FOR THE INNER DOUBLE WALL.

OUTER DUCT WALL THICKNESS

$t = .012''$ RENE 41

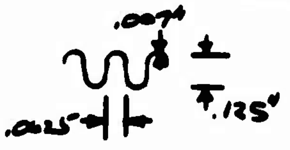
$f_t$ PEAK $= \frac{39.6 PSI \times 1.33 \times 1.5 \times 12''}{.012}$

$= 79000$ PSI

$f_t$ LONG TIME $= \frac{35 PSI \times 12''}{.012} = 35000$ PSI

$M.S = \frac{38000}{35000} - 1 = +.08$

DUCT BELLOWS REF. DWG 395-0909



TYPICAL BELLOWS CROSS SECTION

MATERIAL RENÉ 41

$\alpha = 8.2\times10^{-6}\frac{IN}{IN}$ °F   $E = 25.3\times10^6$ PSI

DUCT WALL LONG TIME TEMP. = 1260°F

DUCT LENGTH = 20"

$\Delta = \Delta t(l)\alpha = (1260-70)(20)8.2\times10^{-6} = .1950''$

USE .040 t .050 FOR BEAM STRUCT

NEGLECT AS THIS RELIEVES THERMAL STRAIN

$I = 2.66\times10^{-9}$   LENGTH OF BELLOWS $= 2\frac{3}{8}''$

$\Delta = \frac{P(\frac{l}{2})^3}{12EI} + \frac{P(\frac{l}{2})\frac{l^2}{16}}{2EI}$

$\Delta_{BELLOWS} = \frac{.1950}{2\frac{3}{8}''/\frac{1}{16}} = .00514$

$.0051 = 1.582\times10^{-3}P$

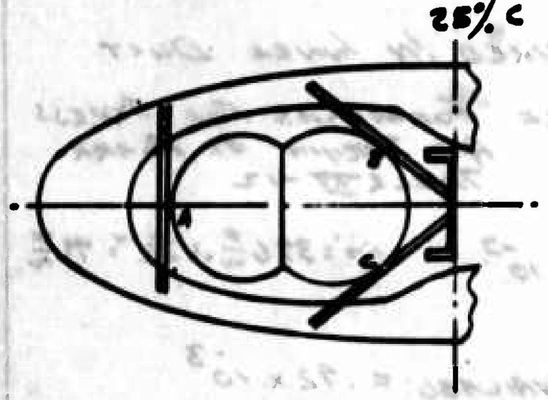
$P = 3.2\#$

$M = \frac{Pl}{2} = .200$ in#   $\frac{M}{S} + \frac{P}{A} = \frac{.200}{8.18\times10^{-6}} + \frac{3.2}{.007} = 25000$ psi

M.S. = [illegible]500 / 25000 − 1 = +.[illegible]

## GAS DUCT – BLADE MOUNTING

REF. DWG 395-0919



WEIGHT OF DUCT

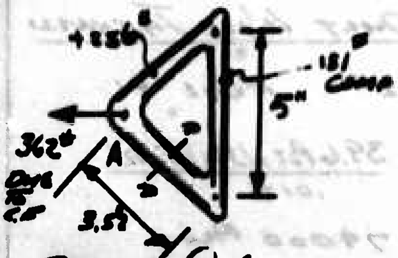
$(15.84+20.14)(.007)(20)(.298\ \#/\text{IN}^3)$ — 35.98

$= 1.49^{\#}$

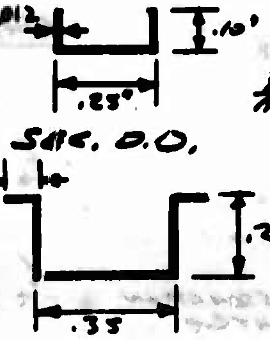
N @ TIP @ 840 F.P.S. TIP SPEED

$= 486$ g's OVER-REV CONDITION

LOAD AT ATTACHMENT A $= 1.49^{\#}(486)\frac{1}{2} = 362^{\#}$ LIMIT



BRACKET @ A
MATERIAL INCONEL 718



AREA = .0054 IN²

$\frac{P}{A} = \frac{256^{\#} \times 1.15}{.0054} = 71000$ PSI

AREA = .011 IN²

$\frac{P}{A} = \frac{181^{\#} \times 1.5}{.011 \text{ IN}^2} = 25000$ PSI

$L/\rho$, $\frac{L}{\rho} = 33$

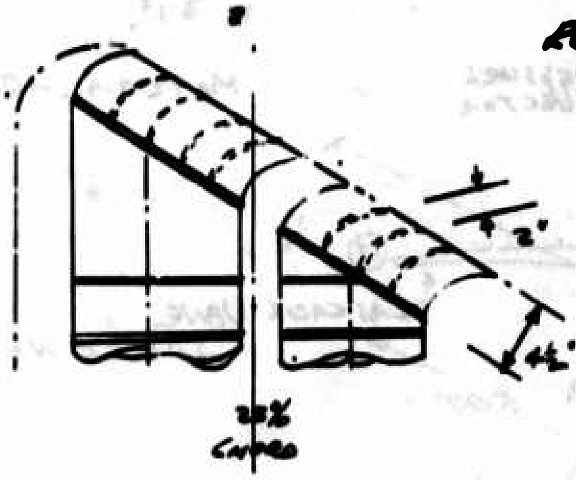
$M.S. = \frac{37000}{25000} - 1 = +.48$

LOAD AT ATTACHMENT B OR C $= 1.49^{\#}\left(\frac{486 \text{ g's}}{2}\right)\frac{1}{2} = 181^{\#}$ LIMIT

NO ANALYSIS SHOWN AS LOADS ARE LESS CRITICAL

## TIP CASCADE

REF-DWG 395-0909

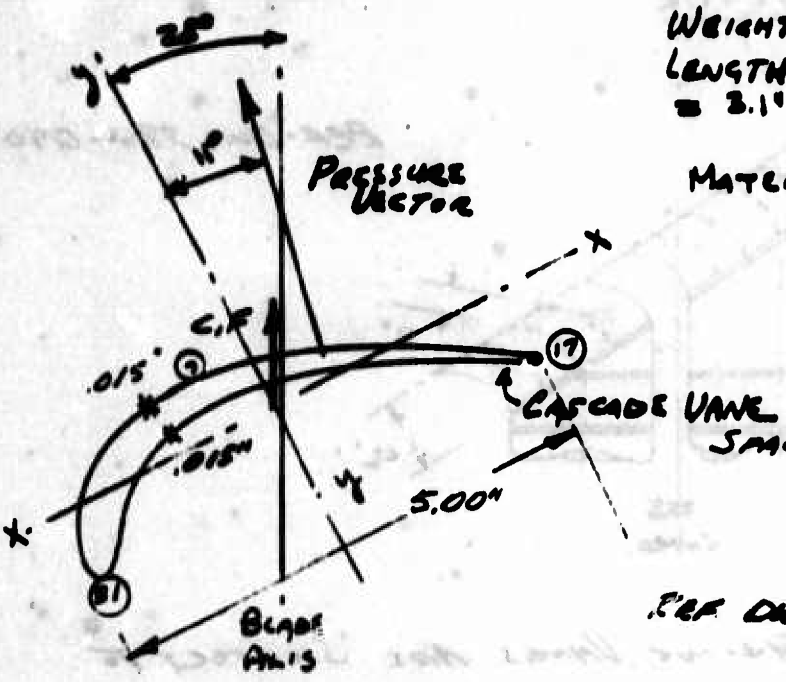


THE TURNING VANES ARE SUBJECT TO GAS PRESSURE AND CENTRIFUGAL LOAD. SINCE THE ALLOWABLE STRESS FOR PEAK CONDITIONS IS 150,000 PSI VS. 38,000 PSI FOR 3600 HOUR OPERATION. THE ANALYSIS IS MADE FOR LONG TIME OPERATION ONLY.

## TIP CASCADE

WEIGHT OF VANE = .30#
LENGTH OF VANE
= 3.1"

MATERIAL - RENE 41



REF DWG. 395-0909

CHECK FOR LONG TIME CONDITIONS

$P = 35$ psi $\quad \eta = 315g$ @ 675 F.P.S.

$C.F. = .30^{\#} \times 315 = 94.5^{\#}$/VANE

AXIAL LOAD ON VANE = 35 psi/2" × 5" = 350#

$$f_7 = \frac{M_{xx}}{Z_{xx}} + \frac{P}{A} = \overset{12800}{\frac{174}{.0137}} + \overset{1170}{\frac{350}{.30}} = 13970 \text{ psi}$$

$$f_{31} = \frac{M_{xx}}{Z_{xx}} + \frac{M_{yy}}{Z_{yy}} + \frac{P}{A} = \overset{15800}{\frac{174}{.011}} + \overset{1050}{\frac{51}{.050}} + \overset{1170}{\frac{350}{.30}} = 18020 \text{ psi}$$

$$f_{17} = \frac{M_{xx}}{Z_{xx}} - \frac{M_{yy}}{Z_{yy}} + \frac{P}{A} = \overset{15800}{\frac{174}{.011}} - \overset{1050}{\frac{51}{.050}} + \overset{1170}{\frac{350}{.30}} = 15920$$

$F_{.2\%}$ CREEP = 38000 psi ALLOW
FOR 3000 HOURS
AT 1200°

$$M.S. = \frac{38000}{18020} - 1 = +1.10$$

9704

# APPENDIX III

# ROTOR BLADE EQUATIONS

## NON-BENDING WEIGHT EQUATIONS - TIP SECTION

(53) $\rho_{NB_t} = D_1 \left(\frac{\bar{\rho}_R}{.528}\right)^{m_1} \left(\frac{t_t}{.18}\right)^{m_8}$ $\qquad \bar{C} = .830\left(\frac{\sigma R}{b}\right)$

$A_D = D_2 \bar{C}^{m_2} \left(\frac{\bar{\rho}_R}{.528}\right)^{m_3} \left(\frac{t_t}{.18}\right)^{m_4}$

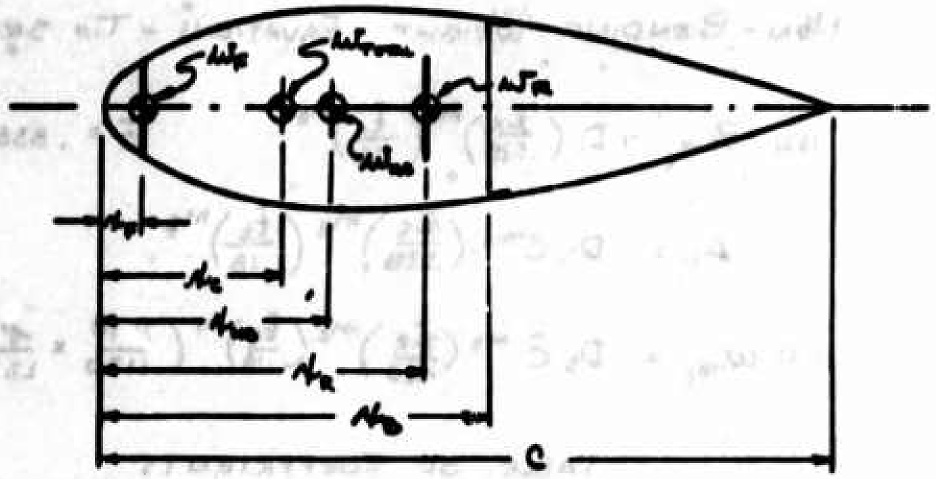
(54) $W_{NB_t} = D_3 \bar{C}^{m_5} \left(\frac{\bar{\rho}_R}{.528}\right)^{m_6} \left(\frac{t_t}{.18}\right)^{m_7} \left(\frac{T-70}{1180} \times \frac{\psi}{27.2}\right)^{m_9}$

### TABLE OF COEFFICIENTS

| BLADE / COEFF | XV-9A DUCTS | FIG. "B" TWO SPAR | FIG. "B" ONE SPAR |
|---|---|---|---|
| $D_1$ | .265 | .236 | .422 |
| $D_2$ | 106 | 98 | 215* 200** |
| $D_3$ | 1.26 | 1.43 | 1.38 |
| $m_1$ | -.31 | 0 | .46 |
| $m_2$ | 2.1 | 2.0 | 2.0 |
| $m_3$ | .85 | 0 | 1.20 |
| $m_4$ | .95 | 1.0 | 1.69* 1.0** |
| $m_5$ | 1.2 | 1.2 | 1.2 |
| $m_6$ | 1.34 | 0 | .47 |
| $m_7$ | .17 | .35 | .35 |
| $m_8$ | 0 | 0 | 0 |
| $m_9$ | .2 | 0 | 0 |

* $.14 \leq t \leq .16$ $\qquad$ ** $.16 \leq t \leq .18$

## BLADE BALANCE EQUATIONS



$$w_F x_F + w_R x_R + w_{NB} x_{NB} = w_T x_c$$

$$w_F (x_F - x_c) + w_R (x_R - x_c) + w_{NB} (x_{NB} - x_c) = 0$$

$$w_R = - w_F \left( \frac{x_F - x_c}{x_R - x_c} \right) - w_{NB} \left( \frac{x_{NB} - x_c}{x_R - x_c} \right)$$

$$K_2 = \frac{x_c - x_F}{x_R - x_c} \qquad K_3 = w_{NB} \left( \frac{x_{NB} - x_c}{x_R - x_c} \right)$$

$$w_R = K_2 w_F - K_3$$

$$w_F = \frac{w_R + K_3}{K_2}$$

$$w_T = w_F + w_R + w_{NB} = w_F (1 + K_2) - K_3 + w_{NB}$$

$$= w_R \left( \frac{1 + K_2}{K_2} \right) + \frac{K_3}{K_2} + w_{NB}$$

$$(55)\quad \omega_{R_t} = \frac{S}{F_{TU}}\left\{1.10\,V_T^2\left[\bar{W}_{CA}\left(\frac{\bar{x}_{CA}-\bar{x}_F}{\bar{x}_R-\bar{x}_F}\right)+\bar{W}_t\left(\frac{\bar{x}_t-\bar{x}_F}{\bar{x}_R-\bar{x}_F}\right)\right]+p_0A_0\right\}$$

$$K_{2_t} = \frac{\bar{x}_{c_t}-\bar{x}_F}{\bar{x}_R-\bar{x}_{c_t}} \qquad K_{4_t} = 1+K_{2_t}$$

$$K_{3_t} = \omega_{NB_t}\frac{\left(\bar{x}_{NB_t}-\bar{x}_{c_t}\right)}{\left(\bar{x}_R-\bar{x}_{c_t}\right)}$$

$$(56)\quad \omega_{F_t} = \frac{\omega_{R_t}+K_{3_t}}{K_{2_t}}$$

$$\omega_{T_t} = \omega_{R_t}+\omega_{F_t}+\omega_{NB_t}$$

$$\bar{x}_{SK} = \frac{\bar{x}_B+\bar{x}_F}{2} \qquad \bar{x}'_{SK} = \bar{x}_B$$

$$\Delta\omega_{SK_2} = .257\,C\left(\bar{t}_{SK_2}-\bar{t}_{SK_t}\right)\left(\bar{x}_B-\bar{x}_F\right) \qquad \Delta\omega'_{SK_2} = .022\,C\left(\bar{t}_{SK_2}-\bar{t}_{SK_t}\right)$$

$$(57)\quad \omega_{NB_2} = \omega_{NB_t}\left(\frac{\bar{t}_2}{\bar{t}_t}\right)^{m_7}+\Delta\omega_{SK_2}+\Delta\omega'_{SK_2}$$

$$(58)\quad \bar{x}_{NB_2} = \frac{\omega_{NB_t}\,\bar{x}_{NB_t}\left(\frac{\bar{t}_2}{\bar{t}_t}\right)^{m_7+m_8}+\Delta\omega_{SK_2}\,\bar{x}_{SK}+\Delta\omega'_{SK_2}\,\bar{x}'_{SK}}{\omega_{NB_2}}$$

SPARS AT STATION 2 ARE DESIGNED BY FATIGUE LOADS. AN INITIAL ESTIMATE FOR DETERMINING CORIOLUS MOMENT COMPONENT IS MADE AS FOLLOWS:

$$(w_{T_2})_6 = 3.275\ \delta\ c\ \bar{t}_R + w_{F_t} + w_{NB_t}$$

$$(\bar{M}_2)_6 = \bar{d}_2^{\,2}\left[\frac{w_{T_t}}{2} + \frac{(w_{T_2})_6}{6}\right] + \frac{\bar{W}_t}{N'_t}\left(N'_t - \bar{l}_2\right)$$

$$(\bar{W}_{B_2})_6 = \frac{\bar{d}_2}{2}\left[w_{T_t} + (w_{T_2})_6\right] + \frac{\bar{W}_t}{N'_t}$$

$$(\bar{M}_{t_2})_6 = (\bar{M}_2)_6 + (\bar{W}_{B_2})_6\left(\bar{l}_2 + \bar{e}_t\right)$$

$$(I_{t_2})_6 = \frac{1}{3}\left\{w_{T_t}\left[1-(\bar{l}_2+\bar{e}_t)^3\right] + \left(\frac{(w_{T_2})_6 - w_{T_t}}{4\,\bar{d}_2}\right)\left[\left[1-(\bar{l}_2-\bar{e}_t)^3\right.\right.\right.$$

$$\left.\left.\left.\left[4-3(\bar{l}_2+\bar{e}_t)\right]\right]\right] + 3\bar{W}_t N'_t\right\}$$

$$(M_{COR_2})_6 = \frac{8\lambda\left(N_n - \bar{e}_t\right)\left(\frac{W}{b}\right)\left(\frac{\bar{l}}{\bar{e}_t}\right)(a_\cdot)}{(w_{T_2})_6\left(\frac{V_T^2}{2.68}\right)(1+\bar{e}_t)(\bar{L})}\left\{\frac{(\bar{I}_{t_2})_6 - (\bar{l}_2+2\bar{e}_t)(\bar{M}_{t_2})_6 + (\bar{W}_{B_2})_6(\bar{l}_2+\bar{e}_t)\bar{e}_t}{(w_{T_2})_6(\bar{L})^2\left(\frac{2+\bar{e}_t}{6}\right)}\right\}$$

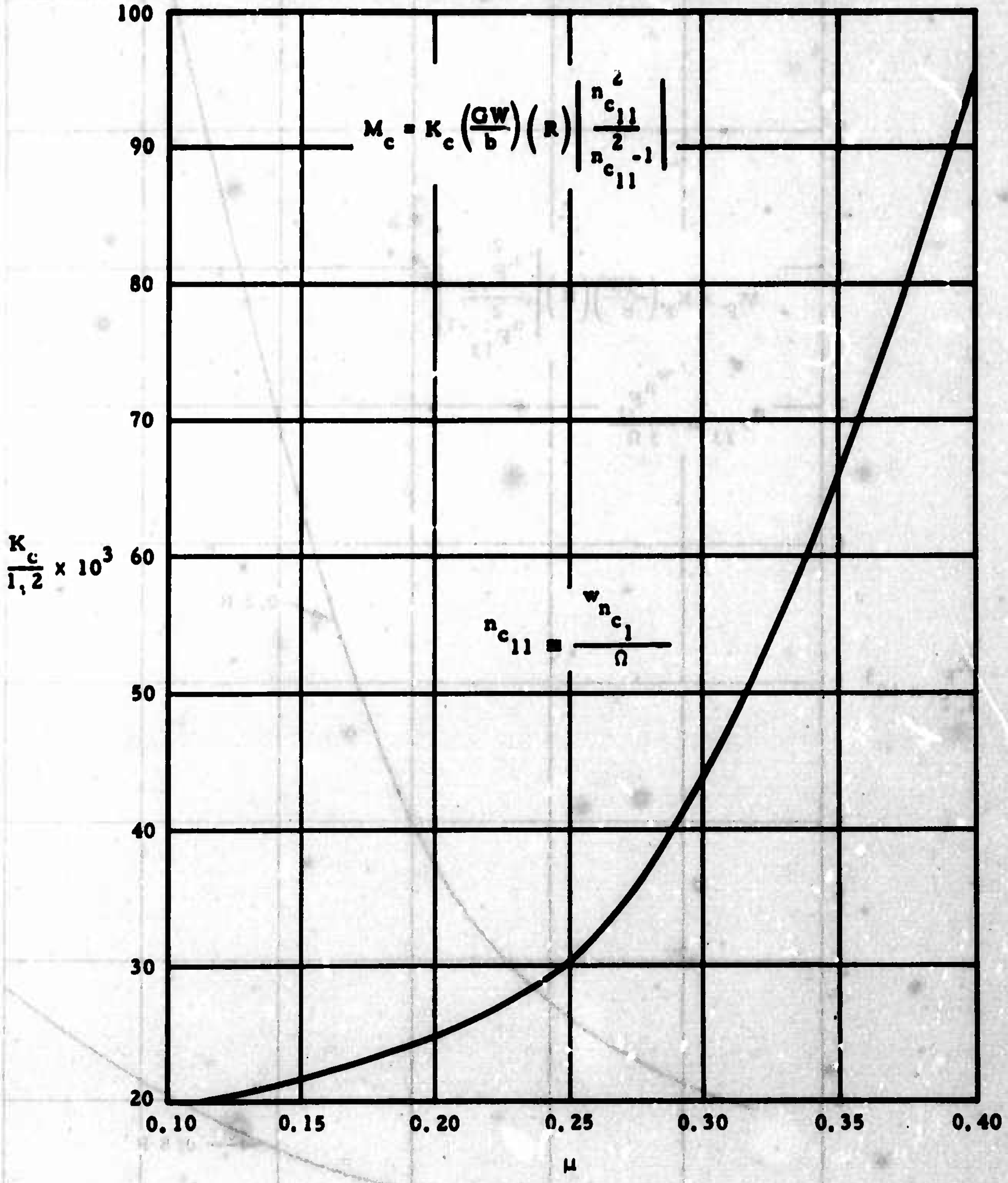
FIRST APPROXIMATION

(59)

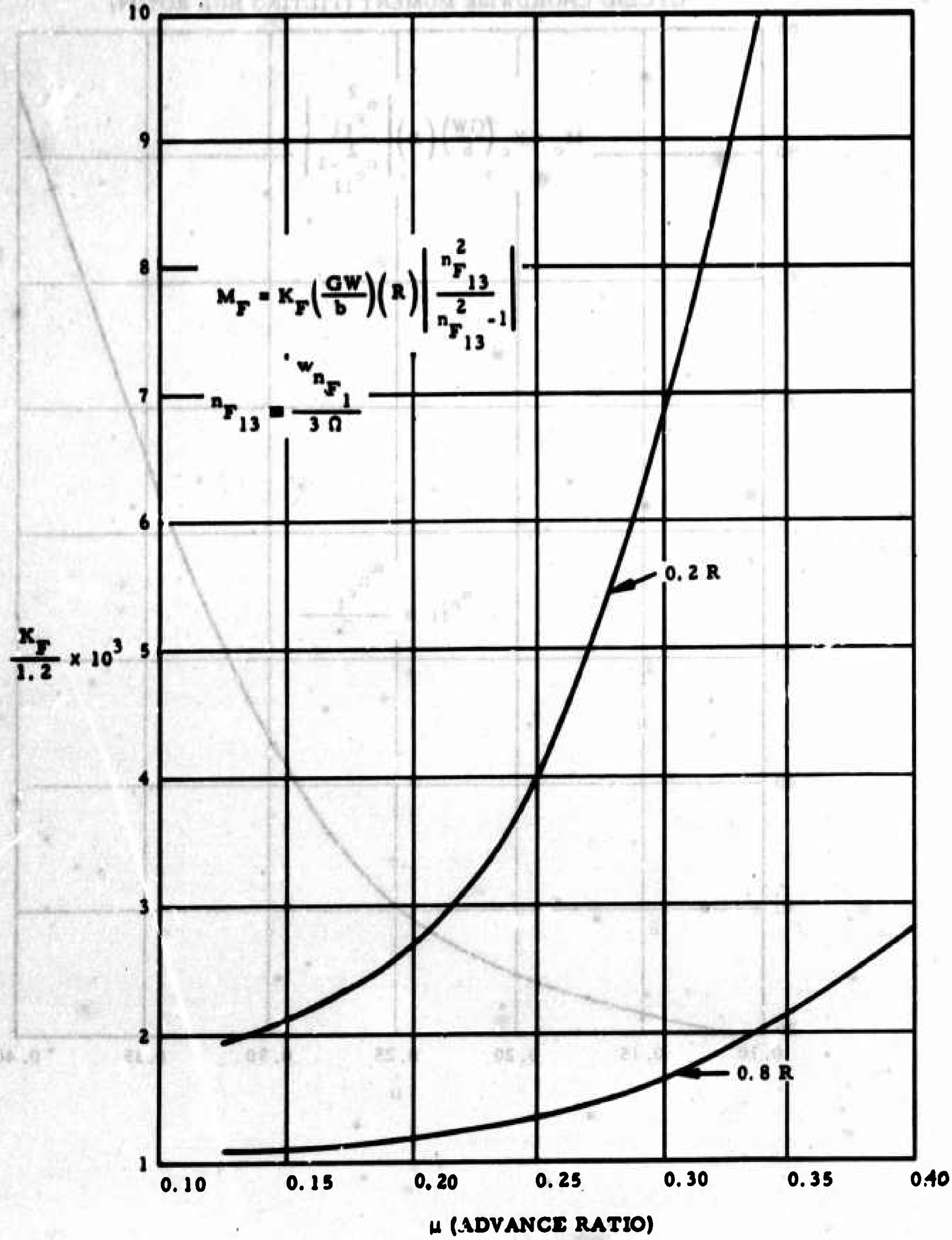
$$\left.\begin{array}{l} M_{C_2} = \left[K_{C_2} + (\bar{M}_{COR_2})_6\right]\left(\frac{W}{b}\right)(R)(29.4) \\ M_{F_2} = K_{F_2}\left(\frac{W}{b}\right)(R)(36.0) \end{array}\right\}\text{(TILTING HUB ROTOR)}$$

$$\left.\begin{array}{l} M_{F_2} = K_{F_2}\,\dfrac{\left(\frac{c}{\delta}\right)\left(\frac{\sigma}{b}\right)^2\left(\bar{\rho}_R\right)^2}{39.1}\left(\dfrac{w_{F_t}K_{4_2} - K_{3_2}}{w_{T_t}}\right)\left(\frac{W}{b}\right)(R)(36.0) \\ M_{C_2} = \dfrac{1090}{W}\,R^{6.4}\left(\frac{\sigma}{b}\right)^{2.4}\left(\dfrac{d_2}{d_1+d_2}\right) \end{array}\right\}\text{(ARTICULATED ROTOR)}$$

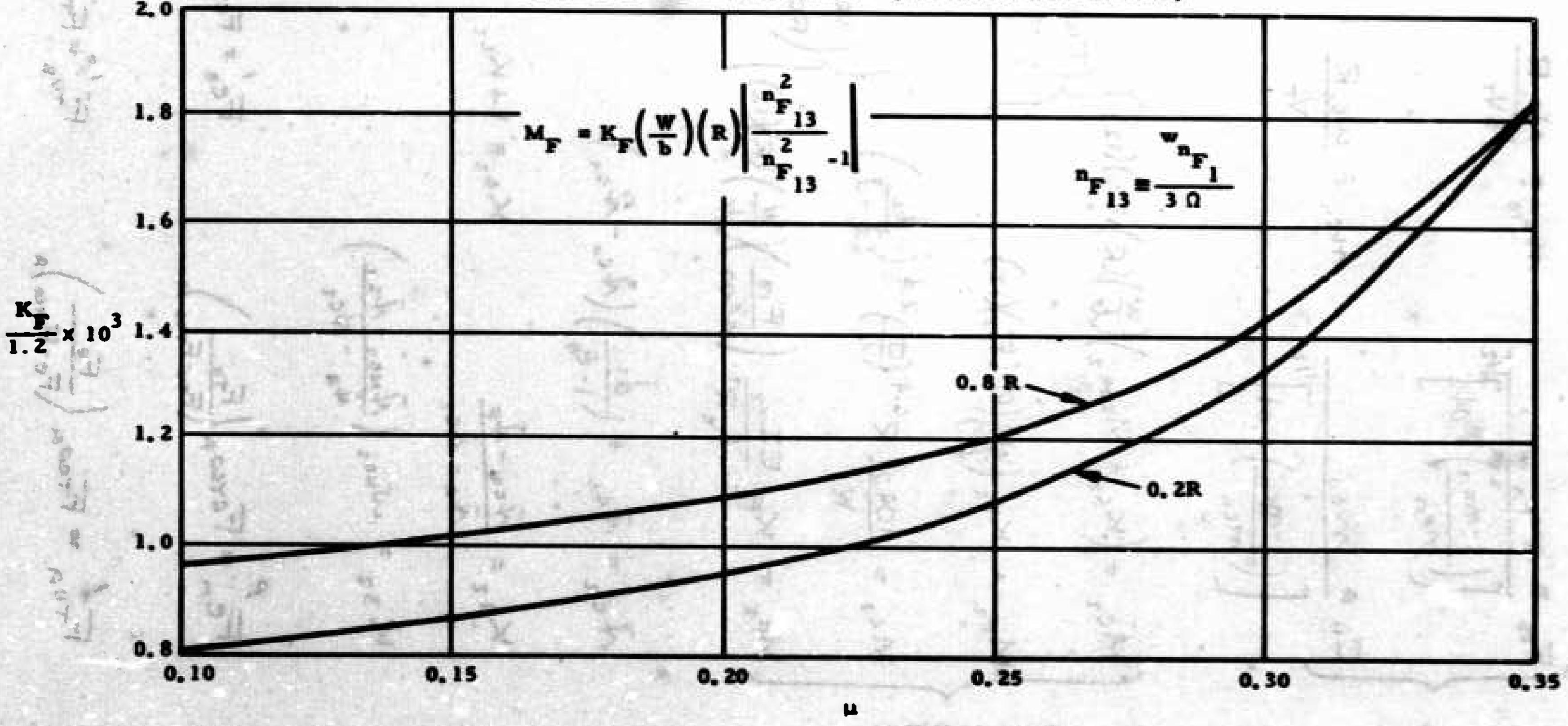
## CYCLIC CHORDWISE MOMENT (TILTING HUB ROTOR)



CYCLIC FLAPWISE MOMENT (TILTING HUB ROTOR)



## CYCLIC FLAPWISE MOMENT (ARTICULATED ROTOR)



(59a)
$$\begin{cases} F_{13} = \dfrac{\omega_{F_{13}}}{\left[\left(\dfrac{1-\omega_{F_{13}}^2}{\omega_{F_{13}}}\right)^2 + .04\right]^{1/2}} & \omega_{F_{13}} = \dfrac{\omega_{F_1} R}{3 V_T} \\ C_{11} = \dfrac{\omega_{c_{11}}}{\left[\left(\dfrac{1-\omega_{c_{11}}^2}{\omega_{c_{11}}}\right)^2 + .04\right]^{1/2}} & \omega_{c_{11}} = \dfrac{\omega_{c_1} R}{V_T} \end{cases}$$

$$c/c_c = \xi = .10$$

(59b) ITERATIONS
$$\begin{cases} M_{c_2} = \left(K_{c_2} + M_{cor_2}\right)\left(\dfrac{W}{b}\right)(R)(C_{11})(12) \\ M_{F_2} = K_{F_2}\left(\dfrac{W}{b}\right)(R)(F_{13})(12) \end{cases} \left(\begin{matrix}\text{TILT HUB} \\ \text{ROTOR}\end{matrix}\right)$$

$$\begin{cases} M_{c_2} = \dfrac{1090}{W} R^{6.4}\left(\dfrac{\sigma}{b}\right)^{2.4}\left(\dfrac{\bar{d}_2}{\bar{d}_1 + \bar{d}_2}\right) \\ M_{F_2} = K_{F_2} \dfrac{EI_{x_2}}{\bar{\omega}_{T_2} R^2}\left(\dfrac{F_{13}}{4500}\right)\left(\dfrac{W}{b}\right)(R)(12) \end{cases} \left(\begin{matrix}\text{ARTICULATED} \\ \text{ROTOR}\end{matrix}\right)$$

$$\bar{r}_{c_2} = \bar{r}_{c_4} + \left(\frac{\bar{d}_2}{1-\bar{\xi}_4}\right)\left(\bar{r}_{c_0} - \bar{r}_{c_4}\right)$$

$$K_{2_2} = \frac{\bar{r}_{c_2} - \bar{r}_F}{\bar{r}_R - \bar{r}_{c_2}} \qquad K_{4_2} = 1 + K_{2_2}$$

$$K_{3_2} = \bar{\omega}_{HB_2}\left(\frac{\bar{r}_{HB_2} - \bar{r}_{c_2}}{\bar{r}_R - \bar{r}_{c_2}}\right)$$

$$F_{e_A}' = F_{eyco_A}\left(\frac{F_{TU}}{F_{TU} - F_{TCO}}\right)_A \qquad F_{e_B}' = F_{eyco_B}\left(\frac{F_{TU}}{F_{TU} - F_{TCO}}\right)_B$$

$$F_{TU_A}' = F_{TCO_A}\left(\frac{F_e}{F_e - F_{eyco}}\right)_A \qquad F_{TU_B}' = F_{TCO_{B}\,95}\left(\frac{F_e}{F_e - F_{eyco}}\right)_B$$

$$a_1 = \frac{\delta V_T^2}{2.68}\left\{(\bar{d}_2)\left[w_{t_2}\left(\frac{1}{2}-\frac{\bar{d}_2}{6}\right)+\left(w_{w_2}-K_{3_2}\right)\left(\frac{1}{2}-\frac{\bar{d}_2}{3}\right)\right]+\bar{W}_4\right\}$$

$$a_2 = \frac{\delta V_T^2}{2.68}\,\bar{d}_2\,K_{4_2}\left(\frac{1}{2}-\frac{\bar{d}_2}{3}\right)$$

$$a_3 = -K_{3_2}$$

$$a_4 = K_{4_2}$$

$$a_5 = M_{F_2}\,\bar{\dot{x}}_F\left(\frac{\delta}{2}\right)$$

$$a_6 = a_3\,\bar{\rho}_R^2\,\bar{c}\,(45)$$

$$a_7 = 45\left(\bar{\rho}_F^2+K_{2_2}\,\bar{\rho}_R^2\right)(\bar{c})$$

$$a_8 = \frac{M_{c_2}\,\delta}{45\,\bar{c}\,(\lambda_R-\lambda_F)}$$

$$a_9 = a_5\left(\frac{\dot{x}_R}{\bar{x}_F}\right)$$

$$a_{10} = K_{2_2}$$

$$A_1 = k_0 a_4 a_7 - k_2 a_7$$

$$A_2 = k_0(a_3 a_7 + a_6 a_4) - k_1 a_7 - k_3 a_6 - a_4 a_8 - a_4 a_7 a_9$$

$$A_3 = k_0 a_3 a_6 - k_1 a_6 - a_3 a_8 - a_9(a_3 a_7 + a_4 a_6)$$

$$A_4 = - a_3 a_6 a_9$$

$$A_5 = k_0 a_{10} a_4 a_7 - k_2 a_7 a_{10} = A_1 a_{10}$$

$$A_6 = k_0(a_3 a_4 a_7 + a_3 a_7 a_{10} + a_4 a_6 a_{10}) - k_1 a_7 a_{10} - k_2 a_6 a_{10} - k_2 a_3 a_7$$
$$- a_4 a_9 a_{10} - a_4 a_7 a_8$$

$$A_7 = k_0 a_3(a_6 a_{10} + a_3 a_7 + a_4 a_6) - k_1 a_6 a_{10} - k_1 \check{a}_3 a_7 - k_3 \check{a}_3 a_6$$
$$- a_9(\acute{a}_3 a_{10} + a_3 \check{a}_4) - a_8(a_3 a_7 + a_4 a_6)$$

$$A_8 = k_0 a_3^2 a_6 - k_1 a_3 a_6 - a_3^2 a_9 - a_3 a_6 a_8 = a_3^2(k_0 a_6 - a_9) - a_3 a_6(k_1 + a_8)$$

$$A_9 = k_3 a_4 - k_5$$

$$A_{10} = k_3 a_3 - k_4 - a_4 a_8$$

$$A_{11} = - a_3 a_8$$

$$A_{12} = k_3 a_4 \dot{a}_{10} - k_5 a_{10}$$

$$A_{13} = k_3 a_3 (a_4 + a_{10}) - k_4 a_{10} - k_5 a_3 - a_4 a_8$$

$$A_{14} = k_3 a_3^2 - k_4 a_3 - a_3 a_8$$

$$\begin{aligned}
k_0 &= Fe_A & k_0' &= Fe_A' \\
k_1 &= a_1 \frac{Fe_A}{F'_{TUA}} & k_1' &= a_1 \frac{Fe'_A}{F_{TUA}} \\
k_2 &= a_2 \frac{Fe_A}{F'_{TUA}} & k_2' &= a_2 \frac{Fe'_A}{F_{TUA}} \\
k_3 &= Fe_B & k_3' &= Fe'_B \\
k_4 &= a_1 \frac{Fe_B}{F'_{TUB}} & k_4' &= a_1 \frac{Fe'_B}{F_{TUB}} \\
k_5 &= a_2 \frac{Fe_B}{F'_{TUB}} & k_5' &= a_2 \frac{Fe'_B}{F_{TUB}}
\end{aligned}$$

$$(60)\quad \left\{\begin{aligned}
&▷1\ (w_{F_2})_a^3 (A_1) + (w_{F_2})_a^2 (A_2) + (w_{F_2})_a (A_3) + A_4 = 0 \\
&(w'_{F_2})_a^3 (A'_1) + (w'_{F_2})_a^2 (A'_2) + (w_{F_2}')_a (A'_3) + A'_4 = 0 \\
&▷1\ (w_{F_2})_b^3 (A_5) + (w_{F_2})_b^2 (A_6) + (w_{F_2})_b (A_7) + A_8 = 0 \\
&(w'_{F_2})_b^3 (A'_5) + (w'_{F_2})_b^2 (A'_6) + (w'_{F_2})_b (A'_7) + A'_8 = 0 \\
&▷1\ (w_{F_2})_c^2 (A_9) + (w_{F_2})_c (A_{10}) + A_{11} = 0 \\
&(w'_{F_2})_c^2 (A'_9) + (w'_{F_2})_c (A'_{10}) + A'_{11} = 0 \\
&▷1\ (w_{F_2})_d^2 (A_{12}) + (w_{F_2})_d (A_{13}) + A_{14} = 0 \\
&(w'_{F_2})_d^2 (A'_{12}) + (w'_{F_2})_d (A'_{13}) + A'_{14} = 0
\end{aligned}\right.$$

$$f_{cF_2} = \frac{a_1 + a_2 w_{F_2}}{a_3 + a_4 w_{F_2}}$$

▷1 If $f_{cF_2} > F_{TCO}$, use primed equations.
Use largest value of $(w_{F_2})_{a,b,c,d}$, $(w'_{F_2})_{a,b,c,d}$, $w_{F_2}$

(61)
$$w_{T_2} = K_{4_2} w_{F_2} - K_{3_2} + w_{NB_\ell}$$

$$w'_{T_2} = K_{4_2} w_{F_2} - K_{3_2} + w_{NB_2}$$

$$\Delta w_{SK_\ell} = .287\ \bar{c}\left(\bar{t}_{SK_\ell} - \bar{t}_{SK_2}\right)\left(\bar{A}_B - \bar{A}_F\right)$$

$$\Delta w'_{SK_\ell} = .022\ \bar{c}\left(\bar{t}_{SK_\ell} - \bar{t}_{SK_2}\right)\left(\frac{\bar{t}_\ell}{.18}\right)$$

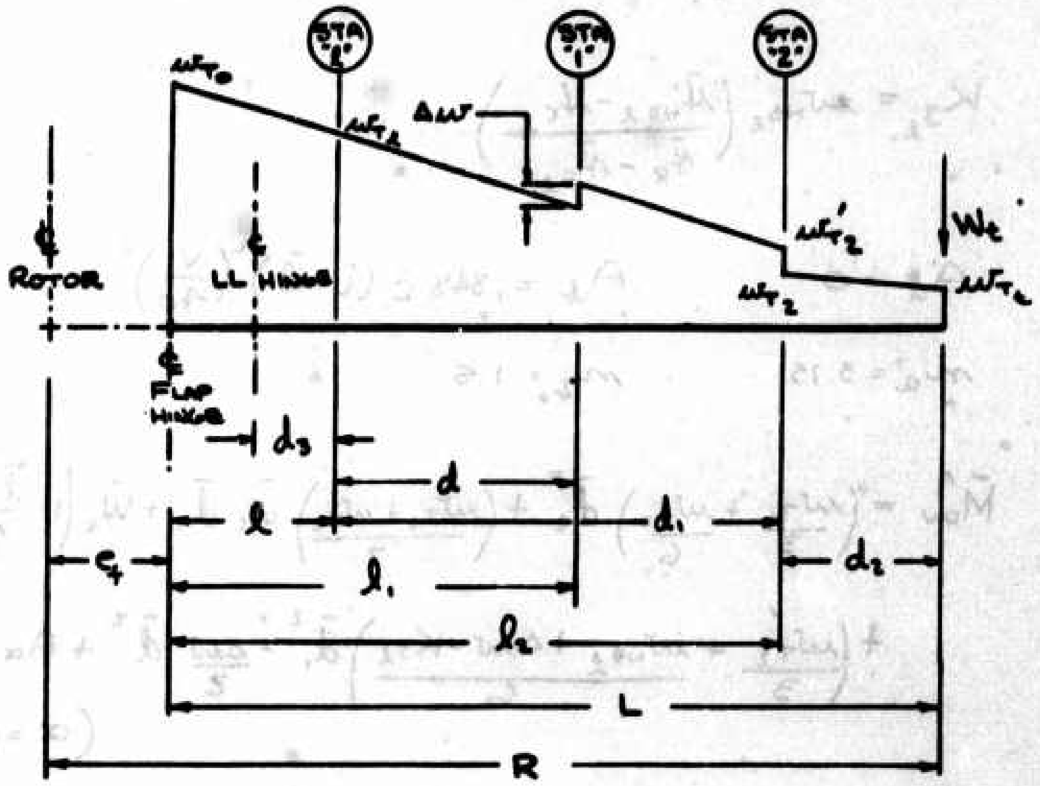
(62) ROUND T.E.
$$\begin{cases} w_{NB_\ell} = D_4\,\bar{c}^{\,m_5}\left(\dfrac{\bar{A}_R}{.528}\right)^{m_{10}}\left(\dfrac{\bar{t}_\ell}{.15}\right)^{m_{11}}\left(\dfrac{T-70}{1130} \times \dfrac{P}{27.2}\right)^{m_9} + \Delta w_{SK} + \Delta w_{SK_\ell} \\ \bar{A}_{NB_\ell} = \dfrac{\left(w_{NB_\ell} - \Delta w_{SK_\ell} - \Delta w'_{SK_\ell}\right)(D_5)\left(\dfrac{\bar{A}_R}{.528}\right)^{m_{12}} + \Delta w_{SK_\ell}\bar{A}_{SK} + \Delta w'_{SK_\ell}\bar{A}'_{SK}}{w_{NB_\ell}} \end{cases}$$

(63) NORMAL T.E.
$$\begin{cases} w_{NB_\ell} = w_{NB_2} + \Delta w_{SK_\ell} + \Delta w'_{SK_\ell} \\ \bar{A}_{NB_\ell} = \dfrac{w_{NB_2}\bar{A}_{NB_2} + \Delta w_{SK_\ell}\bar{A}_{SK} + \Delta w'_{SK_\ell}\bar{A}'_{SK}}{w_{NB_\ell}} \end{cases}$$

TABLE OF COEFFICIENTS

| BLADE<br>COEFF. | XV-9A DUCTS | FIG. "8" TWO SPAR | FIG. "8" ONE SPAR |
|---|---|---|---|
| $D_4$ | .958 | .96 | 1.28 |
| $D_5$ | .286 | .298 | .361 |
| $m_{10}$ | .45 | 0 | .47 |
| $m_{11}$ | .44 | .35 | .35 |
| $m_{12}$ | .64 | 0 | .46 |

## SPANWISE WEIGHT DISTRIBUTION



$$\quad _{NB_2} \qquad _{NB_\ell}$$

$$\bar{h}_{c_\ell} = \bar{h}_{c_t} + \frac{1 - \bar{\ell} - \bar{e}_t}{1 - \bar{e}_t}\left(\bar{h}_{c_0} - \bar{h}_{c_t}\right)$$

$$K_{2_\ell} = \frac{\bar{h}_{c_\ell} - \bar{h}_F}{\bar{h}_R - \bar{h}_{c_\ell}} \qquad K_{t_\ell} = 1 + K_{2_\ell}$$

$$K_{3_\ell} = w_{NB_\ell}\left(\frac{\bar{h}_{NB_\ell} - \bar{h}_{c_\ell}}{\bar{h}_R - \bar{h}_{c_\ell}}\right)$$

$$A_a = 0 \qquad A_b = .848\,\bar{c}\,(\bar{L} - \bar{\ell})^2\left(\frac{V}{40}\right)^2$$

$$n_a = 3.75 \qquad n_b = 1.5$$

(64) $$\bar{M}'_{ow} = \left(\frac{w_{T_t}}{3} + \frac{w_{T_2}}{6}\right)\bar{d}_2^{\,2} + \left(\frac{w_{T_t} + w_{T_2}}{2}\right)\bar{d}_2\,\bar{d}_1 + \bar{W}_t\left(1 - \frac{\bar{\ell} + \bar{e}_t}{h'_t}\right.$$
$$+ \left(\frac{w'_{T_2}}{3} + \frac{w_{NB_\ell} + \Delta w - K_{3_\ell}}{6}\right)\bar{d}_1^{\,2} - \frac{\Delta w}{2}\,\bar{d}^{\,2} + A_\alpha$$
$$(\alpha = a,$$

(65) $$w_{F_\ell} = \frac{6\,n_\alpha\,\bar{M}'_{ow}\,\bar{t}_R\,R + \frac{F_{bw}}{\delta}\,\bar{\rho}_R^{\,2}\,K_{3_\ell}\,\pi\left(\frac{\sigma}{t}\right)}{\pi\left(\frac{\sigma}{t}\right)\left(\frac{F_{bw}}{\delta}\right)\left(\bar{\rho}_F^{\,2} + K'_{2_\ell}\,\bar{\rho}_R^{\,2}\right) - n_\alpha\,\bar{t}_R\,K_{t_\ell}\,\bar{d}_1^{\,2}\,R}$$

$$\left(w_{T_\ell}\right)_G = K_{t_\ell}\,w_{F_\ell} - K_{3_\ell} + w_{NB_\ell}$$

$$\left(w_{T_0}\right)_G = \frac{\bar{\ell}_2}{\bar{d}_1}\,w_{T_\ell} - \frac{\bar{\ell}}{\bar{d}_1}\left(w'_{T_2} - \Delta w\right)$$

$$(\bar{W}_{BU})_G = d_2\left(\frac{w_{T_0}+w_{T_2}}{2}\right) + \frac{\bar{\ell}_2}{2}\left(w'_{T_2}+w_{T_0}+\Delta w\right) - \bar{\ell}_1(\Delta w) + \frac{\bar{W}_t}{n'_t}$$

$$(\bar{I}_{\ell_f})_G = (\bar{I}_{\ell_2})_G + \frac{1}{3}\Bigg[(\bar{\ell}_2+\bar{e}_f)^3\Bigg\{w'_{T_2}\left[1-\left(\frac{\bar{e}_f}{\bar{\ell}_2+e_f}\right)^3\right] + \left(\frac{\bar{\ell}_2+\bar{e}_f}{4d_1}\right)\left(w_{T_2}+\Delta w-w'_{T_2}\right)$$

$$\left[1-\left(\frac{\bar{e}_f}{\bar{\ell}_2+\bar{e}_f}\right)^3\left(4-\frac{3\bar{e}_f}{\bar{\ell}_2+\bar{e}_f}\right)\right]\Bigg\} - \Delta w\left[(\bar{\ell}_1+\bar{e}_f)^3 - e_f^3\right]\Bigg]$$

$$(\bar{M}_{\ell_f})_G = (M'_{ow})_a + \frac{K_{4_\ell}\, w_{F_\ell}\, \bar{d}_1^2}{6} + \bar{\ell}\left[(\bar{W}_{BU})_G - \left(\frac{w_{T_0}}{3}+\frac{w_{T_\ell}}{6}\right)\bar{\ell}\right] + (\bar{W}_{BU})_G\, \bar{e}_f$$

$$(\bar{W}_{B_\ell})_G = (\bar{W}_{BU})_G - \frac{\bar{\ell}}{2}\left(w_{T_\ell}+w_{T_0}\right)$$

$$(\bar{M}_{\ell_\ell})_G = (\bar{M}'_{ow})_a + \frac{K_{4_\ell}\, w_{F_\ell}\, \bar{d}_1^2}{6} + (\bar{W}_{B_\ell})_G\,(\bar{\ell}+\bar{e}_f)$$

$$(\bar{I}_{\ell_\ell})_G = (\bar{I}_{\ell_2})_G + \frac{1}{3}\Bigg[(\bar{\ell}_2+\bar{e}_f)^3\Bigg\{w'_{T_2}\left[1-\left(\frac{\bar{\ell}+\bar{e}_f}{\bar{\ell}_2+\bar{e}_f}\right)^3\right] + \left(\frac{\bar{\ell}_2+\bar{e}_f}{4\bar{d}}\right)\left(w_{T_2}+\Delta w-w'_{T_2}\right)$$

$$\left[1-\left(\frac{\bar{\ell}+\bar{e}_f}{\bar{\ell}_2+\bar{e}_f}\right)^3\left(4-3\,\frac{\bar{\ell}+\bar{e}_f}{\bar{\ell}_2+\bar{e}_f}\right)\right]\Bigg\} - \Delta w\left[(\bar{\ell}_1+\bar{e}_f)^3 - (\bar{e}_f+\bar{\ell})^3\right]\Bigg]$$

$$(\bar{M}_{COR_\ell})_G = \frac{4\lambda(r_n-\bar{e}_f)(W/b)(h/\bar{e}_f)(a)}{\left(\frac{V_T^2}{2.60}\right)(M_{\ell_f})_G}\left[\frac{(I_{\ell_\ell})_G-(2\bar{e}_f+\bar{\ell})(\bar{M}_{\ell_\ell})_G+e_f(\bar{\ell}+e_f)(\bar{W}_{B_\ell})_G}{(I_{\ell_f})_G - e_f(M_{\ell_f})_G}\right]$$

FIRST APPROXIMATION

(66)
$$\begin{cases} M_{F_\ell} = K_{F_\ell}\left(\frac{W}{b}\right)(R)(36.0) \\ M_{C_\ell} = \left[K_{C_\ell}+(M_{COR_\ell})_G\right](W/b)(R)(29.4) \end{cases} \text{TILT. HUB ROTOR}$$

$$\begin{cases} M_{F_\ell} = K_{F_\ell}(W/b)(R)(.752)\left(\frac{E}{S}\right)\left(\frac{\sigma}{b}\right)^2 \bar{D}_R^2\left(\frac{w_{F_\ell}K_{4_\ell}-K_{3_\ell}}{w_{T_t}}\right) \\ M_{C_\ell} = \frac{1090}{W}R^{6.4}\left(\frac{\sigma}{b}\right)^{2.4} \end{cases} \text{(ARTIC. ROTOR)}$$

$$(66a)\quad \text{ITERATIONS}\left\{\begin{array}{l}
\left.\begin{array}{l}
M_{F_\ell} - K_{F_\ell}\left(\frac{W}{b}\right)(R)(F_{13})(12) \\[2ex]
M_{C_\ell} = \left(K_{C_\ell} + \bar{M}_{COR_\ell}\right)\left(\frac{W}{b}\right)(R)(C_{11})(12)
\end{array}\right\}\text{(TILT. HUB ROTOR)} \\[4ex]
\left.\begin{array}{l}
M_{F_\ell} = K_{F_\ell}\,\frac{EI_{x_\ell}}{\omega_{T_t} R^2}\left(\frac{F_{13}}{4500}\right)\left(\frac{W}{b}\right)(R)(12) \\[2ex]
M_{C_\ell} = \frac{1090}{W}\,R^{6.4}\left(\frac{\sigma}{b}\right)^{2.4}
\end{array}\right\}\text{(ARTICULATE ROTOR)}
\end{array}\right.$$

$$\bar{W}'_{B_\ell} = \frac{\bar{\ell}}{2}\left(w_{T_\ell} + w_{T_2}\right) + \frac{\bar{\ell}_1}{2}\left(w'_{T_2} + \Delta w\right) - \bar{\ell}_1(\Delta w) + \frac{\bar{W}_E}{K'_\ell}$$

$$+ \frac{\bar{\ell}}{2}\left(w_{H0_\ell} - K_{3_\ell}\right) - \frac{\bar{\ell}}{2}\left(w'_{T_2} - \Delta w\right)$$

$$b_1 = \frac{\delta V_T^2}{2.68}\left[\bar{M}'_{ow} + \bar{W}'_{B_\ell}(\bar{e}_1 + \bar{\ell})\right]$$

$$b_2 = \frac{\delta V_T^2}{2.68}\left[\frac{\bar{d}_1^2}{c} + \left(\frac{\bar{\ell} + \bar{\ell}_1}{2}\right)(\bar{e}_1 + \bar{\ell})\right](K_{4_\ell})$$

$$b_3 = -K_{3_\ell} \qquad b_4 = K_{4_\ell}$$

$$b_5 = M_{F_\ell}\,\bar{x}_F\left(\frac{\delta}{2}\right) \qquad b_6 = 4\delta c\, b_8 \bar{\rho}_R^2$$

$$b_7 = 4\delta c\left(\bar{\rho}_F^2 + K_{2_\ell}\bar{\rho}_R^2\right) \qquad b_8 = \frac{Mc_\ell\,\delta}{4\delta c(\rho_R - \rho_F)}$$

$$b_9 = b_5\left(\frac{\bar{x}_R}{\bar{x}_F}\right) \qquad b_{10} = K_{2_\ell}$$

$$k_{11} = b_1 \frac{Fe_A}{F'_{TUA}} \qquad k'_{11} = b_1 \frac{Fe'_A}{F_{TUA}}$$

$$k_{22} = b_2 \frac{Fe_A}{F'_{TUA}} \qquad k'_{22} = b_2 \frac{Fe'_A}{F_{TUA}}$$

$$k_{44} = b_1 \frac{Fe_B}{F'_{TUB}} \qquad k'_{44} = b_1 \frac{Fe'_B}{F_{TUB}}$$

$$k_{55} = b_2 \frac{Fe_B}{F'_{TUB}} \qquad k'_{55} = b_2 \frac{Fe'_B}{F_{TUB}}$$

$$B_1 = k_0 b_4 b_7 - k_{22} b_7$$

$$B_2 = k_0 (b_3 b_7 + b_6 b_4) - k_{11} b_7 - k_{22} b_6 - b_4 b_8 - b_4 b_7 b_9$$

$$B_3 = k_0 b_3 b_6 - k_{11} b_6 - b_3 b_8 - b_9 (b_3 b_7 + b_4 b_6)$$

$$B_4 = - b_3 b_6 b_9$$

$$B_5 = k_0 b_{10} b_4 b_7 - k_{22} b_7 b_{10} = B_1 b_{10}$$

$$B_6 = k_0 (b_3 b_4 b_7 + b_3 b_7 b_{10} + b_4 b_6 b_{10}) - k_{11} b_7 b_{10}$$
$$- k_{22} (b_6 b_{10} + b_3 b_7) - b_4 (b_9 b_{10} + b_7 b_8)$$

$$B_7 = k_0 b_3 (b_6 b_{10} + b_3 b_7 + b_4 b_6) - k_{11} (b_6 b_{10} + b_3 b_7) - k_{22} b_3 b_6$$
$$- b_9 b_3 (b_{10} + b_4) - b_8 (b_3 b_7 + b_4 b_6)$$

$$B_8 = k_0 b_3^2 b_6 - k_{11} b_3 b_6 - b_3^2 b_9 - b_3 b_6 b_8$$

$$B_9 = k_3 b_4 - k_{55}$$

$$B_{10} = k_3 b_3 - k_{44} - b_4 b_8$$

$$B_{11} = - b_3 b_8$$

$$B_{12} = k_3 b_4 b_{10} - k_{55} b_{10}$$

$$B_{13} = k_3 b_3 (b_4 + b_{10}) - k_{44} b_{10} - k_{55} b_3 - b_4 b_8$$

$$B_{14} = k_3 b_3^2 - k_{44} b_3 - b_3 b_8$$

$$
(67)\quad \left\{ \begin{array}{l}
\left.\begin{array}{l}
\triangleright 1\; (w_{F_2})_c^3 (B_1) + (w_{F_2})_c^2 (B_2) + (w_{F_2})_c (B_3) + B_4 = 0 \\
(w'_{F_2})_c^3 (B'_1) + (w'_{F_2})_c^2 (B'_2) + (w'_{F_2})_c (B'_3) + B'_4 = 0
\end{array}\right\} \text{FRONT SPAR} \\
\left.\begin{array}{l}
\triangleright 1\; (w_{F_2})_d^3 (B_5) + (w_{F_2})_d^2 (B_6) + (w_{F_2})_d (B_7) + B_8 = 0 \\
(w'_{F_2})_d^3 (B'_5) + (w'_{F_2})_d^2 (B'_6) + (w'_{F_2})_d (B'_7) + B'_8 = 0
\end{array}\right\} \text{REAR SPAR} \\
\left.\begin{array}{l}
\triangleright 1\; (w_{F_2})_e^2 (B_9) + (w_{F_2})_e (B_{10}) + B_{11} = 0 \\
(w'_{F_2})_e^2 (B'_9) + (w'_{F_2})_e (B'_{10}) + B'_{11} = 0
\end{array}\right\} \text{FRONT SPAR} \\
\left.\begin{array}{l}
\triangleright 1\; (w_{F_2})_f^2 (B_{12}) + (w_{F_2})_f (B_{13}) + B_{14} = 0 \\
(w'_{F_2})_f^2 (B'_{12}) + (w'_{F_2})_f (B'_{13}) + B'_{14} = 0
\end{array}\right\} \text{REAR SPAR}
\end{array} \right\} \triangleright 2
$$

$$
t_{cF_2} = \frac{b_1 + b_2 w_{F_2}}{b_3 + b_4 w_{F_2}}
$$

$\triangleright 1$ If $t_{cF_2} > F_{Tco}$, use primed equations.
Use largest value of $(w_{F_2})_{a,b,c,d,e,f}$, $(w'_{F_2})_{c,d,e,f}$.

$\triangleright 2$ SEE NEXT PAGE

▷2 IF FRONT SPAR IS DESIGNED BY STRESS REQUIREMENT ON PRECEDING PAGE, RECOMPUTE $\bar{x}_{c_\ell}$, USING $w_{R_\ell}$ AND $w_{F_\ell}$ AS DETERMINED BY STRESS. IF $w_{R_\ell}$ IS CRITICAL FOR FLIGHT CONDITION, IT IS GIVEN BY: $w_{R_\ell} = w_{F_\ell} K_{2_\ell} - K_{3_\ell}$.

$$\bar{x}'_{c_\ell} = \frac{w_{F_\ell}\bar{x}_F + w_{R_\ell}\bar{x}_R + w_{NB_\ell}\bar{x}_{NB_\ell}}{w_{F_\ell} + w_{R_\ell} + w_{NB_\ell}}$$

RECOMPUTE K'S:

$$K'_{2_\ell} = \frac{\bar{x}'_{c_\ell} - \bar{x}_F}{\bar{x}_R - \bar{x}'_{c_\ell}} \qquad K'_{4_\ell} = 1 + K'_{2_\ell}$$

$$K'_{3_\ell} = w_{NB_\ell}\left(\frac{\bar{x}_{NB_\ell} - \bar{x}'_{c_\ell}}{\bar{x}_R - \bar{x}'_{c_\ell}}\right)$$

RECOMPUTE $k_2$, $k_3$, $k_4$, $k_7$, $k_{10}$, $k_{22}$, $k_{55}$, AND ALL B'S INVOLVED IN CRITICAL CONDITION. RECOMPUTE CRITICAL $w_{F_\ell}$ AND CONTINUE, USING $(K')$'s.

(68) $\bar{w}_{T_\ell} = K_{4_\ell} \bar{w}_{F_\ell} - K_{3_\ell} + \bar{w}_{NB_\ell}$

(69) $\bar{w}_{T_0} = \frac{\bar{\ell}_2}{d_1} \bar{w}_{T_\ell} - \frac{\bar{\ell}}{d_1} \left( \bar{w}'_{T_2} - \Delta w \right)$

(70) $W_{BU} = 12R \left[ \frac{\bar{d}_1}{2} \left( \bar{w}_{T_\ell} + \bar{w}_{T_2} \right) + \frac{\bar{\ell}_2}{2} \left( \bar{w}'_{T_2} + \bar{w}_{T_0} + \Delta w \right) - \bar{\ell}_1 \Delta w + \frac{\bar{W}_4}{N'_4} \right]$

$$(71)\quad \begin{cases} I_{x_t} = \frac{(45\bar{c})^2}{\delta}\left[w_{F_t}\left(\bar{\rho}_F^2 + K_{2_t}\bar{\rho}_R^2\right) - K_{3_t}\bar{\rho}_R^2\right] \\ I_{x_2} = \frac{(45\bar{c})^2}{\delta}\left[w_{F_2}\left(\bar{\rho}_F^2 + K_{2_2}\bar{\rho}_R^2\right) - K_{3_2}\bar{\rho}_R^2\right] \\ I_{x_\ell} = \frac{(45\bar{c})^2}{\delta}\left[w_{F_\ell}\left(\bar{\rho}_F^2 + K_{2_\ell}\bar{\rho}_R^2\right) - K_{3_\ell}\bar{\rho}_R^2\right] \\ I_x(x) = I_{x_t} + (I_{x_2} - I_{x_t})\left(\frac{1-x}{d_1}\right) & \left\{\bar{l}_2 + \bar{e}_t \le x \le 1\right\} \\ I_x(x) = I_{x_2} + (I_{x_\ell} - I_{x_2})\left(\frac{\bar{l}_2 + \bar{e}_t - x}{d_1}\right) & \left\{\bar{e}_t \le x \le \bar{l}_2 + \bar{e}_t\right\} \end{cases}$$

$$(72)\quad \begin{cases} I_{Y_t} = \frac{(45\bar{c})^2}{\delta}(w_{F_t})(\bar{\nu}_R - \bar{\nu}_F)^2\left(\frac{K_{2_t}w_{F_t} - K_{3_t}}{K_{4_t}w_{F_t} - K_{3_t}}\right) \\ I_{Y_2} = \frac{(45\bar{c})^2}{\delta}(w_{F_2})(\bar{\nu}_R - \bar{\nu}_F)^2\left(\frac{K_{2_2}w_{F_2} - K_{3_2}}{K_{4_2}w_{F_2} - K_{3_2}}\right) \\ I_{Y_\ell} = \frac{(45\bar{c})^2}{\delta}(w_{F_\ell})(\bar{\nu}_R - \bar{\nu}_F)^2\left(\frac{K_{2_\ell}w_{F_\ell} - K_{3_\ell}}{K_{4_\ell}w_{F_\ell} - K_{3_\ell}}\right) \\ I_Y(x) = I_{Y_t} + (I_{Y_2} - I_{Y_t})\left(\frac{1-x}{d_2}\right) & \left\{\bar{l}_2 + \bar{e}_t \le x \le 1\right\} \\ I_Y(x) = I_{Y_2} + (I_{Y_\ell} - I_{Y_2})\left(\frac{\bar{l}_2 + \bar{e}_t - x}{d_1}\right) & \left\{\bar{e}_t \le x \le \bar{l}_2 + \bar{e}_t\right\} \end{cases}$$

$$
(73)\quad \left\{ \begin{array}{ll}
I_{P_t} = \frac{(45\bar{c})^2}{386}\left\{ w_{F_t}(\bar{x}_{c_t}-\bar{x}_F)(\bar{x}_R-\bar{x}_F) + w_{NB_t}\left[.052\left(\frac{\bar{x}_R}{.528}\right)+(\bar{x}_{NB_t}-\bar{x}_{c_t})(\bar{x}_{NB_t}-\bar{x}_R)\right]\right\} & \\
I_{P_2} = \frac{(45\bar{c})^2}{386}\left\{ w_{F_2}(\bar{x}_{c_2}-\bar{x}_F)(\bar{x}_R-\bar{x}_F) + w_{NB_t}\left[.052\left(\frac{\bar{x}_R}{.528}\right)+(\bar{x}_{NB_t}-\bar{x}_{c_2})(\bar{x}_{NB_t}-\bar{x}_R)\right]\right\} & \\
I'_{P_2} = \frac{(45\bar{c})^2}{386}\left\{ w_{F_2}(\bar{x}_{c_2}-\bar{x}_F)(\bar{x}_R-\bar{x}_F) + w_{NB_2}\left[.052\left(\frac{\bar{x}_R}{.528}\right)+(\bar{x}_{NB_2}-\bar{x}_{c_2})(\bar{x}_{NB_2}-\bar{x}_R)\right]\right\} & \\
I_{P_\ell} = \frac{(45\bar{c})^2}{386}\left\{ w_{F_\ell}(\bar{x}_{c_\ell}-\bar{x}_F)(\bar{x}_R-\bar{x}_F) + w_{NB_\ell}\left[.028\left(\frac{\bar{x}_R}{.528}\right)+(\bar{x}_{NB_\ell}-\bar{x}_{c_\ell})(\bar{x}_{NB_\ell}-\bar{x}_R)\right]\right\} & \\
w_{F_1} = w_{F_2} + (w_{F_\ell}-w_{F_2})\left(1-\frac{d}{d_1}\right) & \\
I_{P_1} = \frac{(45\bar{c})^2}{386}\left\{ w_{F_1}(\bar{x}_{c_1}-\bar{x}_F)(\bar{x}_R-\bar{x}_F) + w_{NB_2}\left[.052\left(\frac{\bar{x}_R}{.528}\right)+(\bar{x}_{NB_2}-\bar{x}_{c_1})(\bar{x}_{NB_2}-\bar{x}_R)\right]\right\} & \\
I'_{P_1} = \frac{(45\bar{c})^2}{386}\left\{ w_{F_1}(\bar{x}_{c_1}-\bar{x}_F)(\bar{x}_R-\bar{x}_F) + w_{NB_\ell}\left[.028\left(\frac{\bar{x}_R}{.528}\right)+(\bar{x}_{NB_\ell}-\bar{x}_{c_1})(\bar{x}_{NB_\ell}-\bar{x}_R)\right]\right\} & \\
I_P(x) = I_{P_t} + (I_{P_2}-I_{P_t})\left(\frac{1-x}{d_2}\right) & \left\{\bar{\ell}_2+\bar{e}_f < x \leq 1\right\} \\
I_P(x) = I'_{P_2} + (I_{P_1}-I'_{P_2})\left(\frac{\ell_2+\bar{e}_f-x}{d_1-d}\right) & \left\{\bar{\ell}_1+\bar{e}_f < x < \ell_2+\bar{e}_f\right\} \\
I_P(x) = I'_{P_1} + (I_{P_\ell}-I'_{P_1})\left(\frac{\ell_1+\bar{e}_f-x}{d}\right) & \left\{\bar{e}_f \leq x < \bar{\ell}_1+\bar{e}_f\right\}
\end{array}\right.
$$

$$
(74)\quad \begin{cases}
J_t = 17.80\, \bar{t}_{sk_t}\, \bar{c}^3 \left(\frac{\bar{t}_t}{.113}\right)^2 \left[.951 + 2.4\left(\bar{N}_B - .5\right)\right] & \left\{\bar{l}_2 + \bar{e}_t \leq x \leq 1\right\} \\
J_2 = J_t \left(\frac{\bar{t}_{sk_2}}{\bar{t}_{sk_t}}\right)\left(\frac{\bar{t}_2}{\bar{t}_t}\right) & \left\{\bar{l}_1 + \bar{e}_t < x < \bar{l}_2 + \bar{e}_t\right\} \\
J_1 = J_2 \left(\frac{\bar{t}_{sk_1}}{\bar{t}_{sk_2}}\right) & \left\{e_t \leq x \leq \bar{l}_1 + \bar{e}_t\right\}
\end{cases}
$$

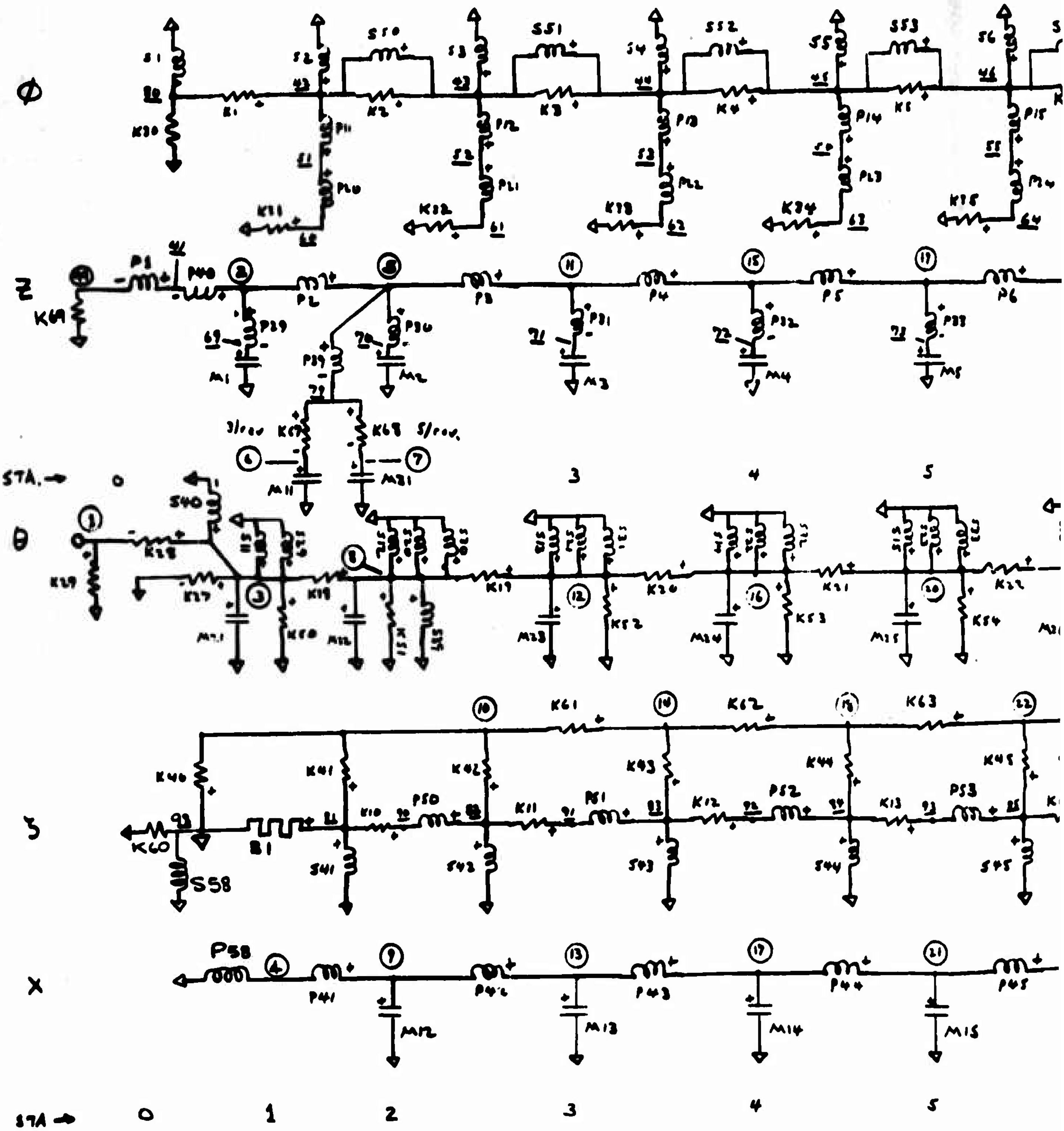
$$
(75)\quad \begin{cases}
w_T = w_{T_t} + \left(w_{T_2} - w_{T_t}\right)\left(\frac{1 - x}{\bar{d}_2}\right) & \left\{\bar{l}_2 + \bar{e}_t \leq x \leq 1\right\} \\
w_T = w'_{T_2} + \left(w_{T_1} - w'_{T_2}\right)\left(\frac{\bar{l}_2 + \bar{e}_t - x}{\bar{d}_1}\right) & \left\{\bar{l}_1 + \bar{e}_t < x < \bar{l}_2 + \bar{e}_t\right\} \\
w_T = w'_{T_2} - \Delta w + \left(w_{T_1} + \Delta w - w'_{T_2}\right)\left(\frac{l_2 + \bar{e}_t - x}{\bar{d}_1}\right) & \left\{\bar{e}_t \leq x \leq \bar{l}_1 + \bar{e}_t\right\}
\end{cases}
$$

$$
(76)\quad I_e = 1.49\, R^3 \Bigg\{ w_{T_t}\left(\bar{L}^3 - \bar{l}_2^3\right) + \left(\frac{w_{T_2} - w_{T_t}}{4\,\bar{d}_2}\right)\left[\bar{L}^4 - \bar{l}_2^3\left(4\bar{L}^3 - 3l_2\right)\right]
$$

$$
+ \bar{l}_2^3\left(w'_{T_2} + \frac{w_{T_1} + \Delta w - w'_{T_2}}{4}\right) - \Delta w\, \bar{l}_1^3 + \frac{3\bar{W}_t}{x'_t}\left(x'_t - \bar{e}_t\right)^3 \Bigg\}
$$

# APPENDIX IV

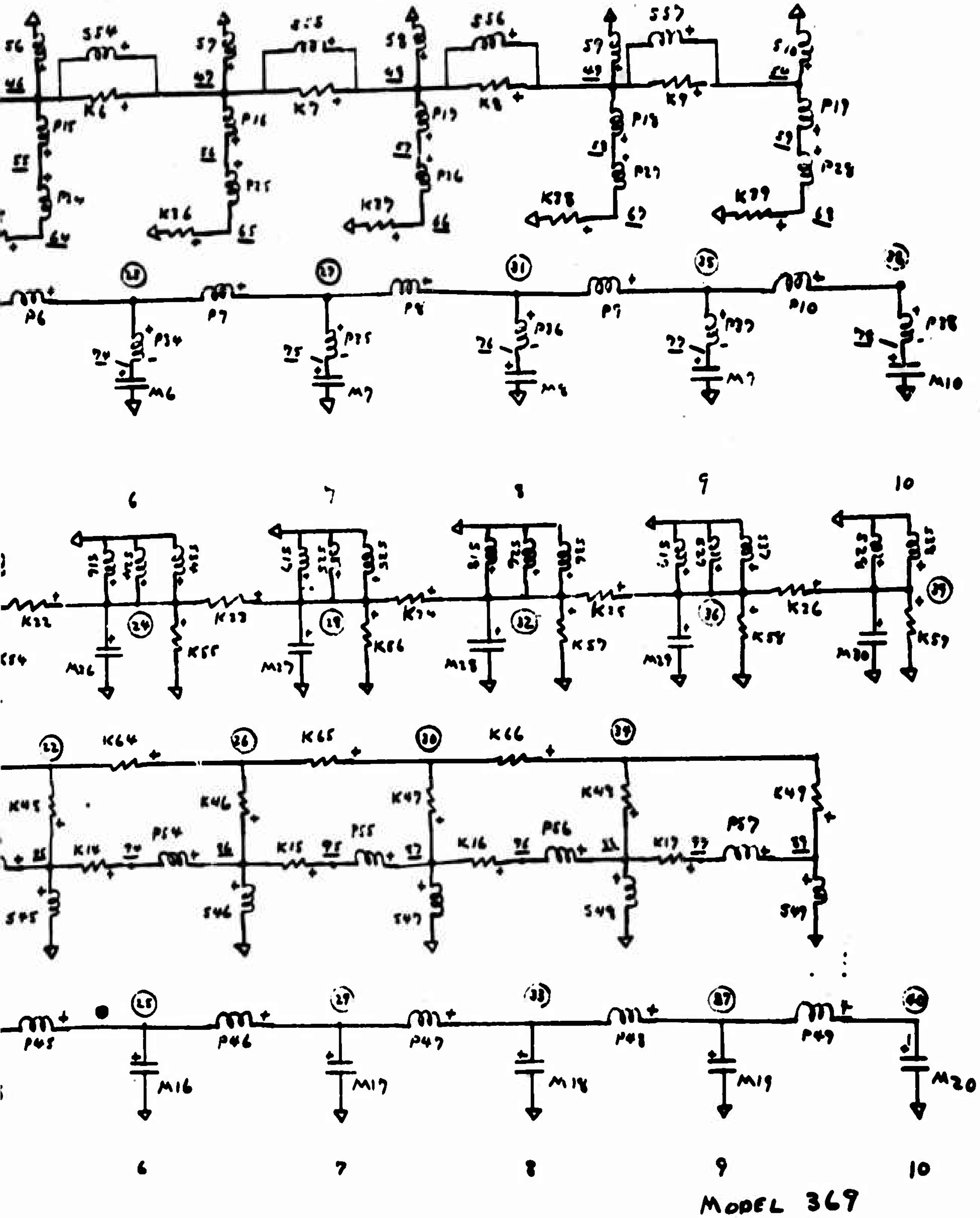
## COMPUTER CIRCUIT DIAGRAM AND INPUT DAT



Connection Diagram - Heavy-Lift Helic

A

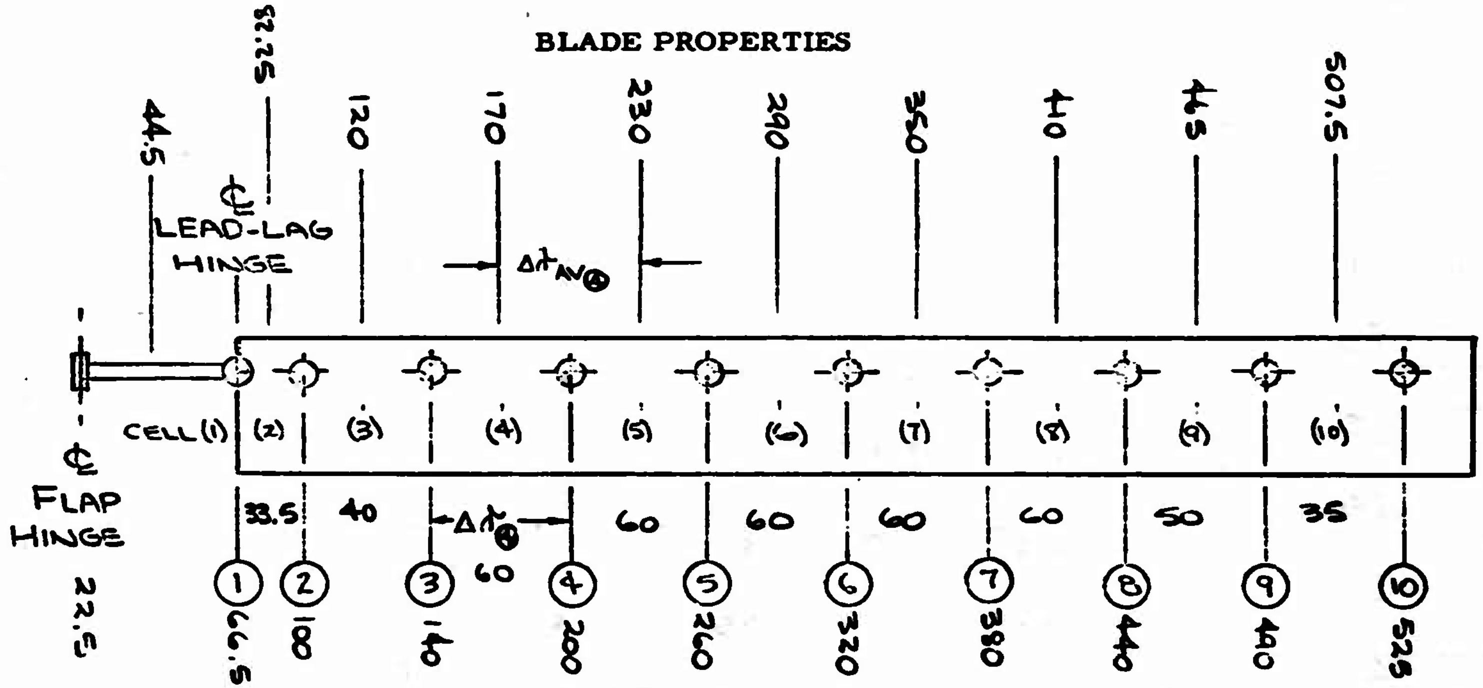
## K IV
## T DATA FOR COUPLED ANALYSIS



t Helicopter Blade Structure

B

BLADE PROPERTIES



Evaluate $EI_f$ and $EI_c$ at the stations

Evaluate GJ between stations

Evaluate $X_{cg}$ and $X_{sc}$ at the stations

Stations and cell ℄'s are measured from ℄ rotor

9704

| ① | ② | ③ | ④ | ⑤ | ⑥ | ⑦ | ⑧ | ⑨ | ⑩ |
|---|---|---|---|---|---|---|---|---|---|
| STA | $r$ | $\Delta r$ | $(\Delta r)_{AV}$ | $\frac{EI_f}{10^6}$ | $\frac{K_1}{10^6}$ | $\frac{EI_c}{10^6}$ | $\frac{K_2}{10^6}$ | $\frac{GJ}{10^6}$ | $\frac{K_3}{10^6}$ |
|  | (IN.) | $②_{m+1} - ②_m$ | $\frac{③_m + ③_{m+1}}{2}$ | (# IN.$^2$) | ⑤/④ | (# IN.$^2$) | ⑦/④ | (# IN.$^2$) | ⑨/$③_{m-}$ |
| 0 | 22.5 |  |  | ~ |  |  |  |  |  |
|  |  | 44 |  |  |  |  |  | 570 | 13.0 |
| 1 | 66.5 |  | 38.75 | 540 | 13.9 | 0 | 0 |  |  |
|  |  | 33.5 |  |  |  |  |  | 570 | 17.0 |
| 2 | 100 |  | 36.75 | 500 | 13.6 | 3830 | 104 |  |  |
|  |  | 40 |  |  |  |  |  | 570 | 14.2 |
| 3 | 140 |  | 50 | 456 | 9.1 | 3780 | 76 |  |  |
|  |  | 60 |  |  |  |  |  | 570 | 9.5 |
| 4 | 200 |  | 60 | 383 | 6.4 | 3670 : | 61 |  |  |
|  |  | ↕ |  |  |  |  |  | 570 | 9.5 |
| 5 | 260 |  | ↕ | 310 | 5.7 | 3480 | 58 |  |  |
|  |  | ↕ |  |  |  |  |  | 570 | 9.5 |
| 6 | 320 |  | ↕ | 240 | 4.0 | 3210 | 53 |  |  |
|  |  | ↕ |  |  |  |  |  | 520 | 8.6 |
| 7 | 380 |  | 60 | 167 | 2.8 | 2860 | 40 |  |  |
|  |  | 60 |  |  |  |  |  | 400 | 6.7 |
| 8 | 440 |  | 55 | 95 | 1.7 | 2380 | 43 |  |  |
|  |  | 50 |  |  |  |  |  | 320 | 6.4 |
| 9 | 490 |  | 42.5 | 35 | .8 | 1790 | 42 |  |  |
|  |  | 35 |  |  |  |  |  | 300 | 8.6 |
| 10 | 585 |  | 32.5 | 13 | .5 | 1120 | 45 |  |  |
|  |  | 15 |  |  |  |  |  |  |  |
|  |  | P/S.&R/S$_2$ |  |  | $K_1$ |  | $K_2$ |  | $K_3$ |

FORM 0707

| ① | ⑪ | ⑫ | ⑬ | ⑭ | ⑮ | ⑯ | ⑰ | ⑱ | ⑲ |
|---|---|---|---|---|---|---|---|---|---|
| STA | $w$ | $\Delta W$ | $W$ | $m_1$ | $m \cdot r$ | $T/\Omega^2$ | $\frac{T \times \Delta r}{\Omega^2}$ | $r_{cc}$ | $r_{sc}$ |
|  | (#/IN.) | (#) | ⑪×④+⑫ | ⑬/386 | ⑭×② | ↑Σ⑮ | ⑯×③ $m^-$ | (IN.) | (IN.) |
| 0 |  |  |  |  |  |  |  |  |  |
| 1 | 6.0 |  | 233 | .604 | 40.2 | 788.6 | 34,700 | 16.44 | 15.00 |
| 2 | 5.6 |  | 206 | .534 | 53.4 | 748.4 | 25,070 | 16.26 |  |
| 3 | 3.2 |  | 160 | .445 | 58.1 | 695.0 | 27800 | 16.02 |  |
| 4 | 2.14 |  | 128 | .332 | 66.4 | 636.9 | 38,210 | 15.66 |  |
| 5 | 2.06 |  | 124 | .322 | 83.7 | 570.5 | 34,230 | 15.36 |  |
| 6 | 2.00 |  | 120 | .311 | 99.5 | 486.8 | 29,210 | 15.0 |  |
| 7 | 1.90 |  | 114 | .296 | 112.5 | 387.3 | 23,240 | 14.70 |  |
| 8 | 1.83 |  | 101 | .262 | 115.3 | 274.8 | 16,490 | 14.34 |  |
| 9 | 1.68 |  | 71 | .184 | 90.2 | 159.5 | 7990 | 14.10 |  |
| 10 | 1.56 | 11 | 51 | .132 | 69.3 | 69.3 | 2430 | 13.86 | 15.00 |
|  |  |  | 1308 | $m_1$ & $m_2$ |  |  | $K_1/\Omega^2$ & $K_2/\Omega^2$ |  |  |

FORM 9707

| ① | ⑳ | ㉑ | ㉒ | ㉓ | ㉔ | ㉕ | ㉖ | ㉗ | ㉘ |
|---|---|---|---|---|---|---|---|---|---|
| STA | X | m.X | Σm.X | $M_{\varphi i}/\Omega^2$ | $M_{\varphi o}/\Omega^2$ | $\mu_{T_i}$ | $\mu_{T_i}-\mu_{3e}$ | $P_4/S_4$ | $\mu_{T_o}$ |
| | ⑱-⑲ | ⑭×⑳ | ↑Σ㉑ | $②_{m-1}$×㉒ | $②_m$×㉒ | -㉓/⑯ | ㉕-⑲ | ㉖/$③_{m}$- | -㉔/⑯ |
| 0 | | | | | | | | | |
| 1 | 1.44 | .870 | 1.723 | 38.8 | 114.6 | -.05 | | -.0011 | -.15 |
| 2 | 1.26 | .673 | .853 | 56.7 | 85.3 | -.08 | | -.0024 | -.11 |
| 3 | 1.02 | .423 | .180 | 18.0 | 25.2 | -.03 | | -.0008 | -.04 |
| 4 | .66 | .219 | -.243 | -34.0 | -48.6 | +.05 | | +.0008 | +.08 |
| 5 | .36 | .116 | -.462 | -92.4 | -120.1 | .16 | | .0027 | .21 |
| 6 | 0 | 0 | -.578 | -150.3 | -185.0 | .31 | | .0052 | .39 |
| 7 | -.30 | -.089 | -.578 | -185.0 | -219.6 | .48 | | .0080 | .57 |
| 8 | -.66 | -.173 | -.489 | -185.5 | -215.2 | .68 | | .0113 | .78 |
| 9 | -.90 | -.166 | -.316 | -139.0 | -154.9 | .87 | | .0176 | .97 |
| 10 | -1.14 | -.150 | -.150 | -73.5 | -79.8 | 1.14 | | .0326 | 1.14 |
| | $P_3/S_3$ | | | | | | | $P_4/S_4$ | |

FORM 9707

FORM 8707

| ① | ㉙ | ㉚ | ㉛ | ㉜ | ㉝ | ㉞ | ㉟ | ○ | ○ |
|---|---|---|---|---|---|---|---|---|---|
| STA | $N_{HTO}-N_{SE}$ | $P_S/S_S$ | $I_P$ | $\theta_0$ | $\cos 2\theta_0$ | $I_P \cos 2\theta_0$ | $\tan \theta_0$ | | |
| | ㉘ - ⑲ | ㉙/③$_m$- | (#IN. SEC$^2$) | | | ㉛ x ㉝ | | | |
| 0 | | | | | | | | | |
| 1 | | -.0034 | 136 | 13.0 | | 122 | .225 | | |
| 2 | | -.0033 | 120 | 12.5 | | 109 | .216 | | |
| 3 | | -.0010 | 93 | 11.9 | | 85 | .206 | | |
| 4 | | +.0013 | 75 | 11.0 | | 69. | .191 | | |
| 5 | | .0035 | 72 | 10.2 | | 67 | .177 | | |
| 6 | | .0063 | 70 | 9.3 | | 66 | .162 | | |
| 7 | | .0095. | 67 | 8.4 | | 64 | .146 | | |
| 8 | | .0130 | 59 | 7.5 | | 57. | .131. | | |
| 9 | | .0194 | 41 | 6.7 | | 40 | .117 | | |
| 10 | | .0225 | 30 | 6.2 | | 29 | .108 | | |
| | | $P_S/S_S$ | $m_3$ | | | $K_L/\Omega^2$ | $P_6/S_6$ | | |

## SUMMARY

| STA | $P_1/S_1$ | $K_1/10^6$ | $K_2/10^6$ | $K_3/10^6$ | $m_1$ | $K_4/10^6$ | $P_3/S_3$ | $P_4/S_4$ | $P_5/S_5$ |
|---|---|---|---|---|---|---|---|---|---|
| 1 | 44 | 13.9 | | | .604 | 7.81 | 1.44 | | |
| 2 | 33.5 | 13.6 | 104 | 17.0 | .534 | 5.64 | 1.26 | -.0024 | -.0033 |
| 3 | 40 | 9.1 | 76 | 14.2 | .415 | 6.26 | 1.02 | -.0008 | -.0010 |
| 4 | 60 | 6.4 | 61 | 9.5 | .332 | 8.60 | .66 | +.0008 | +.0013 |
| 5 | ↕ | 5.2 | 58 | 9.5 | .322 | 7.70 | .36 | .0027 | .0035 |
| 6 | ↕ | 4.0 | 53 | 9.5 | .311 | 6.57 | 0 | .0052 | .0063 |
| 7 | ↕ | 2.8 | 48 | 8.6 | .296 | 5.23 | -.30 | .0080 | .0095 |
| 8 | 60 | 1.7 | 43 | 6.7 | .262 | 3.71 | -.66 | .0113 | .0130 |
| 9 | 50 | .8 | 42 | 6.4 | .184 | 1.80 | -.90 | .0174 | .0194 |
| 10 | 35 | | | 8.6 | .132 | .55 | -1.14 | .0326 | .0325 |
| | S-1 thru S-10 | K-1 thru K-9 ✓ | K-10 thru K-17 ✓ | K-18 thru K-26 ✓ | M-1 thru M-10 ✓ | K-30 thru K-39 ✓ | S-29 thru S-38 | S-11 thru S-19 | S-20 thru S-28 |

| STA | $m_3$ | $\frac{K_6}{10^6}$ | $P_6/S_6$ | $m_2$ | $\frac{K_5}{10^6}$ | $P_5/S_2$ |
|---|---|---|---|---|---|---|
| 1 | 136 | .027 | | | 7.81 | |
| 2 | 120 | .025 | .216 | .534 | 5.64 | 33.5 |
| 3 | 93 | .019 | .206 | .415 | 6.26 | 40 |
| 4 | 75 | .016 | .191 | .332 | 8.60 | 60 |
| 5 | 72 | .015 | .177 | .322 | 7.70 | |
| 6 | 70 | .015 | .162 | .311 | 6.57 | ↕ |
| 7 | 67 | .014 | .146 | .296 | 5.23 | |
| 8 | 59 | .013 | .131 | .262 | 3.71 | 60 |
| 9 | 41 | .009 | .117 | .184 | 1.80 | 50 |
| 10 | 30 | .007 | | .132 | .55 | 33 |
| | M-21 thru M-30 ✓ | K-50 thru K-59 ✓ | S-50 thru S-59 | M-12 thru M-20 ✓ | K-40 thru K-49 ✓ | S-41 thru S-49 |

K-27 = 24,400

K-28 = $3.74 \times 10^6$

K-29 = $10^8$

K-60 = $214 \times 10^6$

K-67
K-68
M-11
M-31
S-39
} none

B-1 = 500,000

S-40 = 0

S-59 = 66.5

K-69 = $10^8$

# APPENDIX V
# STANDARD STRUCTURAL CELL FOR THE REPRESENTATION OF MECHANICAL EFFECTS IN HELICOPTER ROTOR BLADES

## 1. INTRODUCTION

A standard structural cell for rotor blades is described for use in conjunction with the digital computer program, SADSAM IV, developed for Hughes Tool Company by the MacNeal-Schwendler Corporation. Structures are represented in SADSAM IV by combinations of simple springs, masses, dampers and generalized leverage elements (otherwise called "constraints" or "transformers"). In general, the user selects a combination of simple structural elements to represent each particular structure, so that the program is suitable for a very broad range of applications.

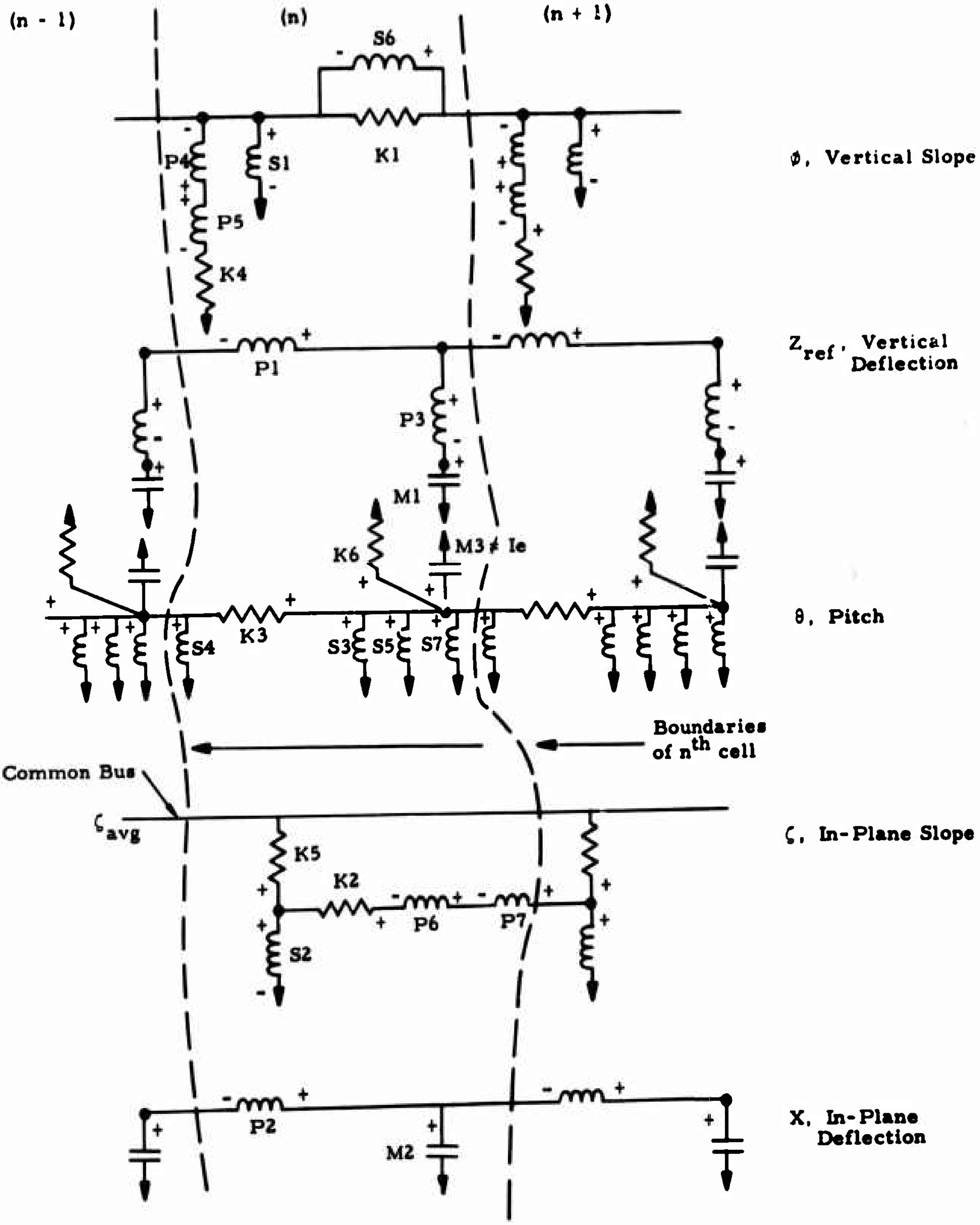
Since the analysis of rotors is a primary application of the program, and since most rotor blades are similar in their primary structural behavior, it is both desirable and feasible to describe a standard structural cell for rotor blades that can be used in rotor blade analysis. The major differences between rotor designs are usually confined to the hub and control system; these are elements which will require separate treatment for each type of rotor system and which can be conveniently treated by the basic computer program due to its flexibility.

The arrangement of elements for the standard structural cell is shown in the diagram on the following page. The electrical circuit notation employed in the diagram is described in the users manual for SADSAM IV and also in References 24 and 25. The identification of the elements in the model, formulas for their calculation, and interpretation of results obtained from the model are described in detail in section 3 of this appendix. The mathematical derivation of the standard structural cell is discussed in the following paragraphs.

## 2. DEVELOPMENT OF STANDARD CELL FOR REPRESENTATION OF MECHANICAL EFFECTS IN BLADES

The standard cell for the representation of mechanical effects in blades is similar to that developed in Reference 24. The present treatment differs in the following respects:

1. A finite-difference beam model is used rather than a "Russell" beam model.



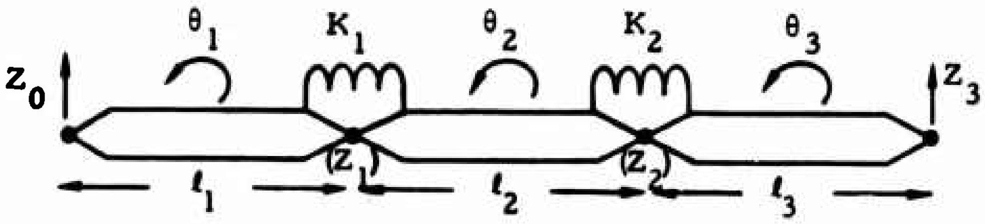
STANDARD STRUCTURAL CELL FOR BLADE

2. Mass coupling between pitching and flapping is treated by chordwise levers rather than by mutual mass coupling.
3. The arrangement of elements to represent coupling between vertical and in-plane bending due to blade pitch is different in order to facilitate measurement of flapwise bending moment.
4. The centrifugal force coupling between pitching and flapping is treated in a more correct manner.

Since the discussion in Reference 24 is quite detailed, only the manner of treating the differences listed above will be described here.

## 2.1 Finite Difference Beam Model

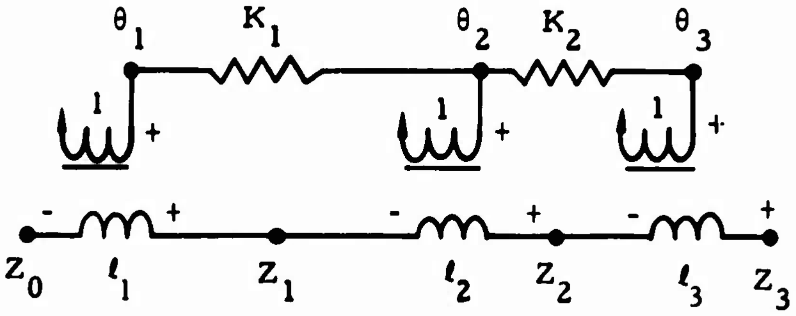
The finite-difference beam model is equivalent to the following arrangement of rigid levers and springs:



The springs resist rotation between adjacent levers. Their values are given by:

$$K_n = \frac{2EI}{\ell_n + \ell_{n+1}}$$

The formal circuit diagram for the finite-difference model is shown below:

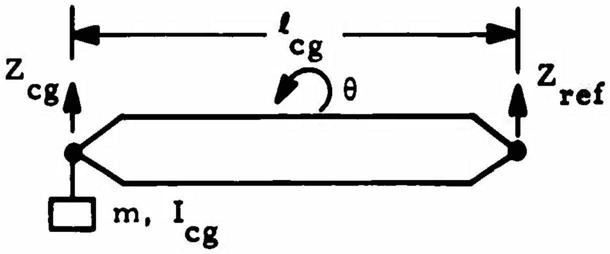


Internal forces in the springs K represent bending moment. Internal forces in the transformers represent shear.

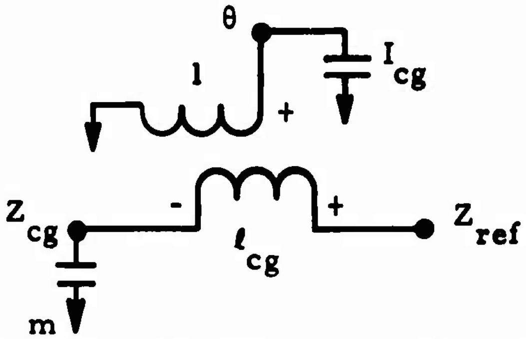
A comprehensive treatment of the finite-difference beam model, including a discussion of finite-difference errors in both static and dynamic analysis, is given in Reference 25. The finite-difference beam model has been chosen for the standard cell because it eliminates rotations as independent degrees of freedom, which is desirable due to the limitation of SADSAM IV to 50 independent degrees of freedom. Note that transverse shear flexibility in the beam is also eliminated for the same reason.

## 2. 2 Mass Coupling Between Pitching and Flapping

Mass coupling between pitching and flapping is treated by means of a lever that locates the chordwise position of the center of gravity of a blade section relative to the elastic axis as shown below:

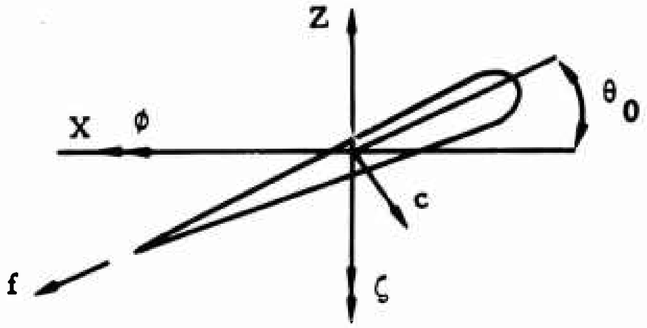


The formal circuit diagram is as follows:



## 2. 3 Elastic Coupling Between Vertical and In-Plane Bending

Consider a blade section that is rotated through an angle $\theta_0$ with respect to a horizontal axis as shown below:



The principal axes, f and c, are also rotated through the angle $\theta_0$. The relationship between moments and curvatures about the vertical and inplane axes may easily be shown to be:

$$\begin{Bmatrix} M_\phi \\ M_\zeta \end{Bmatrix} = \left[\begin{array}{c|c} K_{\phi\phi} & K_{\phi\zeta} \\ \hline K_{\phi\zeta} & K_{\zeta\zeta} \end{array}\right] \begin{Bmatrix} \Delta\phi \\ \Delta\zeta \end{Bmatrix} \tag{77}$$

where

$$K_{\phi\phi} = \frac{\left(EI_f \cos^2\theta_0 + EI_c \cdot \sin^2\theta_0\right)}{\Delta\ell} \tag{78}$$

$$K_{\phi\zeta} = \frac{\left[-\sin\theta_0 \cos\theta_0 \left(EI_c - EI_f\right)\right]}{\Delta\ell} \tag{79}$$

$$K_{\zeta\zeta} = \frac{\left(EI_c \cos^2\theta_0 + EI_f \cdot \sin^2\theta_0\right)}{\Delta\ell} \tag{80}$$

Two simplifying approximations will be made:

1. $\theta_0$ is small so that $\sin\theta_0 \simeq \tan\theta_0$ and $\cos\theta_0 \simeq 1$.
2. $I_c >> I_f$ so that $EI_f$ may be ignored in equations (79) and (80). Note that when this assumption is not valid, $EI_f$ should be retained in equation (79).

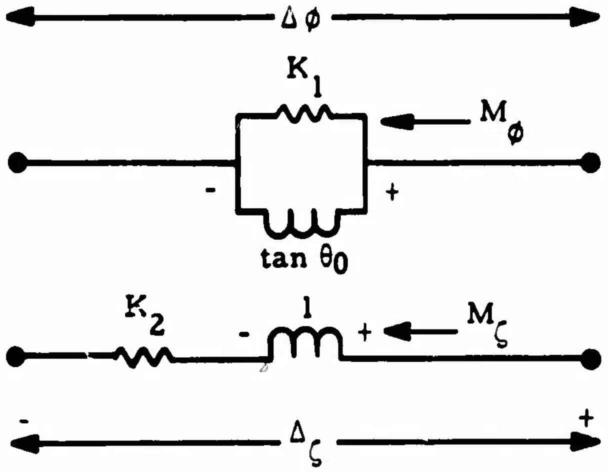
With these simplifications the above equations may be written:

$$K_{\phi\phi} = \frac{\left(EI_f + EI_c \cdot \tan^2\theta_0\right)}{\Delta\ell} \tag{78a}$$

$$K_{\phi\zeta} = \frac{\left(-\tan\theta_0 \cdot EI_c\right)}{\Delta\ell} \tag{79a}$$

$$K_{\zeta\zeta} = \frac{EI_c}{\Delta\ell} \tag{80a}$$

An equivalent circuit model that satisfied equation (77) with these values is as follows:



where $K_1 = EI_f/\Delta\ell$ and $K_2 = EI_c/\Delta\ell$.

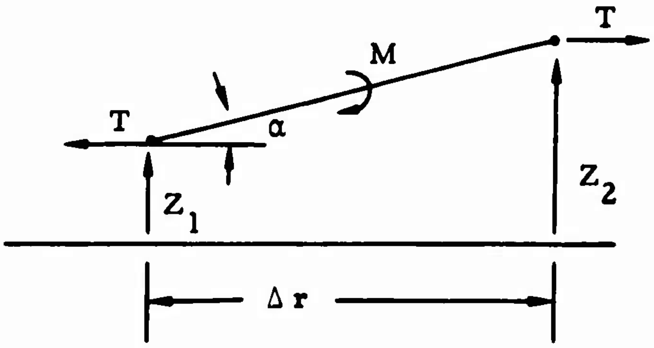
The internal force in $K_1$ is

$$M_{K_1} = M_\phi + \tan\theta_0 \cdot M_\zeta \tag{81}$$

which is approximately equal to the flapwise bending moment (moment about the chord axis).

## 2.4 Centrifugal Force Coupling Between Pitching and Flapping

The idealized element to represent centrifugal force stiffening is a tensioned string that resists rotations about axes normal to the string. Consider a tensioned string element of length $\Delta r$ with tension $T$ as shown below:



The restoring moment is

$$M = T \cdot \Delta r \cdot \alpha \tag{82}$$

which is equivalent to a couple force

$$F_{Z_2} = -F_{Z_1} = \frac{M}{\Delta r} = \frac{T}{\Delta r}(Z_2 - Z_1) \qquad (83)$$

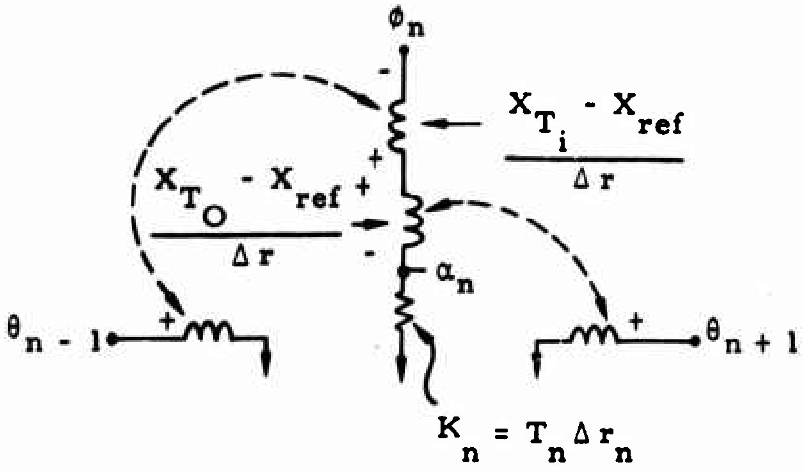
Stiffness coupling between pitching and flapping occurs because the string is not located on the shear center of the blade and therefore the displacements $Z_2$ and $Z_1$ include contributions from both pitching and flapping. The location of the string is indicated in the diagram on page 364 as the "tension axis". The tension axis at any spanwise station is located at the centroid of spanwise tension over the blade cross section and includes contributions from steady aerodynamic chordwise bending as well as from centrifugal force. The tension axis is discontinuous due to the concentration of mass at discrete points.

Referring to equation (83) and the diagram on page 364, the rotation $\theta_n$ for the $n^{th}$ cell is

$$\alpha_n = \frac{Z_2 - Z_1}{\Delta r}$$

$$= \frac{1}{\Delta r}\left[\phi_n - \frac{X_{T_O} - X_{ref}}{\Delta r}\theta_n + \frac{X_{T_i} - X_{ref}}{\Delta r}\theta_{n-1}\right] \qquad (84)$$

This relationship and the spring restraint are represented by the following equivalent circuit model.

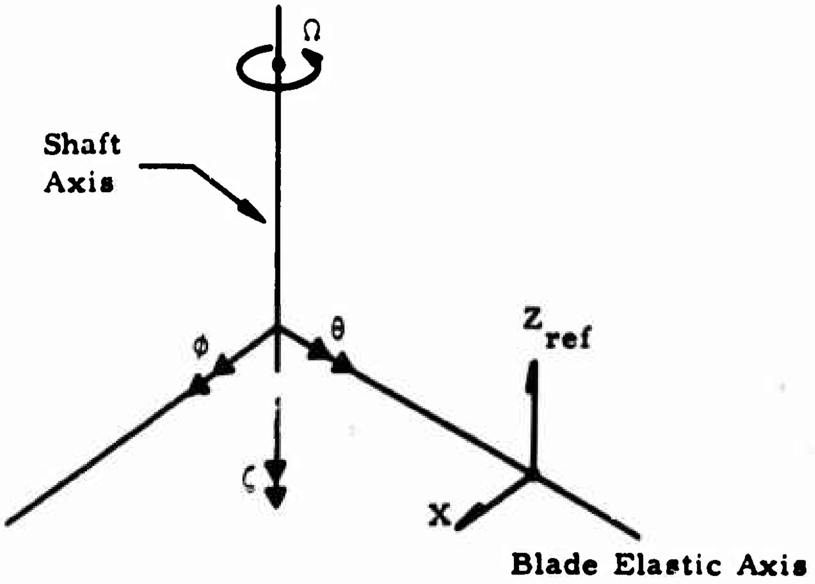


The elements in this model are represented by $K_4$, $P_4/S_4$ and $P_5/S_5$ in the diagram on page 355.

Formulas for computing the magnitude of the tension force and the inboard and outboard locations of the tension axis are given in section 3.4.

## 3. DESCRIPTION OF STANDARD STRUCTURAL CELL

### 3.1 Identification of Coordinate Directions



X: Blade motion perpendicular to shaft axis and to blade elastic axis; positive aft.

$Z_{ref}$: Blade motion parallel to shaft axis. Measured at elastic axis; positive up.

$\phi$: Local vertical blade slope; normal to shaft axis and to elastic axis; positive tip up.

$\zeta$: Local in-plane blade slope; parallel to shaft axis; positive tip aft.

$\theta$: Local blade pitch angle; parallel to blade elastic axis; positive leading edge up.

NOTES:

1. $\theta$ does not include built-in twist; $\theta$ may or may not include collective pitch, depending on whether collective pitch is included in the blade root boundary condition or in the specification of aerodynamic forces. $\theta$ includes cyclic pitch and elastic twist.

2. If the effect of static coning on coupling between pitch and lead-lag motion is included in the analysis, $\zeta$ is measured perpendicular to

the statically deformed blade elastic axis ; i.e., the ζ axis is rotated in a vertical plane through an angle equal to the local blade slope.

## 3. 2 Identification of Internal Forces

In the diagram on page 355:

Element K1: Flapwise bending moment, that is, bending moment about an axis parallel to the blade chord; positive for tip bending up. Computed at mass stations.

Element K2: In-plane bending moment, that is, bending moment about the ζ axis. Positive for tip bending aft. In-plane bending moment does <u>not</u> coincide with chordwise bending moment if blade pitch is not zero. Computed at mass stations.

Element K3: Twisting moment about blade elastic axis; positive for tip twisting up. Computed between mass stations.

Element P1: Total shear force in a vertical plane including elastic shear force and the vertical component of blade tension. Positive for tip up. Computed between mass stations.

Element P2: Total shear force in the in-plane direction, that is, in a plane perpendicular to the ζ axis. Positive for tip aft. Computed between mass stations.

## 3. 3 Identification of Elements in the Model

### Masses

$M_1$: Vertical component of lumped mass.

$M_2$: In-plane component of lumped mass.

$M_3$: Polar moment of inertia of blade section mass about the center of gravity (= $I_p$).

### Springs

$$K_1 = \frac{2EI_f}{\Delta r_n + \Delta r_{n+1}}$$ : Flapwise bending stiffness. Located at mass stations.

$$K_2 = \frac{2EI_c}{\Delta r_n + \Delta r_{n+1}}$$ : Chordwise bending stiffness. Located at mass stations.

$K_3 = \frac{GJ}{\Delta r_n}$ : Torsional stiffness. Located between mass stations.

$K_4 = T_n \cdot \Delta r_n$: Centrifugal force stiffening for vertical motions. Located between mass stations.

$K_5 = T_n \Delta r_n$: Centrifugal force stiffening for in-plane motions. Located between mass stations.

$K_6 = \Omega^2 (I_Z - I_X)$: Centrifugal force stiffening for pitch (tennis-racket effect). Located at mass stations.

### Transformers

$\frac{P_1}{S_1} = T_1 = \Delta r_n$ = Spanwise lever for vertical bending.

$\frac{P_2}{S_2} = T_2 = \Delta r_n$ = Spanwise lever for in-plane bending.

$\frac{P_3}{S_3} = T_3 = X_{cg} - X_{ref}$: Chordwise lever for location of blade center of gravity relative to elastic axis.

$\frac{P_4}{S_4} = T_4 = \frac{X_{T_i} - X_{ref}}{\Delta r_n}$: Chordwise lever for location of the tension axis at the inboard end of the cell. (See diagram on page 364.)

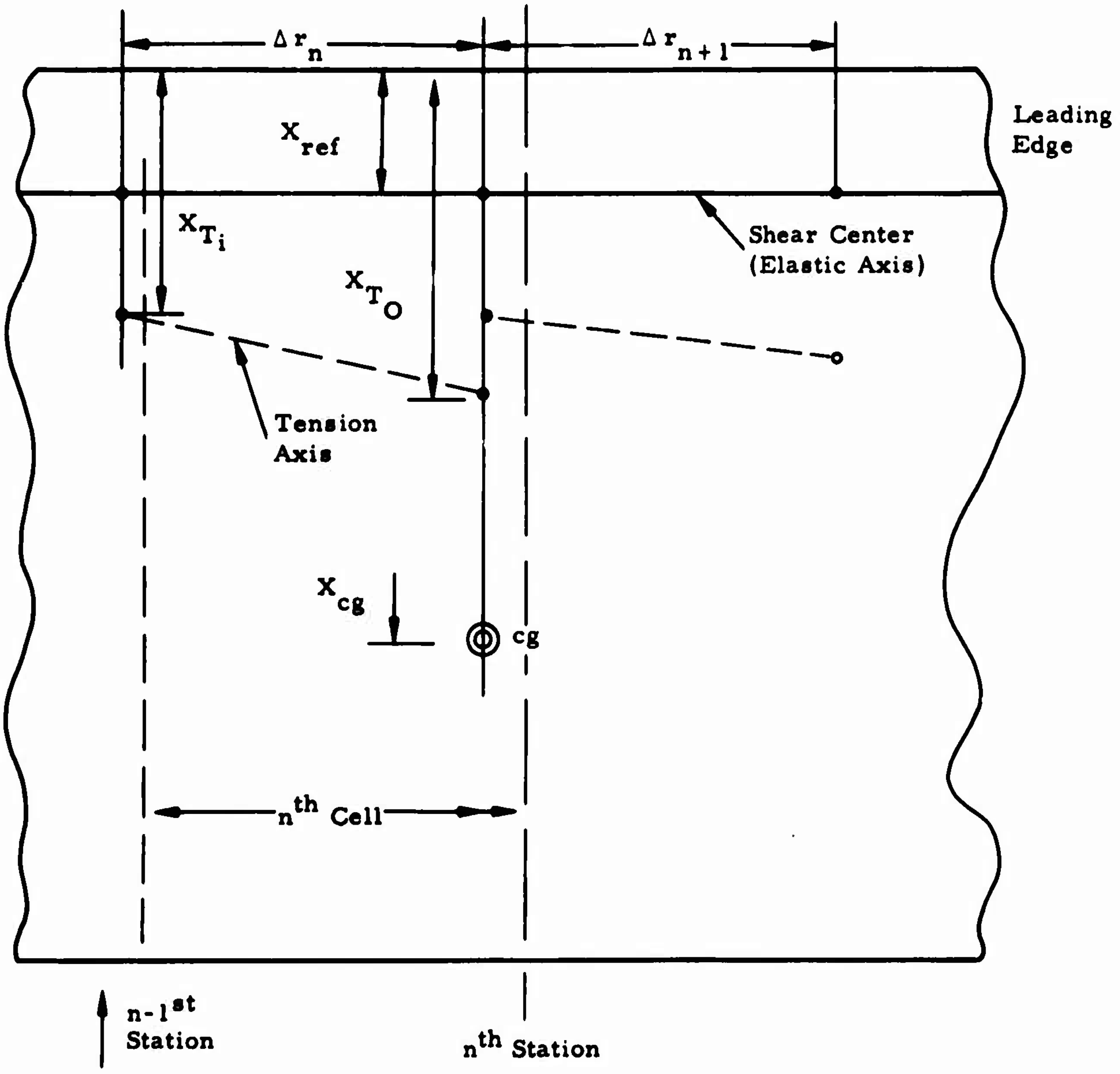
$\frac{P_5}{S_5} = T_5 = \frac{X_{T_O} - X_{ref}}{\Delta r_n}$: Chordwise lever for location of the tension axis at the outboard end of the cell.

$\frac{P_6}{S_6} = T_6 = \tan \theta_0$: Elastic coupling between vertical and in-plane bending due to rotation of the blade principal axes about the pitch axis.

$\frac{P_7}{S_7} = T_7 = \sin(\Delta a_n)$: Change in direction of the $\zeta$ coordinate due to spanwise increment in static blade coning.

## 3.4 Definitions of Mechanical Qualities

### Geometrical Qualities



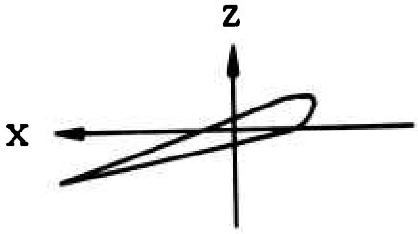
## Other Terms

1. $EI_f$ : Flapwise bending stiffness

2. $EI_c$ : Chordwise bending stiffness

3. $GJ$ : Torsional stiffness

4. $I_Z$ and $I_X$



$I_Z$ = Mass moment of inertia through cg about vertical axis

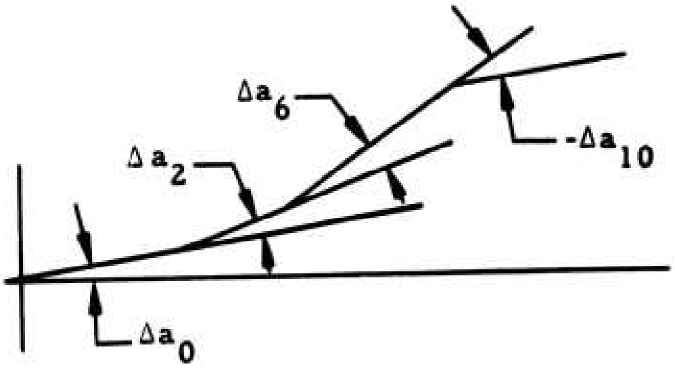
$I_X$ = Mass moment of inertia through cg about horizontal axis

For flat blades: $I_Z - I_X \simeq I_p \cos(2\theta_0)$

where $I_p$ = Mass moment of inertia of blade about cg

5. $\theta_0$ = Local pitch angle of blade relative to cone of rotation

6. $\Delta a_n$ = Change of (static) spanwise blade slope. $\Delta a_n$ may be changed at about 4 points. For example:



7. $T_n$ : Tension in blade

$$T_n = \Omega^2 \sum_{i=n}^{N} m_i r_i$$

where

$m_i$ = mass at ith station

$r_i$ = radius from axis to ith station

$\Omega$ = rotation frequency

N = last station (at tip)

NOTE: It is important that the number and location of stations used in computing $T_n$ be identical to those used in the idealized model.

8. $X_{T_i}$ : Inboard tension axis

$X_{T_O}$ : Outboard tension axis

$$X_{T_i} = - \frac{M_{\zeta_i}}{T_n} \qquad X_{T_O} = - \frac{M_{\zeta_O}}{T_n}$$

$M_{\zeta_i}$ is the (static) chordwise bending moment about the reference axis just outboard of the n-1st station.

$M_{\zeta_O}$ is the (static) chordwise bending moment about the reference axis just inboard of the nth station.

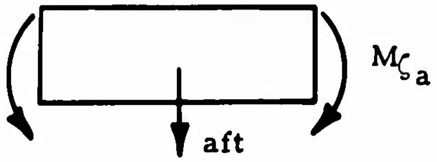
Formulas for computing $X_{T_i}$ and $X_{T_O}$:

$$X_{T_i} = X_{fa} + r_{n-1} \frac{\sum_{i=n}^{N} m_i (X_{cgi} - X_{fa})}{\sum_{i=n}^{N} m_i r_i} - \frac{M_{\zeta_{a,\, n-1}}}{T_n}$$

$$X_{T_O} = X_{fa} + r_n \frac{\sum_{i=n}^{N} m_i (X_{cgi} - X_{fa})}{\sum_{i=n}^{N} m_i r_i} - \frac{M_{\zeta_{a,\, n}}}{T_n}$$

where $X_{fa}$ = location of blade feathering axis, that is, the axis that passes through the center of rotation.

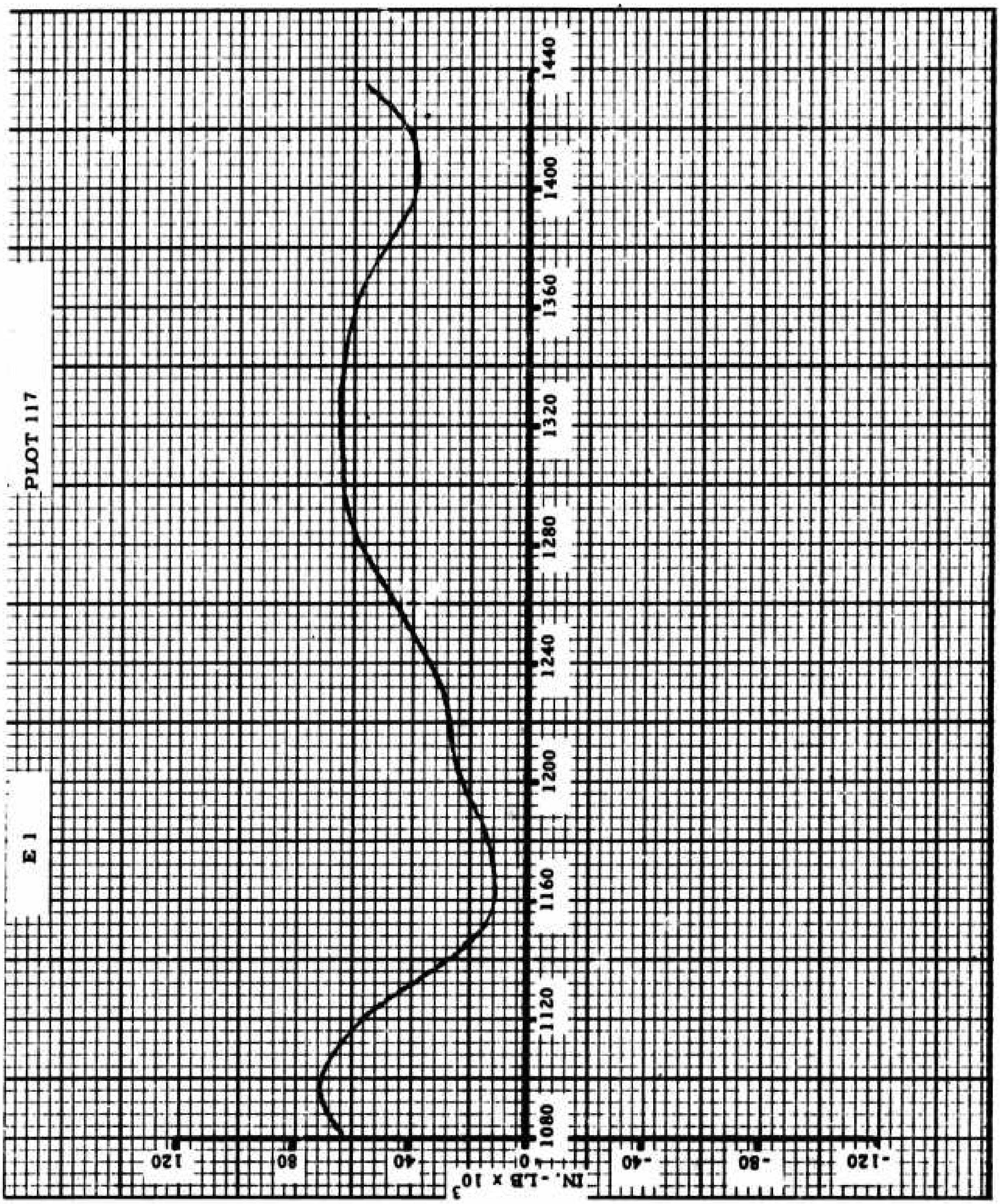
$M_{\zeta_{a,n}}$ is the (static) chordwise bending moment due to aerodynamic drag about the reference axis. at the nth station.



NOTE: It is important that the number and location of stations used in computing $M_{\zeta_i}$ and $M_{\zeta_O}$ be identical to those used in the idealized model.

# APPENDIX VI
## TYPICAL SAMPLES OF COUPLED ANALYSIS UNMODIFIED COMPUTER OUTPUT

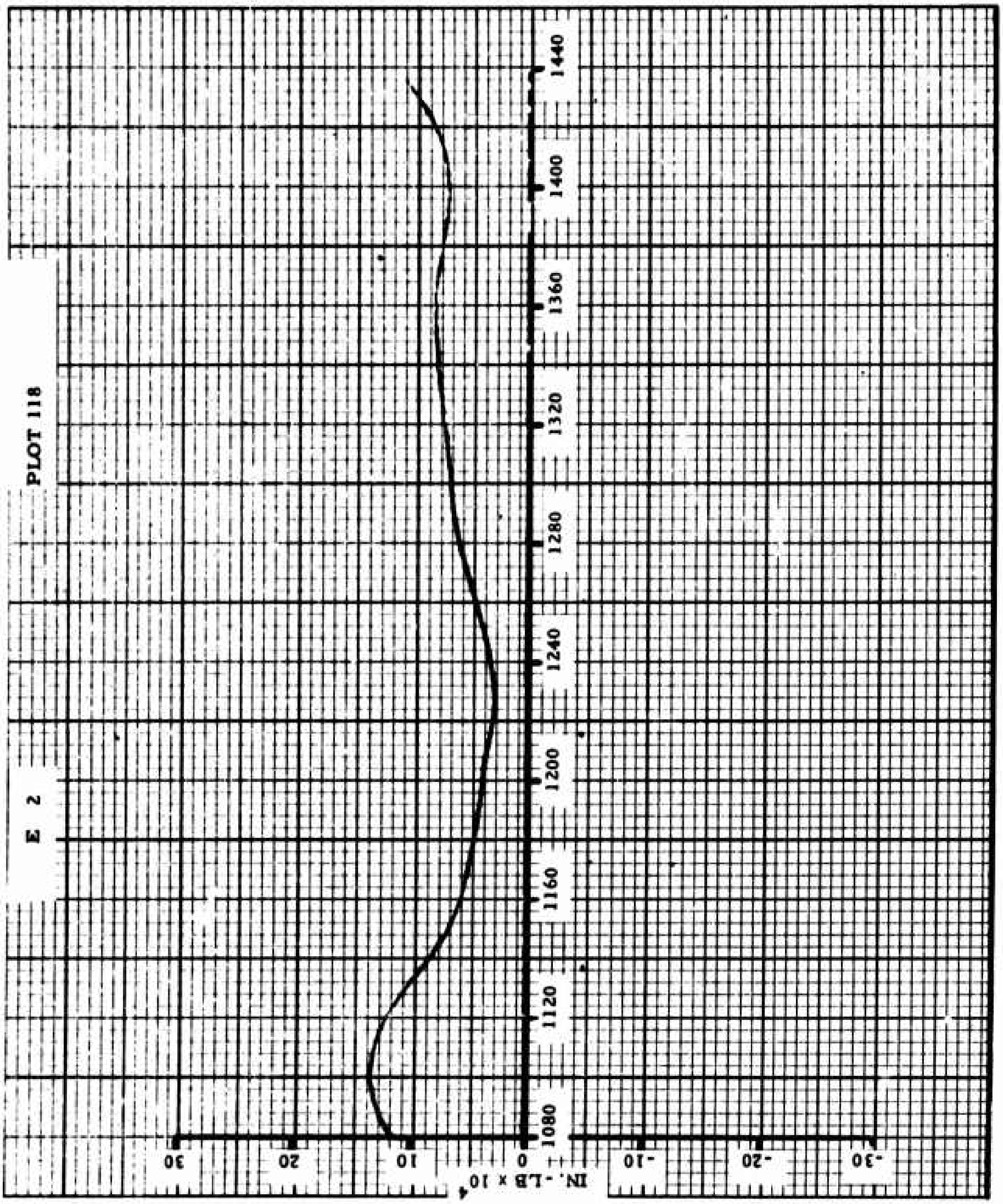
This appendix presents six typical unmodified computer output plots of loads versus azimuth angle (in degrees) for a series of blade stations. Spanwise plots presented as Figures 51 through 54 in the section of this report titled Fully-Coupled Blade Response and Dynamic Stability Analysis Using SADSAM IV were derived from these computer plots. The flight condition represented is level, unaccelerated cruise at 110 knots, 675-fps tip speed, 60,000-pound gross weight, and sea level standard atmosphere. Plots are numbered the same as the elastic elements (K's) in the Connection Diagram - Heavy-Lift Helicopter Blade Structure in Appendix IV, page 345. For example, plot E-69 is the load element K-69 (total vertical force per blade), and plot E-28 is the moment in element K-28 (blade root torsion).

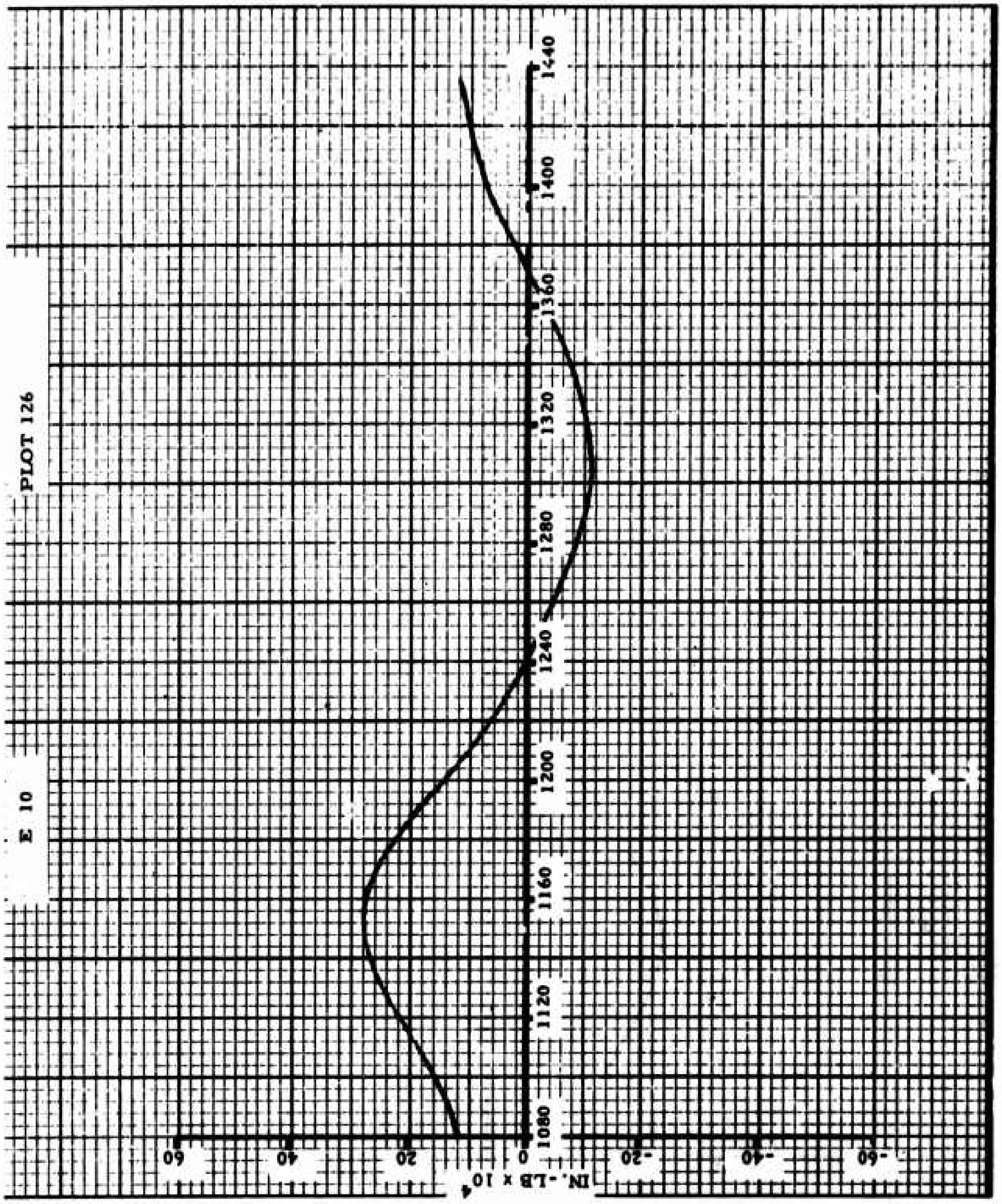


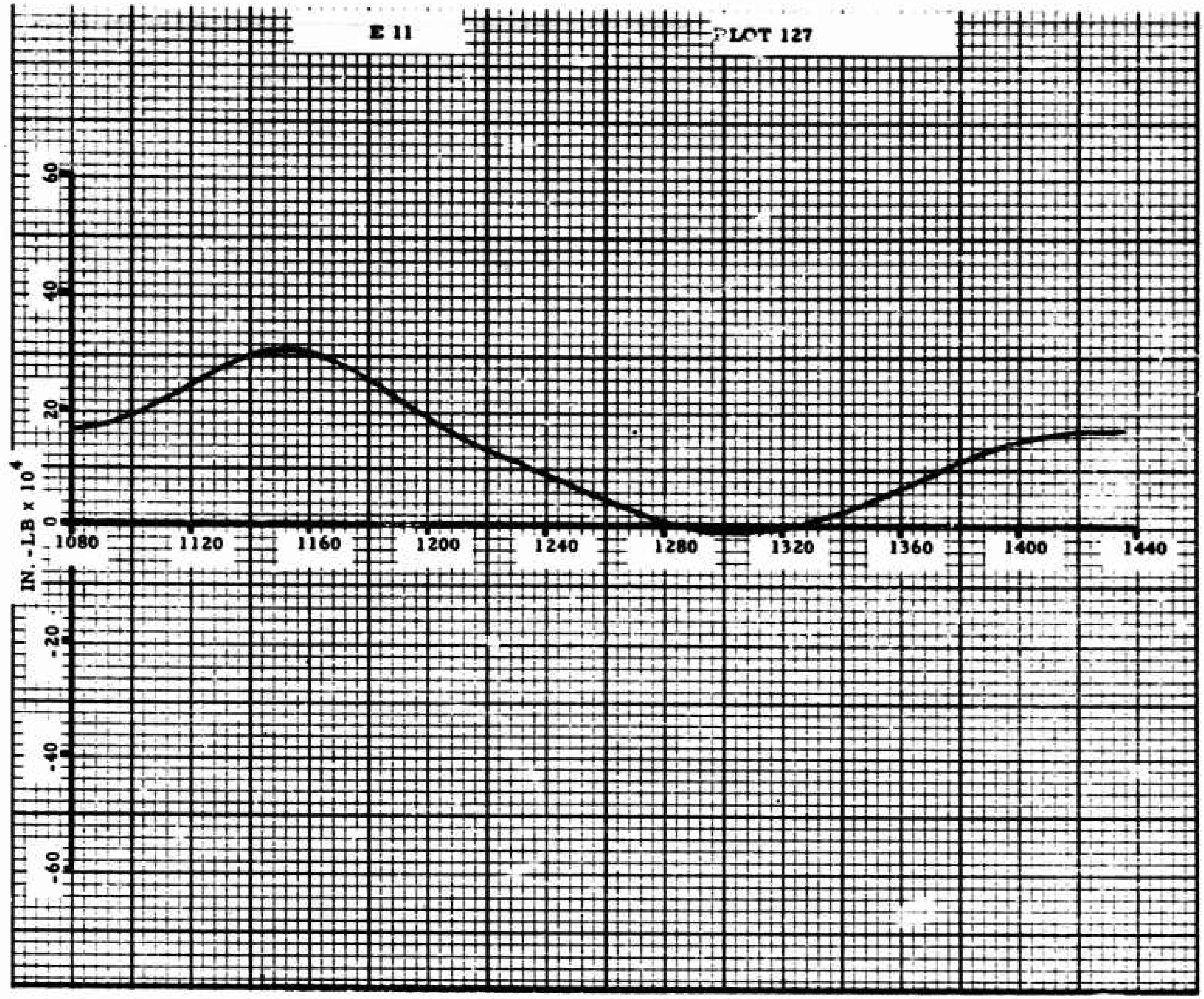
PLOT 117

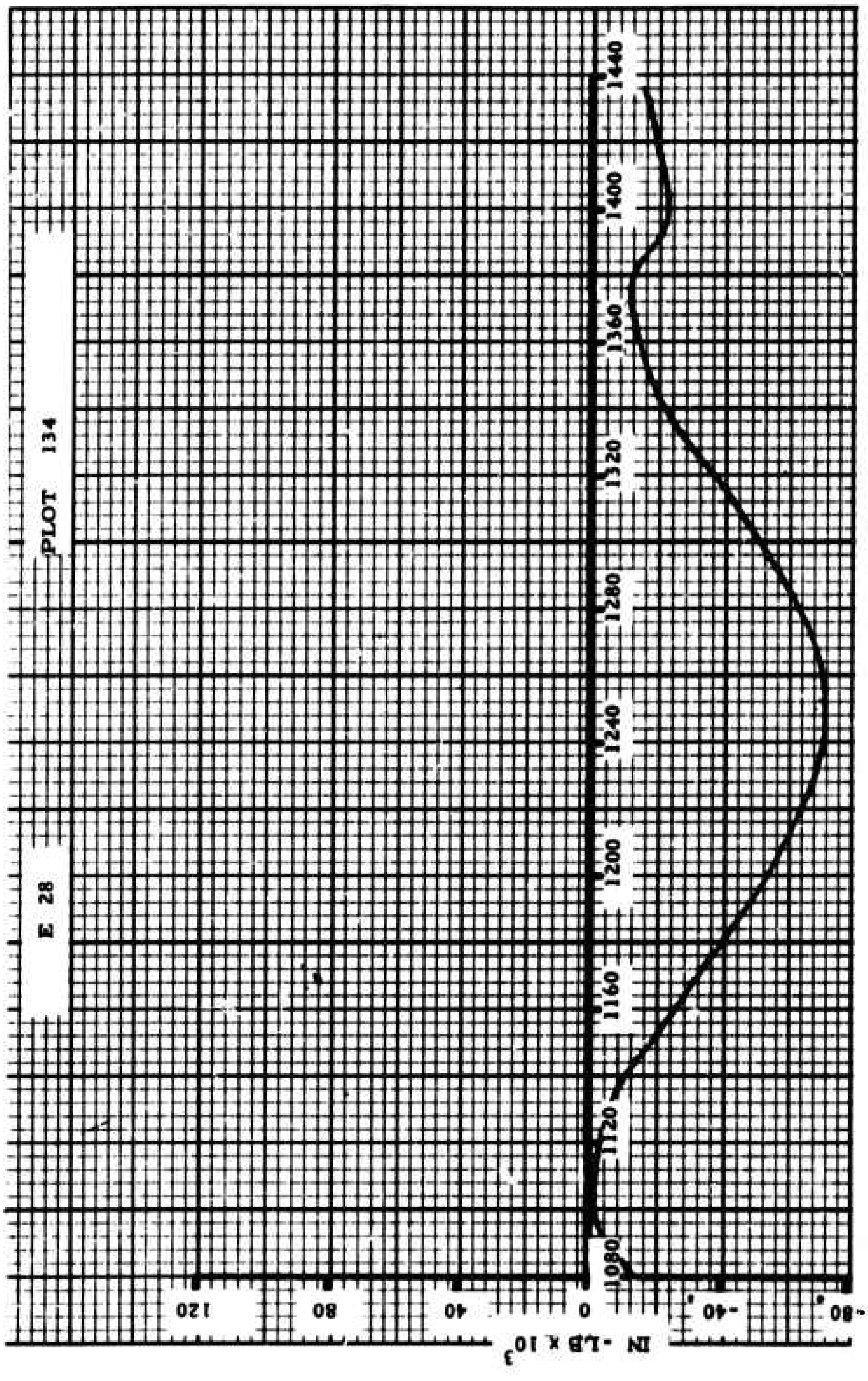
E 1

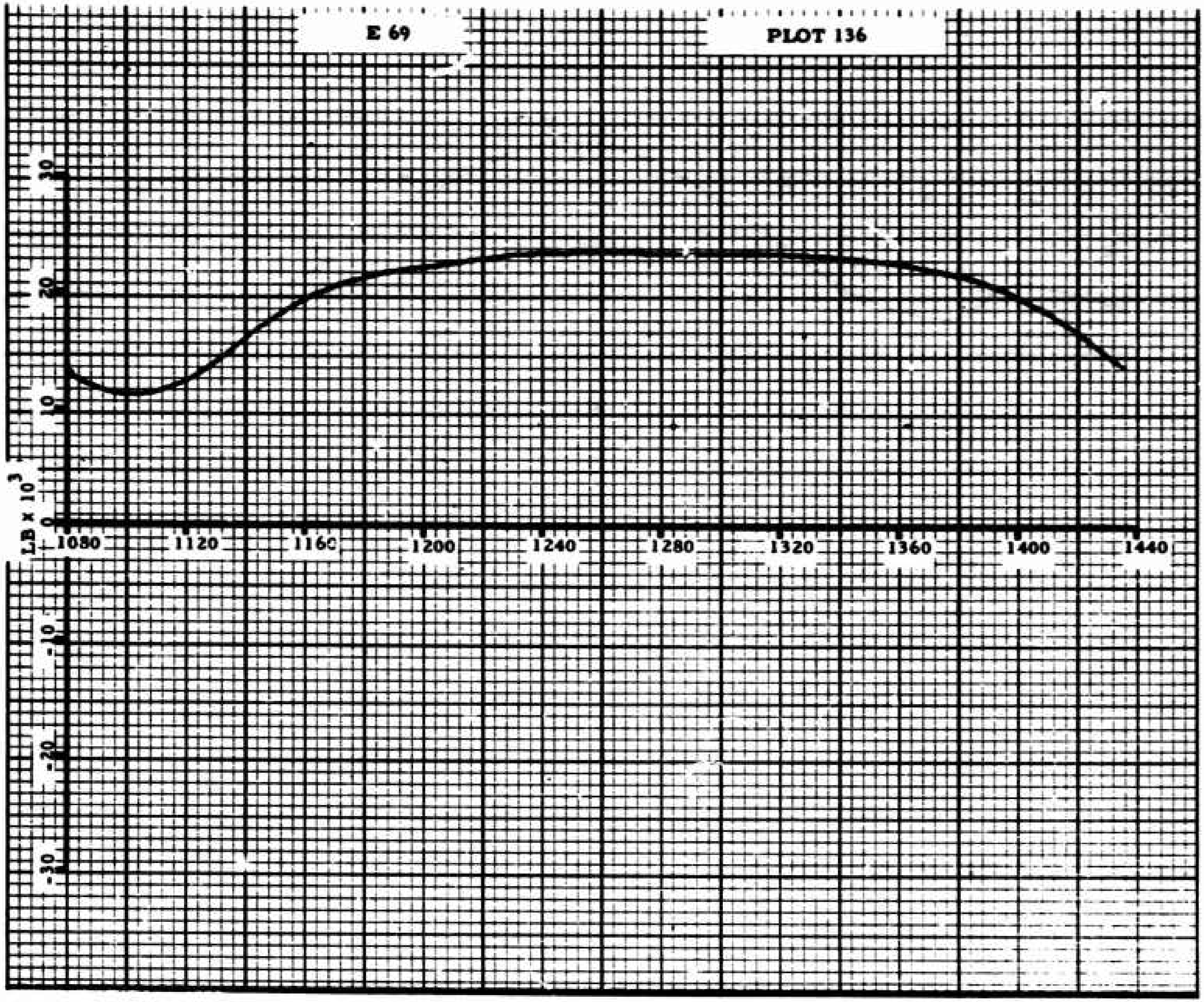
1080
1120
1160
1200
1240
1280
1320
1360
1400
1440
120
80
40
0
-40
-80
-120
IN. - LB x $10^3$


PLOT 118
E 2
1080
1120
1160
1200
1240
1280
1320
1360
1400
1440
30
20
10
0
-10
-20
-30
IN. -LB × 10⁴


PLOT 126
E 10
1080
1120
1160
1200
1240
1280
1320
1360
1400
1440
60
40
20
0
-20
-40
-60
IN.-LB x 10^4


E 11
PLOT 127
IN.-LB x 10$^4$
60
40
20
0
-20
-40
-60
1080
1120
1160
1200
1240
1280
1320
1360
1400
1440


E 28
PLOT 134
1080
1120
1160
1200
1240
1280
1320
1360
1400
1440
120
80
40
0
-40
-80
IN - LB x $10^3$


E 69
PLOT 136
LB x $10^3$
30
20
10
0
-10
-20
-30
1080
1120
1160
1200
1240
1280
1320
1360
1400
1440

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In a preliminary design study of a rotor system for a Hot Cycle heavy-lift helicopter, the following items were accomplished. (1) An analytical procedure was developed that permitted calculation of fully coupled blade response and dynamic stability characteristics; (2) parametric and configuration studies involving basic characteristics of the rotor system were conducted; (3) design layouts, structural design studies, and detailed weight analyses were made (design and analysis were limited to the integrated lift-propulsion system with emphasis on the rotor system); (4) preliminary design was completed, and performance of the optimum rotor for the heavy-lift mission requirements was determined; (5) a fully coupled blade loads analysis of the optimum rotor was made; and (6) a full-scale mockup of the rotor hub area was constructed.

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