# AD NUMBER

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DETAILED FINAL REPORT ON RESEARCH ON
HIGH-SPEED ROTARY-FOIL HELICOPTER

VOLUME IV

SAFETY AIRCRAFT PERFORMANCE DATA

OFFICE OF NAVAL RESEARCH, WASHINGTON, WASHINGTON PROJECT NR. 250-CC-1
CONTRACT NER5-250-1

Report 1904-A

20 December 1950

Serial 16

CONFIDENTIAL
DETAILED FINAL REPORT OF RESEARCH ON
HIGH SPEED ROTARY-FIXED WING AIRCRAFT

VOLUME IV

SAMPLE AIRCRAFT PERFORMANCE DATA

SUBMITTED UNDER Contract N9onr-84901 to the Office of Naval Research,
Amphibious Branch, Project NR 250-001

PREPARED BY R.C. Snyder  H.N. Heek  APPROVED BY K. H. Hohenemser
R.G. Snyder  A.N. Hook

APPROVED BY

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**Basic Assault Mission**

**NORMAL GROSS WEIGHT**

- **Cruise at 220 knots**
  - 27.3 min.
  - To carrier @ S.L.
  - 207.5 gal.

- **Reserve at end of mission**
  - 10% initial fuel load ∗
  - 63.5 gal.

- **Warm up and take off at normal power**
  - 10 min.
  - (Includes 2 min. hovering)
  - 106.5 gal.

- **Cruise at 220 knots**
  - At sea level to airhead
  - 27.3 min.
  - Airhead
  - 207.5 gal.

- **Airhead landing to deposit 36 troops or 6,015 pounds payload**
  - Take off empty or return full payload
  - 5 min.
  - 42 gal.

**Total fuel**

- 627 gal.

**Total mission time**

- 70 min.

**Radius of action**

- 100 nautical miles
6.7 Autorotation Characteristics 36
7. Power Available 37
8. Performance 38

8.1 Sample Calculations 39
8.2 Performance Curves 40
9. References 62
1. SUMMARY

The preliminary estimated performance data characteristics are presented for a rotorcraft of advanced design that fulfills or exceeds the specified requirements for an assault helicopter. This helicopter, designated the Model 78, is propelled by a rotor for take-off, hovering, and slow translational flight, and by propellers for cruise and high-speed flight. For rotor-propelled flight, a pressure-jet rotor system and conventional helicopter controls are utilized. For high-speed flight, the major portion of the aircraft weight is supported by a small fixed-wing surface with the lightly loaded rotor in low-rotor autorotation. Two gas turbine-driven propellers and conventional airplane controls provide propulsion and control.

The vertical and high-speed flight characteristics and high payload of Model 78 are readily adapted to an assault mission. At the maximum level flight speed of 240 knots and an 8135-pound payload (36 troops), thirty-three troops per hour per aircraft can be transported to an airhead, as compared to the 9.7 troops per hour per aircraft just meeting the assault specifications. Therefore, on the first wave, the Model 78 is capable of performing the work of 1.8 aircraft which just meet the assault specification, or on a shuttle basis, is the equivalent of 3.4 such aircraft.

All performance estimates are based upon proven methods of analysis developed by the NACA, or upon wind tunnel model test data obtained in a twenty-month research program under contract to the Office of Naval Research. Much of these test data have shown substantial agreement with data from previous test programs of the NACA and with McDonnell theoretical analyses.
2. INTRODUCTION

The McDonnell Aircraft Corporation presents the preliminary aerodynamic performance estimate for a rotorcraft of advanced design. This aircraft, designated the Model 78, has the cruising speed of an airplane, the lifting capacity of a jet rotor and the ability to load either troops or cargo at any selected point. The design principle is based upon the finding that lifting rotors, are not required to deliver the entire lifting or propulsive force of the aircraft, may advance at far higher speeds than heretofore considered possible. This principle has been confirmed as a result of twenty months of research conducted under contract to the Office of Naval Research. Rotor lift, drag, blade motions, blade stresses, wing interference, aircraft stability, and many other details have been analyzed and tested through a wide range of variables.

The Model 78 incorporates a single lifting rotor with pressure-jet drive, a relatively small sized wing to unload the rotor at high speeds, a conventional empennage for aircraft stability, a twin-engine installation driving variable-pitch propellers and two axial low compressors for rotor propulsion, and side-by-side seating for pilot and copilot. The twin-engine design using available gas turbines and compressors (Allison 501 and Garrett AiResearch 10-Ks respectively) offers reliability and greatly improved performance over that of conventional helicopters. Since the rotor is stationary in forward flight and rotor power is required only for short periods of hovering and acceleration, it is possible to use a jet drive without appreciable penalty from its relatively high fuel consumption. Of the jet rotor drives available, the pressure-jet rotor is the most suitable because of its lower fuel consumption, easier starting and high power, its high
lifting capacity and its ability to fulfill the autorotational requirements at high forward speeds.

For hovering and slow forward flight, the aircraft is flown by rotor propulsion utilizing the pressure-jet power source derived from turbine-driven compressors and conventional single rotor control (i.e., vertical control by collective pitch variation and yaw control by cyclic pitch variation). For high-speed flight, in which the major portion of the weight is supported by a fixed wing and in which the lightly loaded rotor is overspeeded, propulsion is obtained from two gas turbine-driven propellers and control is by conventional airplaneaileron-elevator-rudder systems. The transition from rotor-driven to propeller-driven flight is performed at nearly constant altitude by shifting from pressure-jet power to propeller power with the intermediate power being supplied by the residual rotor kinetic energy and a change in velocity kinetic energy.

Although Model 78 is designed to producely-considered practical values of blade tip speed and maximum advance ratio in order to guarantee its immediate usefulness in military operation, rotor model tests conducted up to an advance ratio of 2.5 have shown that, even for each number of the advancing blade less than 0.5, flight speeds over 200 knots may be attained in the future. The most surprising result of these model tests, confirmed by theoretical studies, was the increase of aerodynamic efficiency with increasing advance ratio. A lift to drag ratio of the autorotating model rotor (excluding hub) of 11.5 was measured at an advance ratio of 2.3. This indicates full-scale lift to drag ratios of the same order of magnitude as those for a fixed wing. A number of problems pertaining to rotor control, blade motions and blade stresses have to be studied prior to the
utilization of tip speeds and advance ratios very much in excess of those used in the normal operation of model 78.

The preliminary performance estimates for the model 78 are based upon wind tunnel model test data, obtained in a research program sponsored by the Office of Naval Research, and in the conventional helicopter or rotor propulsion advance ratio range, upon proven methods of analysis.
3.1 Summary Performance Tables and Figures

<table>
<thead>
<tr>
<th>Description</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Take-off weight</td>
<td>30,000 pounds</td>
</tr>
<tr>
<td>Fuel</td>
<td>3750 pounds</td>
</tr>
<tr>
<td>Payload</td>
<td>4135 pounds</td>
</tr>
<tr>
<td>Engine power (normal ratio) *</td>
<td>3870/14000 Bhp/hr</td>
</tr>
<tr>
<td>Disc loading (ft/lb)</td>
<td>4.4 lbs./ft-lb</td>
</tr>
<tr>
<td>Power loading</td>
<td>7.75 lbs./ft-lb</td>
</tr>
<tr>
<td>Maximum speed - sea level</td>
<td>240 knots</td>
</tr>
<tr>
<td>Rate of climb - sea level</td>
<td></td>
</tr>
<tr>
<td>Rotor propulsion</td>
<td>3030 ft./min.</td>
</tr>
<tr>
<td>Propeller propulsion</td>
<td>1950 ft./min.</td>
</tr>
<tr>
<td>Time to 5000 ft</td>
<td></td>
</tr>
<tr>
<td>Rotor propulsion</td>
<td>1.76 min.</td>
</tr>
<tr>
<td>Propeller propulsion</td>
<td>2.94 min.</td>
</tr>
<tr>
<td>Time to 10,000 feet</td>
<td></td>
</tr>
<tr>
<td>Rotor propulsion</td>
<td>4.48 min.</td>
</tr>
<tr>
<td>Propeller propulsion</td>
<td>6.51 min.</td>
</tr>
<tr>
<td>Vertical rate of climb - sea level</td>
<td>3540 ft./min.</td>
</tr>
<tr>
<td>Absolute hover (stall limitation)</td>
<td>10,000 feet</td>
</tr>
<tr>
<td>Combat radius/Average velocity</td>
<td>100 m./220 knots</td>
</tr>
<tr>
<td>Maximum endurance/Average velocity</td>
<td>1.28 hrs./200 knots</td>
</tr>
<tr>
<td>Ferry range (1880 gal. fuel)</td>
<td>776 nautical miles</td>
</tr>
</tbody>
</table>

* Power available, considering losses
Figure 1

Level Flight Performance

HORSEPOWER REQUIRED VS VELOCITY

- ADVANCING BLADE MACH NUMBER LIMITATION
- WING STALL LIMITATION
- BLADE STALL LIMITATION

HORSEPOWER REQUIRED

VELOCITY - KNOTS

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FIGURE - 3

MODEL 78

PAYLOAD vs. COMBAT RADIUS

ALTITUDES - SEA LEVEL
CRUISE AT 220 KNOTS

- OVERLOAD
- GROSS WEIGHT (50,000 LBS)
- DESIGN-TANK CAPACITY (677 GALL.)

- NORMAL
- GROSS WEIGHT (35,000 LBS)

- 15 MINUTES @ N.R.P.
- 13 MIN. WARM-UP, TAKE-OFF
- 2 MIN. Hovering
- 5 PERCENT increase in SPEC. SFC
- 10 PERCENT RESERVE
- WEIGHT OF OVERLOAD TANKS = 1/2 LBS GALL. FUEL
LEVEL FLIGHT PERFORMANCE

AVERAGE SPEED vs COMBAT RADIUS

NORMAL GROSS WEIGHT
ALTITUDE - SEA LEVEL

MAXIMUM SPEED

AVERAGE SPEED - KNOTS

COMBAT RADIUS - NAUTICAL MILES

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4. DISCUSSION

The principle of a combined rotary-fixed wing aircraft was studied to practice in the early days of the autogyro, and an extensive flight test program has been conducted with such aircraft by the NACA (references 311 and 316). This program included conditions up to an advance ratio of the lifting rotor of 0.7 and up to a load on the fixed wing of 35% of the total aircraft weight. The aircraft tested by the NACA was controlled by conventional aileron, elevator, and rudder controls with no means provided to change the relative attitude of the wing and lifting rotor or the blade pitch angle in flight. The main conclusions from these tests were that a wide variation of rotor speed as a function of airspeed may be obtained by suitable adjustments of the relative wing and rotor attitude (which were made on the ground during the test program) and that the interference of the wing on the lifting rotor is negligible in the tested range.

As compared to this early version of rotor-fixed wing aircraft, model 75 incorporates the following additional features: a rotor attitude control, longitudinally and laterally; a collective blade pitch control; a jet rotor drive for vertical take-off and forward acceleration up to 110 knots. In rotor-propelled, or pressure-jet flight, which is possible between zero and 110 knots, the aircraft is controlled by the longitudinal and lateral rotor attitude control with the fixed surface controls relatively ineffective. In propeller propulsion, or turbo-prop flight, which is possible between 20 and the limiting speed of 300 knots, the aircraft is controlled by the fixed surface controls. The rotor lateral attitude control is still connected to the control stick, though relatively ineffective, while the rotor longitudinal attitude control is disconnected from
the control stick and an automatic rotor altitude control is incorporated to achieve rotor speed stability.

The design features of the engine are selected to insures immediate usefulness in military operation. Available power plants, compressors, and practical limits to torque advance ratio, rotor diameter, etc., are used to guarantee such operation. Though the gas turbine has, when operating at the lower altitudes, poor throttle, and on very summer days, a higher fuel consumption than a reciprocating engine, the saving in weight and in aerodynamic drag for offsets these disadvantages for the relatively short range that is required for an assault aircraft. Gas turbine development to be expected during the prototype design stage of this aircraft should force or enhance their selection.
5. TABLED DATA

5.1 Notation and Symbols -

\[\begin{align*}
\alpha &= \text{Lift curve slope} \\
AR &= \text{Aspect ratio} \\
A &= \text{Area, square feet} \\
N &= \text{Number of blades} \\
A_s &= \text{Span, feet} \\
c &= \text{Chord, feet} \\
C_T &= \text{Rotor thrust coefficient} = \frac{1}{\rho \pi R^2 \Omega^2} \\
C_T \sigma &= \text{Aerodynamic blade loading} \\
C_{L_R} &= \text{Rotor lift coefficient} = \frac{L_R}{\rho \pi R^2 \Omega^2} \\
C_{L_W} &= \text{Fixed wing lift coefficient} = \frac{L_W}{\rho \pi R^2 \Omega^2} \\
C_Q &= \text{Rotor torque coefficient} = \frac{2}{\rho \pi R^2 \Omega^2} \\
C_{V_L} &= \text{Equivalent drag-lift ratio} \\
f &= \text{Parasite drag area, square feet} \\
N &= \text{Pressure-jet thrust, pounds} \\
K_v &= \text{Ratio} \quad \frac{\text{Excess power}}{\text{Effective vertical climb power}} \\
L &= \text{Total lift force, pounds} \\
L_R &= \text{Rotor lift force, pounds} \\
L_W &= \text{Fixed wing lift force, pounds} \\
L/D &= \text{Lift-drag ratio}
\end{align*}\]
Q = Rotor torque, foot-pounds
q = Dynamic pressure, pounds/square foot
d = Rotor radius, feet

\( \frac{h}{c} \) = Maximum rate of climb, feet/minute
T = Rotor thrust, pounds
\( \frac{T}{Y} \) = Hovering merit factor

\( v_i \) = Rotor induced velocity, feet/second
\( V \) = Flight path velocity, feet/second
\( V_v \) = Vertical rate of climb, feet/second
\( \mu \) = Advance ratio, \( \frac{V}{\Omega} \)
\( \Omega \) = Rotor angular velocity, radians/second
\( \rho \) = Air density, slugs/cubic feet
\( \sigma \) = Rotor solidity, \( \frac{b_c}{R} \)
\( \theta \) = Rotor blade angle, degrees
\( \alpha \) = Retracting blade tip angle of attack, degrees

**Subscripts**

i = Induced
J = Tip jet
o = Profile
P = Parasite
R = Rotor
W = Fixed-wing
5.2 Dimensional Data -

5.2.1 Fixed Wing:

Span, inches .................................................. 540
Chord, inches:  
  root ...................................................... 97.75
  tip ....................................................... 57
  MAC ......................................................... 90
Projected area, sq.ft. ........................................ 332
Airfoil section - root ........................................ 23018
  tip ...................................................... 23012
Incidence, degrees ........................................... 3
Effective aspect ratio ........................................ 6.1
Aileron area, sq.ft. .......................................... 21.50
Split-flap area, sq.ft. ....................................... 18.00

5.2.2 Rotors:

Number of rotors ............................................. 1
Number of blades per rotor ................................ 3
Rotor diameter, feet ......................................... 65
Rotor disc area, sq.ft. ....................................... 3320
Disc loading, lbs./ft.² ...................................... 3.04
Rotor solidity .................................................. .09
Blade chord, inches .......................................... 37
Blade twist, degrees ......................................... 0
Blade airfoil section ......................................... MACA 23015

Rotor tip speed, ft./sec. -
  Hovering ................................................... 750
  Helicopter forward flight ................................ 750, 700
  Propeller forward flight ................................ 700, 600, 420
### Aircraft Specifications (7-type)

<table>
<thead>
<tr>
<th>Section</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing section - root</td>
<td>NACA 0012</td>
</tr>
<tr>
<td>Wing section - tip</td>
<td>AOA 000</td>
</tr>
<tr>
<td>Effective aspect ratio</td>
<td>4.85</td>
</tr>
<tr>
<td>Maximum deflection</td>
<td>45</td>
</tr>
<tr>
<td>Normal incidence, °</td>
<td>0</td>
</tr>
<tr>
<td>Low aerodynamic chord, inches</td>
<td>52.25</td>
</tr>
<tr>
<td>Total area, sq.ft.</td>
<td>142.0</td>
</tr>
<tr>
<td>Center surface area, sq.ft.</td>
<td>44.0</td>
</tr>
</tbody>
</table>

### Propellers

| Number of propellers          | 2              |
| Number of blades per propeller| 3              |
| Manufacturer                  | Aero Products  |
| Model designation             | A 65F          |
| Propeller Diameter, Feet      | 10             |
| Activity factor               | 400            |
| Propeller gear ratio          | 7.96:1         |
| Propeller speed, rpm          | 1780           |

### Weight Data

<table>
<thead>
<tr>
<th>Item</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Normal empty weight, pounds</td>
<td>30,000</td>
</tr>
<tr>
<td>Weight empty</td>
<td>12,004</td>
</tr>
<tr>
<td>As full load</td>
<td>15,946</td>
</tr>
<tr>
<td>Fuel load</td>
<td>9,135</td>
</tr>
</tbody>
</table>
5.4.2 Maximum T.O. weight, pounds .............. 36,000
Empty weight ................................ 10,964
Useful load .................................... 14,048
Payload ........................................ 11,138

5.4 Power Plant Data *

5.4.1 Engine Data

Number of engines ......................... 2
Manufacturer ................................ Allison division, General Motors
Model designation ......................... Allison Model 501 power section

Engine ratings -
Specification normal rating ............... 2235/14000
Performance normal rating ** ............. 1535/14000

5.4.2 Compressor Data

Manufacturer ................................ Westinghouse
Model designation ....................... 19X3

5.4.3 Pressure-Jet Data

Manufacturer ................................ McDonnell Aircraft Corporation

* For more complete Power Plant data, see reference 9.

** Includes losses for inlets, ducts, etc., see reference 9.
6. AERO Dynamic DATA

6.1 Parasite drag estimate - For estimating parasite drag power losses, a breakdown of the component parasite drag areas is made. The following tables of component parasite drag areas were prepared to cover the requirements of the performance estimate. The component drag coefficients were obtained from reference 9. Also, note that the wing is not considered in Table I, since the drag effect of the wing in forward flight is included in the L/D of the wing. Also, the hub effect is treated as a separate component in the rotor-powered flight performance, and for turbo-prop flight, the hub drag is contained in the L/D for the rotor.

<table>
<thead>
<tr>
<th>Component</th>
<th>Area (sq.ft.)</th>
<th>$C_D$</th>
<th>$F$ (sq.ft.)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuselage</td>
<td>68.5</td>
<td>.11</td>
<td>7.54</td>
</tr>
<tr>
<td>Pylon</td>
<td>26.5</td>
<td>.016</td>
<td>.42</td>
</tr>
<tr>
<td>Nacelles</td>
<td>29.5</td>
<td>.10</td>
<td>2.95</td>
</tr>
<tr>
<td>Empennage</td>
<td>120.0</td>
<td>.012</td>
<td>1.44</td>
</tr>
<tr>
<td>Landing gear (retracted)</td>
<td>-</td>
<td>-</td>
<td>.35</td>
</tr>
<tr>
<td>Interference (10% assumed)</td>
<td>-</td>
<td>-</td>
<td>1.50</td>
</tr>
<tr>
<td>Hub (based on disc area, $C_D$ from model test data)</td>
<td>-</td>
<td>.0013</td>
<td>4.32</td>
</tr>
</tbody>
</table>

Total                                                   18.92 **

- Turbo-prop flight (wing drag, induced and profile, included in L/D wing; hub drag, included in L/D of rotor).
- Powered rotor flight (wing drag, induced and profile, included in L/D wing).
### TABLE II

**Parasite Drag Estimate (Vertical Flight)**

<table>
<thead>
<tr>
<th>Component</th>
<th>Area (sq.ft.)</th>
<th>C_d</th>
<th>F (sq.ft.)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuselage</td>
<td>415</td>
<td>0.35</td>
<td>145</td>
</tr>
<tr>
<td>Wing</td>
<td>218</td>
<td>1.00</td>
<td>218</td>
</tr>
<tr>
<td>Fusellos</td>
<td>166</td>
<td>0.30</td>
<td>50</td>
</tr>
<tr>
<td>N</td>
<td>66</td>
<td>1.00</td>
<td>66</td>
</tr>
<tr>
<td>Tail</td>
<td>82</td>
<td>1.00</td>
<td>82</td>
</tr>
</tbody>
</table>

Total Parasite Drag Area = 346 sq.ft.

**Hovering Downwash Area Estimate**

<table>
<thead>
<tr>
<th>Component</th>
<th>Area (sq.ft.)</th>
<th>C_d</th>
<th>F (sq.ft.)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuselage</td>
<td>245</td>
<td>0.35</td>
<td>26</td>
</tr>
<tr>
<td>Hind</td>
<td>148</td>
<td>1.20</td>
<td>148</td>
</tr>
<tr>
<td>Fusellos</td>
<td>166</td>
<td>0.30</td>
<td>50</td>
</tr>
</tbody>
</table>

Total Hovering Downwash Area = 263 sq.ft.

---

* In vertical rate of climb calculations, the total platform area is used to obtain the parasite drag load. For calculations of rates of climb at forward speeds, it is necessary to obtain the effect of parasite drag on the rotor in a vertical direction. Therefore, the ideal area of the wing is subtracted from the total platform area and considered separately. The parasite drag area resulting is 346 sq.ft. minus 362 sq.ft. which equals 214 sq.ft.

**To get the hovering power required considering the effect of rotor downwash, an estimate is made of the area in the path of the downwash velocity.**
5.2 Fixed wing characteristics - The Model 78 fixed wing of NACA 23012 to 23016 airfoil section has a 6.1 aspect ratio. The airfoil section characteristics for infinite aspect ratio obtained from reference 1.7 are corrected to the infinite aspect ratio of 6.1 by equations:

\[ \alpha_{AR} = \alpha_{\infty} + \frac{1.24 C_{Lw}}{AR} \]

\[ C_{Lw} = C_{L_{\infty}} + C_{Lw} \]

A further correction on the lift-drag ratio is made to account for tip, taper in accordance with reference 1.7. Figure 14 presents the corrected airfoil characteristics used in the aerodynamic performance estimates. The variation of lift coefficient with forward velocity - level flight is shown in Figure 10.

6.3 Propeller characteristics -

6.3.1 Primary - The primary turbo-prop installation consists of two, three-bladed full-feathering, zero ducted propellers driven by two Allison Model 501 gas turbines through a variable 1-38 gear boxes. During helicopter operation, the propeller pitch is set at that which results in little power absorption by the propellers. For a preliminary estimate, this propeller setting is assumed to absorb 80% of the available engine shaft power over turn and the helicopter flight range. (See reference 1.7.) The preliminary propeller data are presented in section 6.3.1.

6.3.2 Propeller efficiencies - The preliminary propeller efficiency curve, Figure 1, is estimated from the data presented by reference 1.7. The method is as follows:
Step 1 - Assume velocity, altitude

220 knots, sea level

Step 2 - Determine engine power, propeller speed

2390 HP, 1780 rpm

Step 3 - Compute J

\[ J = \frac{220 \times 220 \times 1.15}{1780 \times 10} = 1.27 \]

Step 4 - Compute \( C_p \)

\[ C_p = \frac{1 \times (HP/1000)}{E/\rho \left( \frac{n}{1000} \right)^2 \left( 1/10 \right)^5} \]

\[ C_p = \frac{0.5 \times 2.29}{(1/6)^3 (1)^5} = 0.210 \]

Step 5 - Determine \( X \) and \( C_{P_X} \)

\( X = 0.80 \)

\( C_{P_X} = 0.350 \)

\[ C_{P_X} = \frac{C_p}{X} = \frac{0.210}{0.80} = 0.263 \]

Step 6 - Compute \( \sqrt[3]{J/(C_p)} \)

\[ \sqrt[3]{\frac{J}{C_p}} = \frac{1.27}{(0.350)^{1/3}} = 1.27 / 1.05 = 1.22 \]

Step 7 - From chart (reference page 195) read \( \eta = 0.85 \)

Figure 15, propeller efficiency against velocity, is obtained by assuming various velocities and repeating the steps required to obtain propeller efficiency. These propeller efficiencies are used in transforming the shaft horse power and jet thrust to horsepower available for level flight performance calculations.
6.4 Flight limitations

6.4.1 Blade stall - Retreating blade stall is considered a limiting flight velocity criteria because of loss of control and objectionable vibration. ACA flight tests, reference 8.8, indicate that a retreating blade tip angle of attack of 12 degrees is the beginning of blade stall. Operation at tip angles greater than this causes increased profile power loss and objectionable vibration with loss of control occurring about 4 degrees above the initial stall angle.

Blade stall is primarily dependent upon the advance ratio and the aerodynamic blade loading \( \frac{C_T}{\sigma} \) which is a measure of the mean blade angle of attack. Figure 17 presents the relationship of initial stall \( \frac{C_T}{\sigma} \) with rotor shaft power parameter \( \frac{P}{L} \) for constant advance ratios, \( \mu^* \). A discussion of this graph and its source is presented in section 6.5.4. For the model 73, because of the aerodynamic blade loading and because of the effect of the fixed wing in forward flight, blade stall is avoided in the helicopter level flight condition, except for operation at or near \( C_{Lm} \) of fixed wing equal to zero. Other limits are more critical for the higher fixed-wing lift coefficients. In propeller flight, the increased drag losses, because of blade stall, are accounted for in the model test lift-drag ratio of the rotor; and since control is attained by a conventional aileron-elevator system, blade stall is not a limiting criteria.

6.4.2 Advancing blade velocity - An advancing blade velocity limitation is considered necessary to avoid increased power loss caused by Mach number drag divergence and objectionable vibration, fatigue, and control characteristics caused by blade lift loss and center of pressure movement. A limit of forward velocity plus rotational tip speed \( (V + \Omega R) \) of 900 feet per second is assumed,
which gives rise to a .80 Mach number at sea level. Reference 9.10 shows that
rearward shifts of center of pressure are avoided if the Mach number is limited
to this value. However, operation at higher advancing blade Mach numbers than
.80 is probably practical because of the intermediate nature of the profile.
Further wind tunnel research and full-scale flight test programs should provide
additional information on this limitation.

6.4.3 Wing still - The minimum propeller-driven flight speed is assumed to
be dictated by the maximum wing lift coefficient. Actually, this is not a physical
limit, since at these minimum speeds, the rotor is supporting sufficient
portion of the weight to maintain control. However, for analytical purposes, the
maximum wing lift coefficient is used as a minimum velocity limit.

6.5 Pressure-jet flight condition

6.5.1 Hovering - The hovering aerodynamic efficiency of a jet rotor is best
represented by the ratio of rotor thrust to jet thrust which may be written non-
dimensionally as:

\[ \frac{T}{F} = \frac{C_T}{C_Q} = \frac{\rho \pi R^2 (\Omega R)^2 R}{\rho \pi R^2 (\Omega R)^2} = \frac{C_T}{C_Q} \]

The hovering jet rotor torque requirements are the profile and the induced
torques,

\[ C_Q = C_{Q_0} + C_{Q_1} \]

For an ideally twisted rotor, the profile torque coefficient in terms of the
NASA three-term drag polar which is representative of smooth, well-contoured
Blades is by reference 9. 

\[
\Delta C_{Q_0} = \frac{\sigma C_0 + s}{\sigma} + \frac{S_{T_1}}{S_{T_0}} + 2 \frac{C_{Q_1}}{C_{Q_0}} \left[ \frac{C_{Q_1}}{C_{T_0}} \right]^2
\]

and the induced torque coefficient is,

\[
C_{Q_1} = \frac{1}{\sqrt{3}} \frac{\sqrt{y}}{B}
\]

For the model 71, it is assumed that the profile drag, thus torque, is independent of blade twist and that the induced drag is increased by percent to account for the variations from uniform inflow encountered with rectangular untwisted blades. The tip loss factor assumed is that presented by Sissingh in reference 9.13,

\[
B = 1 - \sqrt{\frac{X_T}{X_T}}
\]

Since the jet thrust presented in reference 4.9 is gross internal thrust excluding jet external drag, the hovering rotor thrust - jet thrust ratio is modified to account for the drag torque of the jet units. An equivalent parasite area of .11 square feet per blade is assumed and the T/P ratio corrected accordingly,

\[
\Delta C_Q = \frac{b_T f_d}{\sigma} \frac{\rho (\Omega b)^2}{(\Omega b)^2} - \frac{b_T f_d}{2(\pi b)^2}
\]

\[
T/P = C_Q \left( C_{Q_0} + C_{Q_1} + \Delta C_Q \right)
\]

- Figure 16 presents the variation of the rotor thrust-jet thrust ratio with aerodynamic blade loading (C_{T/\sigma}). This figure is the basis of all hovering and vertical climb performance estimates.
6.5.2 Vertical climb - In a vertical climb, the rotor handles a greater mass of air (due to climb velocity) than in hovering, and, therefore, needs to accelerate the air mass less to produce the same thrust. As a result, the induced power losses in a climb are less than those in hovering. Results of SACA tests (reference 9.12) were used to obtain the variation of the ratio of the excess horsepower to the effective climb horsepower with climb velocity (figure 16). The vertical rate of climb was calculated using this figure and the calculated excess horsepower.

\[ V_v = \frac{H_P e \times 35,000}{W} \]

The effective climb horsepower, \( H_P e \), was determined with due consideration given the increased rotor lift required to overcome fuselage - fixed wing parasite drag in vertical climb. (See sample calculation).

6.5.3 Forward flight - Helicopter steady state forward flight performance is calculated by SACA methods of analysis (reference 9.8). Individual power losses are expressed as the energy dissipated per second by an equivalent drag force moving at the translational velocity of the aircraft. The sources of power loss are the rotor profile and induced drags, the jet unit external drags, the wing profile and induced drags, and the fuselage parasite drag.

An equivalent drag balance divided by lift is the basis of all steady state flight performance calculations. This drag balance is modified to account for a portion of the total lift being carried by the fixed wing with the resulting drag-lift equation reading:

\[ L/L = L_R/L \left[ (\alpha/L_R) \cdot (D/L_R) + (D/L_R) \right] + L_w/L \left[ (\alpha/L_w) \cdot (D/L_w) \right] \]
In the helicopter flight condition, the jet external drag does not affect rotor characteristics such as blade angle and flapping coefficients, but does affect the power required, since gross internal thrust is used as jet thrust available. For these reasons, both \( \text{(D/L)_{TOT}} \) including and excluding \( \text{(D/L)_{J}} \) are calculated. (See sample calculation.) The power required is then calculated from the \( \text{(D/L)_{TOT}} \) including the jet drag-lift ratio by:

\[
\text{HP(REQ)} = \left[ \frac{\text{D/L}}{\text{TOTAL \times L \times V}} \right] \times 330
\]

The drag-lift ratios used in the total drag-lift balance are developed individually:

- **Rotor profile drag-lift ratio** \( \text{(D/L)_{0}} \) - The rotor profile drag-lift ratios for the various flight conditions are determined from the NASA charts of reference 9.8. These charts are developed for assumptions of zero twist and a profile drag polar \( (C_D = .0087 - .0216 \alpha + .40 \alpha^2) \) which is representative of smooth, accurately-contoured blades.

- **Rotor induced drag-lift ratio** \( \text{(D/L)_{i}} \) - The rotor induced drag-lift ratio is calculated by treating the rotor as a lifting wing of \( 4/\pi \) aspect ratio. Thus:

\[
\text{(D/L)_{i}} = \frac{C_{D_i}}{\text{CL}_{R}} = \frac{\text{CL}_{R}^2}{\pi AR \text{CL}_{R}} = \frac{\text{CL}_{R}}{4}
\]

- **Fuselage parasite drag-lift ratio** \( \text{(F/L)_{p}} \) - The fuselage drag-lift ratio is calculated from the estimated equivalent parasite area. (See table 1.) Thus:

\[
\text{(F/L)_{p}} = \frac{\int 1/2 \rho \text{A} \text{V}^2 \text{dA}}{W}
\]
Jet external drag-lift ratio \((D/L)J\) - The jet external drag-lift ratio is determined from an estimated cold drag coefficient of .115 which is based upon experience gained under the Air Material Command Y-99 rotor jet rotor contract. The maximum cross-sectional area of each pressure jet unit is 108 square inches. Therefore, the equivalent parasite area per unit is .11 square feet. Having established the equivalent parasite area, the jet drag-lift ratio may be determined by:

\[
(D/L)J \times L_{R} \gamma = \frac{1}{2\pi} \int_{0}^{\pi} b_{RFJ} (\Omega L + V_{Sim} \psi)^{3} d\psi
\]

which integrates to:

\[
(D/L)J = \frac{b_{RFJ}}{C_{R}^{3} \pi R^{2}} \left[ \frac{1}{\mu^{3}} + \frac{3/2\mu}{\gamma} \right]
\]

For quick estimation of the jet drag-lift ratio, non-dimensional plots of \((D/L)J\) against the reciprocal of the rotor lift coefficient for a ratio of \(b_{RFJ}/\pi R^{2}\) equal to unity are presented as figure 20 and 20a. The value read from these charts must be multiplied by the actual ratio of cold jet equivalent parasite area to rotor disc area which is .00010 for Model 79.

For rotor-powered flight, the jet unit drag has no effect on the rotor characteristics, such as blade angle, angle of attack and a) flapping, but as already stated, does affect the power required. Therefore, the \((D/L)J\) ratio is subtracted from the total \(D/L\) ratio for the determination of rotor characteristics other than power required.

Wing drag-lift ratio \((D/L)W\) - The wing drag-lift ratio is obtained from a plot of the wing airfoil characteristics (figure 18) for the flight condition assumed, i.e., at the given wing lift coefficient. The variation of wing lift
The maximum rate of climb in rotor-powered flight, it is necessary to calculate the rate of climb for several forward velocities because of the limiting criteria. The curves are obtained, one, considering power limiting, and a second curve, considering blade stall as a limiting factor. The intersection of these two curves determines the maximum rate of climb. (See Figure 10). In considering blade stall as a limit, it is convenient to obtain a plot of \( C_{L} / \sigma \) at initial stall for corresponding values of \( \beta / \lambda \) and \( \mu \). Figure 17 is such a plot and is obtained by converting the \( C_{L} / \sigma \) values at initial blade stall from the ACA chart of reference 13 to \( C_{L} / \sigma \) for various \( \mu \) and \( C_{L} \) values. This plot is used in conjunction with the assumed stall limit rotor load for checking the maximum rate of climb as shown in sample calculation 3.1.2.

The method used in constructing the rate of climb curves is the typical ACA analysis for rotor-powered flight.

\[
P/L = (\mu/L)_{o} + (\mu/L)_{a} + (\mu/L)_{d} + (\mu/L)_{w} = (\mu/L)_{c} + (\mu/L)_{w}
\]

A tail and error method is required to determine the actual operating conditions and power to be used during climb. (See sample calculations 3.1.5.1 and 3.1.5.2.)
with an autorotating rotor. Wind tunnel model test programs contracted to the
Office of Naval Research show that lightly loaded autorotating rotors may advance
at far higher advance ratios than heretofore considered practical. Through these
programs, rotor lift, drag, blade motion, blade stresses, fixed-wing interference,
aircraft stability, and many other details have been analyzed and tested through
a wide range of variables. The results of these model test programs and studies
form the basis for the Model 73 aerodynamic performance estimates in the turbo-
prop flight condition. Applicable test data are presented in figures 5, 6, 7, and 8;
for further data, see .A.O Engineering Letters, reference 9.18.

Figure 5, "Rotor Lift Coefficient Against Advance Ratio", presents a compar-
ison of McDonnell wind tunnel tests at the high advance ratios and other pertinent
test data from previous NACA programs. Figure 6 gives a mean curve used in the
performance estimates.

Figure 7, "Rotor Lift-Drag Ratio Against Advance Ratio", shows comparative
results of NACA tests with a ten-foot rotor model (reference 9.17) and with a
full-scale Sikorsky rotor (reference 9.16) and McDonnell tests with an eight-foot
rotor model, together with the results of Kordell theory. Accounting for Rey-
olds number effect, all the different test results and theoretical results are
in satisfactory agreement. Figure 8 gives the rotor lift-drag curve used in the
Model 73 preliminary performance estimates.

6.6.2 Forward Flight - Level flight power required for forward velocities
is obtained in a manner similar to that described in the section on rotor-powered
forward flight (section 6.5.3). A modified drag-lift equation is used for turbo-
prop flight. The power loss due to drag of the rotor is based on autorotational
wind tunnel data which does not separate the profile and induced losses in the rotor (see Figure 8). In these tests, which compare favorably with other wind tunnel and theoretical data, the rotor and its hub are considered as one unit. Therefore, it is only necessary to add the tip jet drag contribution to the rotor for a drag power loss due to the autorotating rotor.

It should also be noted that the parasite drag area used for the parasite drag loss differs in the rotor-powered and turbo-prop power required calculations. This difference is due to the fact that the hub drag is included with that measured for the rotor in the autorotational test data. In the rotor-powered flight, the hub drag is taken as a component of the parasite drag area for the whole ship and included with the parasite drag power loss.

The resulting drag-lift equation is as follows:

\[ \frac{D}{L} = \frac{L_r}{L} \left( \frac{D}{L_p} \right)_h + \left( \frac{D}{L_p} \right)_j + \frac{L_w}{L} \left( \frac{D}{L_w} \right)_w + \left[ \frac{C}{L} \right] \]

The power required is then calculated from the \( \frac{D}{L} \) using the following equation:

\[ hP(\epsilon, Q) = \left( \frac{D}{L} \right)_{TOTAL} \times L \times V \div 550 \]

(See sample calculation 9.1.4).

The percent load carried by the rotor throughout the flight velocity range is presented as figure 11. The dash lines represent the percent load carried by the rotor in helicopter flight, while the solid lines are for turbo-prop flight at various autorotating rotor tip speeds.
FIGURE 5

LIFT COEFFICIENT $c_l$ OF ROTOR IN AUTOROTATION

ACCORDING TO WIND TUNNEL TESTS
CONFIDENTIAL

FIGURE 6.6
LIFT COEFFICIENT C\textsubscript{l}\textsubscript{1} OF ROTOR IN AUTOROTATION
ASSUMED FOR PERFORMANCE ESTIMATES

BLADE SQUINCH: \( \alpha = -0.09 \),
BLADE PITCH ANGLE: \( \theta = 0.2 \).

\( \alpha \) ADVANCE \( \beta \) RATIO 1.0: 1.2
FIGURE 7

Assumed for Performance Estimate

McConnell Theory

Wind Tunnel Tests
NACA 10 FE Rotor
L/D = 2.85

Wind Tunnel Tests
My Experimental Rotor
L/D = 2.40

Lift to Drag Ratio in Autorotation
According to Wind Tunnel Tests & Theory
As Compared to Assumed L/D
The Performance Estimates

Blade Semi-Angle 8 = 10°
Blading Pitch Angle 8 = 2°
6.4 Acceleration of climb - The "maximum Required" rate of climb, is the point of maximum excess power; it is the difference between the "maximum rate of climb". The excess power over this rate is available for other applications.

\[
\text{Max. } / = \frac{\text{Power}}{\text{Weight}} \text{ Ft./in.}
\]

6.7 Aircraft Characteristics - The one-engine engine is not designed to operate at the limit of its operating characteristics. The maximum specific power is reduced in order to ensure the engine's ability to operate at or above its rated performance. The maximum descent rate is calculated from the equation:

\[
\frac{1}{V} = \frac{1}{X}
\]

The \( \frac{1}{X} \) values for this equation are used to determine the level flight. The required calculation for turbo-prop operation is similar. In that case, the engine is not necessarily at its maximum power. Taking the maximum Mach number and \( \frac{1}{V} \) for total power of flight at their respective values will give the velocities, the ratio of which are calculated in the above equation.
V. Atoms AVAILABLE

A detailed explanation of the performance characteristics and there are to obtain operational Apple are with a constant power, of course, in order to obtain a flat operation and to meet all control characteristics. For the preliminary performance evaluation, it is assumed that this will variation in tip speed would effect an appreciable change in the overall tip jet thrust or the jet area, and specific fuel consumption. The data taken at various reference speed and a temperature change are concerned, and the jet thrust losses, jet losses for propeller efficiency at low rpm, etc. With this in mind, jet exit losses accounted for in the performance of the engine.
1. Fundamentals

1.1 Earth pressure and frictional force effect

Altitude = sea level
Air speed = 30,000 f.p.m.
Air temperature = 2800 

Lever tip speed = 700 ft. per second
Lever solidity = .35

Lever nose radius = 40,000 ft. per lever radius

Assume covering downward load to be 1100 pounds, calculate wind velocity.
From figure 1, hovering, \( \frac{V}{R} = 11 \).

\[
\text{required jet thrust} = \frac{11500}{11} = 1045 \text{ pounds}
\]

\[
\text{required hover} = \text{power} = \frac{1045}{0.33} = 3180 \text{ HP}
\]

\[\text{1.2} \quad \text{oversea power required (considerusal):}\]

\[P = 0.002222\]

\[C_2 = \frac{811 \text{ ft}}{3500 \text{ ft} \cdot \text{lb} / \text{sec}^2} = 0.0074\]

\[C_0 = \frac{0.0074 \cdot 10^{10}}{0.0074}\]

\[C_{\text{total}} = 10.44 \text{ (from 1.1)}\]

\[\text{31150 = 2670 pounds jet thrust}\]

\[\text{required rotor HP} = \frac{2 \cdot 31150 \times 7}{0.35} = 2620 \text{ HP}\]

From figure 30, reference 1.4, the takeoff thrust available at a hover power day is 4400 pounds at \( \Omega = 700 \text{ ft} / \text{sec} \). Assuming an equal available thrust at \( \Omega = 700 \text{ ft} / \text{sec} \), the propeller available becomes

\[4400 \times 7.0 = 30800\]

\[\frac{30800 \times 10}{0.35}\]
1.1. Initial vertical climb

<table>
<thead>
<tr>
<th>Gross weight</th>
<th>30,000 lbs</th>
</tr>
</thead>
<tbody>
<tr>
<td>Motor diameter</td>
<td>8 ft. 3 in</td>
</tr>
<tr>
<td>Motor height</td>
<td>4 ft. 8 in</td>
</tr>
<tr>
<td>Max. take-off weight</td>
<td>30,000 lbs (compared to 20)</td>
</tr>
<tr>
<td>Hard take-off</td>
<td>11,000 lbs</td>
</tr>
</tbody>
</table>

\[
\text{Power level, max. climb} = \frac{P(\text{max})}{S} = \frac{30,000}{12,000} = 2.5\text{ ft/s}
\]

To allow for lesser power at lower altitudes, assume \( P = 20,000 \text { lb-ft} \), yields 2000 ft/min.
\[ \rho = \rho_0 \left( \frac{4}{\pi} \right)^{1/2} \]

\[ p^2 + q^2 = 12 \text{ in.} \]

\[ \frac{1}{3} \rho^2 \text{ in.}^4 \]

\[ \frac{1}{3} \text{ in.}^4 \]

\[ \frac{1}{3} \text{ in.}^4 \]
### Pressure-jet level flight power required

#### Altitude = Sea level

| Cross weight | 20,000 lbs. | Total parasite area | 18.32 ft.$^2$ |
| ROTOR DISC AREA | 3320 ft.$^2$ | Lined wing area | 332 ft.$^2$ |
| ROTOR TIP SPEED | 700 ft./sec. | MIN. LIFT COEFFICIENT | .5 |
| ROTOR SOLIDITY | .09 | L/S FOR WING | 20.9 |

#### Table

| $\mu$ | ASSUMED | .10 | .15 | .20 | .25 | .30 | .35 |
| $V$ | FT./SEC. | 70 | 105 | 140 | 175 | 210 | 245 |
| $V_n$ | KNOTS | 41.5 | 62.3 | 83 | 105.6 | 127.5 | 145.2 |
| $q$ | $\frac{1}{2} p V^2$ | 0.02 | 0.12 | 0.23 | 0.36 | 0.44 | 0.52 |
| $L_w$ | $C_lw x A_w x q$ | 9.65 | 2100 | 3265 | 5040 | 7700 | 11550 |
| $L_r$ | $L - L_w$ | 28035 | 27320 | 23385 | 23960 | 21300 | 18150 |
| $C_{L_r}$ | $L_r / 5320 q$ | 1.66 | .839 | .518 | .1873 | .1224 | .0727 |
| $1/C_{L_r}$ | | | .67 | 1.57 | 2.66 | 5.05 | 8.18 | 13.05 |
| $C_{p_{e}}$ | NACA CHARTS | | | | | | |
| $D_{L_{m}}^{2}$ | $C_{L_{m}} / 4$ | .1750 | .1170 | .0930 | .0840 | .0845 | .0910 |
| $D_{L_{m}}^{1}$ | $C_{L_{m}}$ | .3750 | .1800 | .0946 | .0496 | .0306 | .0192 |
| $D_{L_{m}}^{0}$ | Figure 20 & 20a | .0650 | .0475 | .0500 | .0530 | .0340 | .0360 |
| $D_{L_{m}}^{R}$ | $L_r \left( \frac{P_l}{P_s} + \frac{P_r}{P_s} + \frac{P_w}{P_s} \right)$ | .5930 | .3610 | .1872 | .1345 | .1066 | .0665 |
| $D_{L_{m}}^{R}$ | $10^{-3} \frac{q}{L}$ | .0030 | .0020 | .0110 | .0222 | .0220 | .0435 |
| $D_{L_{m}}^{R}$ | $L_r \left( \frac{1}{P_s} \right)$ | .0015 | .0035 | .0052 | .0087 | .0139 | .0190 |
| $D_{L_{m}}^{R}$ | $\frac{\left( D_{L_{m}}^{R} \right)_p + \left( D_{L_{m}}^{R} \right)_w}{\left( D_{L_{m}}^{R} \right)_p}$ | .6991 | .3125 | .2075 | .1664 | .1525 | .1510 |
| $F_j$ | $\frac{P_{L_{m}}}{L_r} \times \frac{L_{m} x V}{700}$ | 1808 | 1410 | 1245 | 1749 | 1373 | 1373 |
| $H_{s_{req}}$ | $F_j \times \frac{700}{550}$ | 23.00 | 17.95 | 15.05 | 13.60 | 17.49 | 20.20 |
| $D_{L_{m}}^{R}$ | $* \frac{NACA CHARTS}{9.00}$ | | | | | | | | |
| $\theta$ | | | | | | | | |

* $\% L_r$ calculated as the sum of the induced, profile, wing, and fuselage drag-lift ratios used on rotor lift in order to determine $\left( \% L_r \right)_p$ and blade angle $\theta$ (see section 2.5.3).
### Turbo-prop LEVEL LIFT or required

**Assumed Level**: $\frac{\text{L ram}}{\text{L lift}} = 0.2375$

- **Cross section**: 600 sq. in.
- **Semi-swept area**: 12.5 sq. ft.
- **Cambered**: 1.42
- **Camber**: 0.72
- **Camber**: 0.36
- **Camber**: 0.5

<table>
<thead>
<tr>
<th></th>
<th>Assumed</th>
<th>0.1</th>
<th>0.2</th>
<th>0.3</th>
<th>0.5</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_L$</td>
<td>Figure 6</td>
<td>0.1</td>
<td>0.2</td>
<td>0.3</td>
<td>0.5</td>
</tr>
<tr>
<td>$L/CL$</td>
<td>2.48</td>
<td>4.1</td>
<td>7.09</td>
<td>16.42</td>
<td>14.5</td>
</tr>
<tr>
<td>$V$</td>
<td>It. Sec.</td>
<td>100</td>
<td>150</td>
<td>17</td>
<td>210</td>
</tr>
<tr>
<td>$V_{km}$</td>
<td>200</td>
<td>30</td>
<td>19.4</td>
<td>192.0</td>
<td>11.2</td>
</tr>
<tr>
<td>$q$</td>
<td>$\frac{\text{L ram}}{\text{L lift}}$</td>
<td>10.1</td>
<td>20.2</td>
<td>30.4</td>
<td>60.8</td>
</tr>
<tr>
<td>$L_m$</td>
<td>$\frac{C_L(\text{B/A})_q}{L}$</td>
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<td>1909</td>
<td>171.0</td>
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<td>11400</td>
<td>12360</td>
<td>12300</td>
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<tr>
<td>$C_L_m$</td>
<td>$\frac{L_m}{332q}$</td>
<td>2.706</td>
<td>1464</td>
<td>1.064</td>
<td>760</td>
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<tr>
<td>$L_m/(\rho \cdot \delta)$</td>
<td>Figure 16</td>
<td>9.04</td>
<td>14.0</td>
<td>17.0</td>
<td>20.0</td>
</tr>
<tr>
<td>$L_m/(\rho \cdot \delta)$</td>
<td>Figure 8</td>
<td>9.10</td>
<td>9.10</td>
<td>6.65</td>
<td>4.7</td>
</tr>
<tr>
<td>$(D/L_r)_R$</td>
<td>*</td>
<td>1.2</td>
<td>1.3</td>
<td>1.5</td>
<td>1.5</td>
</tr>
<tr>
<td>$(D/L_r)_J$</td>
<td>**</td>
<td>0.95</td>
<td>0.9</td>
<td>0.9</td>
<td>0.8</td>
</tr>
<tr>
<td>$C/L_r$</td>
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<td>0.013</td>
<td>0.17</td>
<td>0.074</td>
<td>0.333</td>
</tr>
<tr>
<td>$(L/L_r)_w$</td>
<td>$\frac{L_m}{L - L_m}\cdot\frac{1}{(\rho \cdot \delta)}$</td>
<td>0.667</td>
<td>0.008</td>
<td>0.020</td>
<td>0.210</td>
</tr>
<tr>
<td>$L_{tot}$</td>
<td>$\sqrt{L_m^2 + L_{rad}^2 + L_m L_{rad}^2}$</td>
<td>1983</td>
<td>1791</td>
<td>1150</td>
<td>1000</td>
</tr>
<tr>
<td>$D_{tot}$</td>
<td>$\frac{L_{tot}^2}{1 + V}$</td>
<td>1810</td>
<td>1650</td>
<td>1780</td>
<td>2000</td>
</tr>
</tbody>
</table>

1. $(D/L_r)_R = L_m/L \left[ 1/(\rho \cdot \delta) \right]$
2. $(D/L_r)_J = L_m/L \left[ \text{values: \rho = 20, \delta = 0.1} \right]$

**Confidential**
0.1. Cross wind direction of \( \theta \):

\[
\text{Cross wind} = 15 \pm 1.00
\]

High speed:

\[
\text{Air curve} = 1.00, \text{High speed} = \Omega = 800 \text{ FPM}
\]

Assume wind \( U_W = 12 \pm 4 \text{ FPM} \):

\[
\text{High speed} = 15
\]

\[
\frac{15 \times 0.62}{Y} = \frac{15}{105}
\]

\[
\text{Angle} = 11.89
\]

\[
\text{Angle of attack} = -11.89 + 1.8 = -10^\circ
\]

From figure 1, \( S_n = -12 \), \( S_1 = 11 \)

\[
\text{Max load} = S_n \times \rho \times A \times \frac{1}{2}
\]

\[
= 11 \times 1 \times 4.5 \times 11 \times 10^{-3} = 62 \text{ lbs}
\]

At standard load: Load = 1 \times 1

\[
= 11 \times 0.5 \times (11)^2 = 112 \text{ lbs}
\]

\[
\text{Total load on a CFR} = 30 + 50 = 83 \times 112 = 70,082 \text{ lbs}
\]
Assuming 32,462 lb. to be still limit rotor load at 100 ft./sec.,
calculate power required for this same condition:

\[ C \tau \sigma = \frac{32,462}{\tau \sigma \times 3200(700)^2} = 0.0008 \]

Fig 17 shows for initial stall \( C/\sigma \) of .88 at \( \mu = .16 \),
and \( P/L \) for rotor = .64.

For \( P/L = .64 \), calculate the following using ACA charts:

\[ C_{1}/\sigma = \frac{32,462}{\tau \sigma \times (C/\sigma)^2 \times 3200(100)^2} = 8.31 \; ; \; C_L = .740 \]

From ACA charts and by the methods of section 6.1.3
\[
\begin{align*}
(L/L)_0 &= .1160 \\
(r/L)_1 &= .1670 \\
(D/L)_0 &= .0400 \\
(r/L)_0 &= .0674 \\
L_{\text{ext}}/L(L/L)_w &= .0007 \\
&= \frac{.3901 \times (\tau \sigma \text{TOTAL})}{.3800} \\
(r/L)_0 &= .3400 - .321 = .1190 \\
\end{align*}
\]

\[ B = .1480 \times 3200 \times 100 = 1140 \]

\[ C/\sigma = \frac{1140 \times 33000}{30000} = 1200 \text{ ft./min.} \]

which checks the assumed value of 1200 ft./min.
1.2 Considering power available is the limiting factor:

Assume \( \mu = 1.0 \), nose velocity = 100 ft./sec., at \( \Omega = 700 \) ft./sec.
maximum \( \rho/\pi = 3440 \) ft./min.

Climb angle:

\[
\tan \Theta = \frac{\rho}{\pi} \frac{\Omega}{\rho_{/2}}
\]

\( \Theta = 30^\circ \)

Let angle of attack = \( -30 + 3 = 0^\circ \)

From figure 1, \( CL_0 = -1.0 \), \( L/D = 5.7 \)

In down load = \( -1.0 \times \frac{\rho}{2} \times 332 \times 11,700 \)

= 9750 lbs.

Permits drag load = \( A \times q \)

\[ = 214 \times \frac{\rho}{2} \times (\chi \times \Delta)^2 = 850 \text{ lbs.} \)

Total rotor load = 30,000 + 9750 + 850 = 40,600 lbs.

Rotor \( P/L = \frac{800 \times 560}{4060} \times \frac{1000}{4060} \times 100 \)

\[ = 346 \]

Rotor \( C_{L} \sigma = \frac{40,600}{\sigma \times \frac{\rho}{2} \times 3320(100)^2} = 10.46 \]
From NACA charts at \( \alpha/L = .30 \), which is permissible, since \((\alpha/L)_o\) varies but little with \( \alpha/L \) change in the low advance ratio range. (See reference 9.8).

\[
(\alpha/L)_o = .1205 \\
(\alpha/L)_i = .2355 \\
(\alpha/L)_j = .0320 \\
(\alpha/L)_p = .0089 \\
L_{RL}/L(\alpha/L)_w = .0275 \\
\frac{4.214}{(\alpha/L)_{TOTAL}} \\
(\alpha/L)_c = .3480 - .4214 = .4266 \\
\text{HP/ft/min} = .4266 \times 40686 \times 105 = 3320 \text{ ft./min.} \\
\frac{3320 \times 33000 = 3550 \text{ ft./min.}}{\text{30000}}
\]

which checks the assumed value of 3640 ft./min.

8.1.7 Propeller propulsion maximum rate of climb

Excess horsepower at 140 knots = 1680

(Reference figure 9.)

Therefore, \( R/C = \frac{1680 \times 33000}{30000} = 1850 \text{ ft./min.} \)
8.1.3 Normal fuel load calculation

The calculated total fuel load requirements of the model 73 consist of the following:

a. 15 minutes at normal rated power
   (1) 2 minutes hovering
   (2) 15 minutes at 100-knot cruising
b. 100-mile combat radius at cruise speed of 220 knots
c. 10% reserve
d. 5 increase in all 73 for service variation

Average fuel required = 6770 lbs./hr. (Conservative estimate from reference 3.1)

Turbo-prop fuel required = 2878 lbs./hr. (reference 1.3) (normal rated power)

Turbo-prop fuel required = 2610 lbs./hr. (reference 3.3) (at cruise hover)

15 minutes warm-up

\[
\frac{6770 \times 1.05 \times 2}{100} = 237 \text{ lbs.}
\]

\[
\frac{2878 \times 1.05 \times 13}{100} = 354 \text{ lbs.}
\]

100-mile cruise radius at 220 knots

\[
\frac{2 \times 10 \times 1.05 \times 250}{720} = 2450 \text{ lbs.}
\]

10% reserve

\[
= 378 \text{ lbs.}
\]

TOTAL = 5787 lbs.
FIGURE - 9

MODEL 78

LEVEL FLIGHT PERFORMANCE

POWER REQUIRED vs VELOCITY

ALTITUDE - SEA LEVEL
NORMAL GROSS WEIGHT

POWER AVAILABLE @ STANDARD CONDITIONS

- WING STALL LIMITATION ($C_L = 1.65$)
- BLADE STALL LIMITATION ($\alpha_{1/4} = 12^\circ$)
- ADVANCING BLADE MACH NUMBER LIMITATION ($V + \frac{dV}{dt} = 900/\text{sec}$)

---

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LEVEL FLIGHT PERFORMANCE

WING LIFT COEFFICIENT vs VELOCITY

ALTITUDE - SEA LEVEL

\[ C_{L} = 1.6 \text{ at stall} \]

\[ \Omega r = 700 \]

\[ 360 \]

\[ 420 \]

\[ 0 \quad 40 \quad 80 \quad 120 \quad 160 \quad 200 \quad 240 \quad 280 \]

VELOCITY - KNOTS

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LEVEL FLIGHT PERFORMANCE

ROTOR LOAD vs. VELOCITY

GROSS WT = 35,000 LB.
ALTITUDE = SEA LEVEL

120
100
80
60
40
20
0

0 40 80 120 160 200 240 280

VELOCITY - KNOTS

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FIGURE - 18

MODEL 78

MAXIMUM RATE OF CLIMB vs VELOCITY

NORMAL GROSS WEIGHT
ALTITUDE - SEA LEVEL

MAXIMUM RATE OF CLIMB - FEET/MINUTE

POWER LIMITATION

BLADE STALL LIMITATION

VELOCITY - KNOTS

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FIGURE - 18

MODEL 78

RATIO OF EXCESS POWER TO EFFECTIVE CLimb POWER

VERSUS

VERTICAL RATE OF CLIMB

\[ K_v = \frac{\text{EXCESS POWER}}{\text{EFFECTIVE CLIMB POWER}} \]

VERTICAL RATE OF CLIMB ~ FEET PER MINUTE

BASED ON NACA FLIGHT TEST RESULTS
REPORTED IN REFERENCE

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Figure 17

Model 78

$C_{y}/\alpha$ at Stall vs $\%_L$ of Rotor

Reference: 9.8
Figure 20a
Jet Drag-Lift Ratio vs Reciprocal of Rotor Lift Coefficient

\[ \frac{d_r}{f_r} = 1 \]

FOR MODEL 78
MULTIPLY ORDINATE
BY 0.00010

Jet Drag-Rotor Lift Ratio = \( d_r \)

Reciprocal of Rotor Lift Coefficient = \( f_r \)

Same as pg. 60 but scale is changed.
9. REFERENCES


9.2 "BuAer Specification for the Aerodynamic, Structural, and Power-Plant Requirements for Helicopters", NAVAER SL-189, effective 1 August 1950.


9.18 McDonnell Aircraft Corporation Engineering letters submitted to the Office of Naval Research:

9.18.1 2136-701-1380, dated 6 July 1949
9.18.2 2136-701-1390, dated 28 August 1949
9.18.3 2136-701-1400, dated 14 November 1949
9.18.4 2136-701-1441, dated 13 January 1950
9.18.5 2136-701-1498, dated 11 March 1950
9.18.6 2136-701-1504, dated 15 May 1950
9.18.7 2136-701-1505, dated 14 July 1950
9.18.8 2136-701-1524, dated 13 August 1950
9.18.9 2136-701-1544, dated 15 September 1950
9.18.10 2136-701-1567, dated 20 October 1950
9.18.11 2136-701-1675, dated 7 November 1950