NOISE REDUCTION OF A TILT-ROTOR AIRCRAFT
INCLUDING EFFECTS ON WEIGHT AND PERFORMANCE

JUNE 1973

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National Aeronautics and Space Administration
and
United States Army Air Mobility Research and Development Laboratory
Ames Directorate

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LIST OF SYMBOLS

A total rotor disc area, $ft^2$

$\mathcal{A}$ aspect ratio

B number of blades per rotor

BRS best range speed, knots

C chord, ft.

$C_P$ rotor power coefficient, $\frac{RHP 550/\rho \pi P^2 V_t^3}{f}$

$C_{TH}$ rotor thrust coefficient, $\frac{T/\rho \pi R^2 V_t^2}{f}$

$C_T$ helicopter lift coefficient in forward flight, $\frac{L/\rho \pi R^2 V_t^2}{f}$

E$\text{PNL}$ effective perceived noise level, $\frac{\text{EPNdB}}{f}$

FM hovering aerodynamic efficiency, figure-of-merit

$H$ height of acoustically lined engine inlet duct, $ft$.

$HP$ horsepower

$HP_f$ horsepower required in cruise flight

Hz Hertz, cycles per second

I moment of inertia, slug-ft$^2$

$L$ lift, pounds

$L$ length of acoustically lined engine inlet duct, $ft$.

$M$ Mach number, freestream velocity/speed of sound

$M_{MO}$ aircraft maximum operating Mach number

$M_T$ Mach number at rotor tip

$N$ noise, dB

OASPL overall sound pressure level, $dB$

$P$ perceived noise level, $PNdB$

$R$ rotor radius, $dB$
LIST OF SYMBOLS (CONT'D)

PEQ required
RHP rotor horsepower
\( \text{RHP}_{\text{ind}} \) induced rotor horsepower required
RPM revolutions per minute
SPL sound pressure level, dB
\( S_w \) wing area, \( \text{ft}^2 \)
T thrust, \( \text{lb} \).
TLS transmission limited speed, knots
\( T/W \) thrust-to-gross weight ratio
V aircraft speed of flight, knots, \( \text{ft/sec} \).
\( V_{\text{block}} \) (mission range)/(mission total elapsed time), including dead time and hovering time, knots
\( V_{\text{dive}} \) maximum dive speed, knots equivalent airspeed
\( V_{\text{max}} \) maximum cruising speed, knots
\( V_{\text{MO}} \) maximum operating speed, knots equivalent airspeed
\( V_{\text{opt}} \) speed at which the aircraft maximum lift-to-drag ratio occurs, knots, \( \text{ft/sec} \).
\( V_t \) rotor tip speed, \( \text{ft/sec} \).
W gross weight, \( \text{lbs} \).
\( W/A \) nominal rotor disc loading, \( \text{lb/ft}^2 \)
\( W_E, W_F \) weight empty, \( \text{lbs} \).
\( W_h \) hover gross weight, \( \text{lbs} \)
\( W_u \) useful load, \( \text{lbs} \).
XMSN transmission
\( X_{\text{FLL}} \) steady-state rotor performance parameter determining the rotor unsteady airload harmonic decay
### LIST OF SYMBOLS (CONT'D)

<table>
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<tr>
<th>Symbol</th>
<th>Description</th>
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<tr>
<td>$a$</td>
<td>airfoil lift curve slope, rad$^{-1}$</td>
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<tr>
<td>$b$</td>
<td>wing span, ft.</td>
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<tr>
<td>$c$</td>
<td>speed of sound in air, knots or ft/sec</td>
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<tr>
<td>$dB$</td>
<td>decibel</td>
</tr>
<tr>
<td>$f$</td>
<td>frequency, Hz or ops</td>
</tr>
<tr>
<td>$f_e$</td>
<td>parasite drag equivalent flat plate area, ft$^2$</td>
</tr>
<tr>
<td>$g$</td>
<td>acceleration of earth gravity, (32.17 \text{ ft/sec}^2)</td>
</tr>
<tr>
<td>$k_{ind_h}$</td>
<td>hover nonideal induced power correction</td>
</tr>
<tr>
<td>$n$</td>
<td>exponential slope of rotor air speed as a function of rotor rotational harmonic</td>
</tr>
<tr>
<td>$v$</td>
<td>ideal rotor induced velocity, ft/sec</td>
</tr>
<tr>
<td>$\omega_b$</td>
<td>span loading, $W/b^2$, lbs/ft$^2$</td>
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<tr>
<td>$\omega_f$</td>
<td>equivalent flat plate parasite drag loading, $W/f_e$, lbs/ft$^2$</td>
</tr>
<tr>
<td>$\alpha_p$</td>
<td>angle between thrust line and freestream velocity, degrees</td>
</tr>
<tr>
<td>$\lambda$</td>
<td>incremental quantity</td>
</tr>
<tr>
<td>$\Delta I_\lambda$</td>
<td>harmonic induced force on one rotor blade</td>
</tr>
<tr>
<td>$\eta_{ACC}$</td>
<td>accessory losses</td>
</tr>
<tr>
<td>$\eta_{CR}$</td>
<td>propeller cruise design point efficiency</td>
</tr>
<tr>
<td>$\eta_I$</td>
<td>installation losses</td>
</tr>
<tr>
<td>$\eta_P$</td>
<td>propeller efficiency</td>
</tr>
<tr>
<td>$\eta_{TR}$</td>
<td>transmission efficiency</td>
</tr>
<tr>
<td>$\theta_t$</td>
<td>rotor blade twist measured from root to tip, tip-up positive, degrees</td>
</tr>
<tr>
<td>$\theta_{.75}$</td>
<td>rotor collective angle at $.75R$ radial station, degrees</td>
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<tr>
<td>$\lambda$</td>
<td>wavelength, meters</td>
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LIST OF SYMBOLS (CONT'D)

$\lambda_h$ rotor inflow ratio, $(V \cos \alpha_p + \nu)/V_t$

$\mu$ rotor advance ratio, $V/V_t$

$\pi$ 3.14159

$\rho$ air density, slugs/ft$^3$

$\sigma$ rotor solidity ratio, blade area/disc area

SUBSCRIPT

$h$, $H$ in hover

$o$ original or baseline value
NOISE REDUCTION OF A TILT-ROTOR AIRCRAFT INCLUDING EFFECTS ON WEIGHT AND PERFORMANCE

by

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SUMMARY

Various methods for far-field noise reduction of a tilt-rotor acoustic signature and the performance and weight tradeoffs which result from modification of the noise sources are considered in this report. In order to provide a realistic approach for the investigation, the Boeing Tilt-Rotor Flight Research Aircraft (Model 222 as defined in Ref. 1) was selected as the baseline. This aircraft has undergone considerable engineering development. Its rotor has been manufactured and tested in the Ames full-scale wind tunnel. Therefore, the study reflects the current state-of-the-art of aircraft design for far-field acoustic signature reduction and is not based solely on an engineering feasibility (paper) aircraft. This report supplants a previous study investigating reduction of noise signature through the management of the terminal flight trajectory (Ref. 2).

The following tasks comprise this study:
A. Review of rotor acoustic phenomena dependence on design parameters.
B. Definition of the tilt-rotor aircraft performance, weight and acoustic signature with respect to aircraft design parameters.

C. Definition of two new "quiet" aircraft using design ground rules applicable to the Model 222.

On the basis of Task B, the following four design and/or operational parameters were selected as potentially representing the most important inputs to noise reduction at the source using current technology.

1. Tip Speed
2. Number of blades
3. Disc loading
4. Rotor blade area.

A quantitative study of the effectiveness of those parameters was performed in Task B. Although the main effort was directed toward various aspects of noise reduction generated by the rotors, acoustic problems of the powerplants were also briefly discussed. The results of the studies performed in Task B are generalized by presenting them under the form of various derivatives about the Model 222 tilt-rotor design parameters. Rotor tip speed was identified as the most effective design parameter for noise reduction.

Finally in Task C, the most effective methods of noise attenuation at the source, resulting in the most favorable performance and/or weight tradeoffs, were applied to the design of two tilt-rotor aircraft. These aircraft were designed to have the same...
basic performance (1298-pound payload over a 100 n.mi. radius mission), structural envelope and flying qualities.

In addition to the above constraints, the acoustic signatures of both aircraft (in comparison with the Model 222) must be reduced at a distance of 500 feet while hovering out-of-ground effect. For one aircraft, this reduction should amount to 10 PNdB in the perceived noise level, while for the other, 10 dB in the overall sound pressure level. This results in an increase of design gross weight of 25.3 and 5.8 percent, respectively.

I. INTRODUCTION

Reduction of the far-field noise intensity through acoustic improvements at the source and management of terminal flight trajectories of aircraft represent two of the most important inputs into improving acoustic signature on the ground. This latter aspect is significant from the military (detection, exposure of ground personnel, etc.) as well as the civilian (annoyance or the population) point of view.

Noise abatement possibilities offered by management of the terminal flight trajectories have been studied for a transport-type tilt-rotor aircraft of the 46,000-pound gross weight class (Ref. 2). In that study, penalties in increased fuel consumption and/or time required to reach prescribed cruise conditions (altitude and speed) were also indicated.
In order to complete the picture of the effectiveness of various approaches to noise abatement on the ground, it was necessary to investigate aspects of noise reduction at the source. For this task, the Ling tilt rotor flight research aircraft (Fig. 1-1, 12,000 pound gross weight) was selected as the reference from which acoustic improvements and associated weight and performance penalties were evaluated. The selection of this aircraft as a baseline results in a more realistic study because of the five-year design effort, wind-tunnel configuration studies, and construction of a full-scale rotor and control system.

The Model 222 is not necessarily representative of the whole spectrum of tilt-rotor aircraft that may be developed in the future (from utility to transports) because of its design gross weight and weight breakdown. However, the relative (non-dimensional) trends developed in this study would be indicative of the noise reduction - performance and/or weight tradeoffs - of other tilt-rotor aircraft as well, after accounting for the differences in wing span loading, equivalent parasite drag loading, and weight-empty to gross-weight ratio. In order to best accomplish the envisioned tasks, the whole study is divided into the following sections:

- Review of the Influence of Design Parameters on Rotor Acoustics
- Turboshaft Engine Noise
FIGURE 1-1. MODEL 222-1 TILT ROTOR NASA RESEARCH AIRCRAFT
• Rotor Design Parameters Tradeoff Studies (weight, performance, and noise)
• Partial Derivatives with Respect to Operational and Design Parameters
• Conceptual Design of Quiet Tilt Rotors
• Conclusions
• Recommendations

Each of the above sections represents, to some extent, a closed entity, but together, they show efficient methods to reduce the aircraft far-field acoustic signature as well as to indicate performance and/or weight penalties to be encountered.

In general, this report may be looked upon as a review of current design practices for the reduction of aircraft noise. However, it should be emphasized that all of the above studies reflect the present state of the art; thus, the penalties shown may be reduced with advancements in aircraft technology. Consequently, directions for R&D efforts which may contribute to smaller performance and weight penalties associated with noise reduction are pointed out.

II. INFLUENCE OF DESIGN PARAMETERS ON ROTOR ACOUSTICS

General Discussion

A major part of the acoustic/performance tradeoff study of tilt-rotor aircraft was devoted to an evaluation of state-of-the-
art regarding minimization of rotor noise. A digest of the findings is presented herein and additional data forming the background are reviewed in Appendix A.

There has been substantial data accrued in the last five years by government and industry to document the effect of various design and operational parameters on the noise of rotors, but not all of this has been applicable to tilt-rotor acoustics.

An area of particular interest was the applicability of current noise prediction methods to the tilt-rotor. The method presented in Ref. 3 has proven to be reliable for low-twist rotors; however, it had never been evaluated or substantiated for highly-twisted tilt-rotors due to a lack of suitable full-scale tests. As of this writing, it appears that a representation of airloading in terms of azimuth position and blade passage harmonics is sufficient to predict rotary-wing noise, whether helicopter or tilt-rotor. Unfortunately, the state-of-the-art for high harmonic airload prediction is not sufficient to define any but the lowest harmonics of noise. However, recent data from a 26-foot diameter rotor (Model 222) in the NASA Ames 40 x 80 facility should substantially aid the advancement of tilt-rotor acoustics.

To optimize the design of a quiet tilt-rotor aircraft, the sensitivity of trades of noise with all the operating and design variables which affect vehicle performance must be known. For
example, tip speed plays an important role in establishing rotor acoustic signature. However, while this aspect for all types of airscrews has been generally well known, precise knowledge of this particular parameter on tilt-rotor acoustic characteristics was lacking. Furthermore, most of the available data had been investigated on an individual basis, but not viewed totally.

In this study, all the meaningful data which could be amassed were evaluated for their influence on tilt-rotor noise characteristics. There are many design and operational parameters which can be applied to the design and operation to influence rotor acoustics. Some of them offer substantial control of the acoustic signature while others have only a second-order effect on the noise. An investigation into the effectiveness of the following major parameters is discussed in this report:

1. Tip Speed
2. Blade Geometry (planform and twist)
3. Airfoil Sections
4. Design Alternatives within Rotor Itself (number of blades, radii of blades, angle between blades, blade area, disc loading, power loading, etc.)
5. Special Devices (such as blade tips, leading edge modifications, boundary layer control devices, tip blowing, etc.).

Some of the information used in the study was assembled from a recent comprehensive search of contemporary literature.
Other inputs were obtained from Boeing-Vertol research as well as many years of continuous review of literature (primarily from the University of Pittsburgh's Knowledge Availability Systems Center, Abstract Search Facility). This, supplemented by personal contact with researchers working on special devices, provided the basis for the material presented in this section.

In describing the acoustic signature of a rotor, it is desirable to define the frequency spectra using a narrow filter in order to investigate harmonic as well as broadband components of noise. In so doing, a detailed definition of all frequencies reveals three spectral regions which display definable characteristics (see Figure 2-1).

![Figure 2-1 Characteristic Frequency Ranges for Rotors](image)
1. The very low frequency range (harmonic numbers less than 10) where the lowest harmonics may be below the audible frequency threshold. This region of the frequency spectra is characterized by a decrease in harmonic amplitude with increasing harmonic number. In this case, good correlation with theory is typical (5dB).

2. The intermediate frequency range (harmonic numbers above 10) where harmonics of blade passage are definable. Harmonic levels increase in amplitude with increasing frequency, later decreasing and finally blending in with the broadband noise. This characteristic of rotor noise may result from incident and reflected wave phasing.

3. The high frequency range (no harmonics of blade passage present) where the noise is of a broadband characteristic. The broadband noise spectra generally displays an amplitude maximum somewhere in this range.

A discussion of the effectiveness of idealized variations in a large number of design variables on far-field noise is included in Section IV.

While the impact of each of the design variables previously noted will be discussed separately, a combined summary of all the findings is presented in Figure 2-2. A review of this figure shows that for a constant thrust, the controlling parameter is tip speed, and this appears to be the factor which exerts the major
CONSTANT THRUST (10,000 - 20,000 lb)

For effect of a change in number of blades, calculate the change in tip speed required to provide constant thrust.

**Figure 2-2** Effect of Rotor Design on Noise
order of magnitude influence on the generated noise. Variations in number of blades, airfoils, planforms, and twist may combine to provide the illustrated scatter about the tip speed trend line. These tend to be of second-order influence at lower speeds, but become substantial as drag divergence is approached and thin airfoils at the blade tip become important.

In actual design practice where a given payload must be carried a specified distance, the effect of decreasing tip speed will be to increase the gross weight which would have the effect of rotating the Figure 2-2 trend line clockwise about the 750 fps point thus reducing the net acoustic benefit.

The above discussion deals with overall sound pressure level which, for propellers and rotors, is set by the lowest few harmonics. The details of airfoil and blade design, however, affect the higher harmonic noise generation in a manner which appears to defy systematic categorization. Although this has no effect on overall sound pressure level, and a very minor effect on calculated perceived noise level, it has been observed that blades of different designs have noticeably different sounds which influence subjective acceptability.

Unfortunately, the tip speed reduction comes at a performance price as illustrated in Figure 2-3. This figure, from the Ref. 4 paper by Stepniewski and Schmitz, illustrates the adverse weight trend of transmissions, blades, and control systems which accompany
reduced tip speeds. Although this figure was based on helicopter designs, it is indicative of the expected trends.

![Diagram](image)

**Figure 2-3** Weights, \( v_{\text{max}} \) and DOC trends with noise reduction by lowering \( V_t \).

**Tip Speeds**

Reduction in helicopter rotor noise as obtained by lowering tip speed is generally predictable with good accuracy and has been documented by many investigators. The data reviewed by this study
included the following:

(1) Substantial quantities of noise levels recorded on the Boeing-Vertol experimental whirl tower facility (Figure 2.4) on CH-2, CH-46, and CH-47 rotor blades (Ref. 5). In addition it included two-, three-, and four-bladed rotors with blades which were otherwise identical in design and construction, and numerous wind-tunnel noise surveys, as well as data obtained on a 13-foot diameter propeller with 41° twist.

(2) Full-scale rotor data published by Hubbard and Maglieri (Ref. 6).

(3) Whirl tower data published by Stuckey and Goddard on a Westland rotor (Ref. 7).

The above data has been summarized in the characteristic tip speed trend shown in Figure 2-5. The data shown are for overall, or low harmonics of noise. Trends of specific sets of data display variations of 3-6 dB per 100 ft/sec change in tip speed. Grouped together, the combined data shows a 4.2 dB/100 fps slope. Higher harmonics are not as consistent with variations in tip speed and, in fact, tend to be somewhat erratic. As shown in the Appendix, the higher harmonics display a slope of 3 dB/100 fps, or less.
FIGURE 2-4  BOEING VERTOL EXPERIMENTAL
WHIRL TOWER AND MICROPHONE ARRAY
FIGURE 2-5 EFFECT OF TIP SPEED ON ROTOR NOISE LEVELS
Number of Blades

There are several design variables that lead to noise reduction of a rotor system, but which are not clearly identifiable. This is due to the fact that they are generally incorporated into the rotor, together with changes to other rotor parameters which, in themselves, may also contribute to noise reduction. Number of blades is one of these. For example, there is no available data regarding changes of blade number where the solidity of the rotor remained constant, since all published data had been obtained from programs which added blades of the same geometrical configurations to the rotor, and thus increased rotor solidity. Adding additional blades also creates a change in blade loading. Depending on how the data on blade number is compared, several conclusions may be drawn. First, blade number at constant tip speed will be examined.

Figure 2-6 illustrates the effect of blade number of measured sound levels as a function of thrust as obtained on a whirl tower. When viewed in this manner, adding blades to the rotor clearly improves the acoustic signature.

Leverton (Ref. 8), from research at the Institute of Sound and Vibration Research, Univ. of Southampton, concludes that rotational noise will decrease by 4-5 dB per blade added, while broadband noise should decrease by the factor 10 log (B + blades added)/B).
FIGURE 2-6  WHIRL TOWER NOISE OF 2 AND 4-BLADED ROTORS
Figure 2-7 compares noise produced by the 2- and 4-bladed rotors at constant thrust. For this size rotor, the noise increases rapidly with tip speeds above approximately 725 fps. Thus, for minimum noise, the number of blades should be high enough to produce the desired thrust while operating at a tip speed of 725 fps or less.

However, when 2- and 4-bladed data is compared on the basis of \( C_T/\sigma \), as is typical for performance evaluation, the data collapses to a large scatterband and the value of increasing the number of blades disappears (Figure 2-8). It appears, hence, that the real advantage of adding similar blades of constant area to a rotor results from an ability to reduce tip speed because of the increased lift from the added blade.

Broadband noise produced by the 2- and 4-bladed rotors is compared in Figure 2-9. At the low thrust (9,000 pounds), the 2-bladed rotor is approximately 5 dB less noisy than the 4-bladed one throughout the range of tip speeds tested. At 18,000-pound thrust, the broadband noise again displays a 5 dB difference at \( V_T = 900 \) fps. This difference decreases with tip speed, both rotors producing the same noise below 700 fps.

In addition to permitting a reduction in tip speed, an increase in the number of blades will also result in a noise reduction if the rotor rpm and radius are maintained unchanged. The Army-NASA OH-6A Quiet Helicopter (Ref. 9) was modified by increasing the number of main rotor blades from 4 to 5 and the
FIGURE 2-7 WHIRL TOWER ROTOR NOISE COMPARISON OF 2 AND 4-BLADED ROTORS
FIGURE 2-8  WHIRL TOWER NOISE OF 2 AND 4-BLADED ROTORS
BOEING VERTOL
WHIRL TOWER DATA
ROTOR DIA. = 60 FT
CH 47B/C BLADES
MIC. LOC. 3 DIA. FROM TOWER
50 FT BELOW ROTOR

LEGEND

9,000 LB THRUST

BROAD BAND SOUND PRESSURE LEVEL - 68 m/s^2 x 10^-5 N/m^2

TIP SPEED, V_T, FPS

18,000 LB THRUST

BROAD BAND SOUND PRESSURE LEVEL - 48 m/s^2 x 10^-6 N/m^2

TIP SPEED, V_T, FPS

FIGURE 2-9 BROADBAND NOISE COMPARISON OF 2 AND 4-BLADED ROTORS
number of tail rotor blades from 2 to 4 without changing their original dimensions. Other minor modifications were also made, but when both the standard and modified aircraft were operated at the same rpm, the increased number of blades was primarily responsible for an average 9 dB noise reduction measured during a 100-foot altitude fly-by.

Thrust

An increase in rotor thrust results in an increase in noise level of the rotor, but it is both tip-speed and frequency dependent. Figure 2-10 was selected to illustrate the point because the data it represents was recorded on a system which measured all frequencies from 2 Hz to 10,000 Hz with the same frequency response. The increase in noise level with thrust for the 1st harmonic of blade passage frequency increases with very predictable regularity at the rate of 0.85 dB/1000 pounds of thrust at the high tip speeds (650 fps) and drops to 0.3 dB/1000 pounds of thrust at the high tip speeds (900 fps). The higher harmonics display significantly more scatter as well as increase at a lower rate: the slopes at the equivalent tip speeds being 0.5 dB/1000 pounds at \( V_t = 650 \), and 0.2 dB/1000 pounds at \( V_t = 900 \). The scatter in the higher harmonics appears to be typical of rotor data and apparently, results from the ambient condition effects such as wind gusts which affect the higher harmonic airloading. Additional data on the effect of thrust on noise may be found in the Appendix.
FIGURE 2-10 EFFECT OF THRUST ON ROTATIONAL NOISE HARMONICS
Blade Tip Shape

A wide variety of tip shapes have been applied to rotor blades to reduce noise. However, these attempts have met with minimal success in reducing noise below that produced by a standard square tip. Trapezoidal and square tips appear to be the two most common nonstandard shapes. A test of full-scale trapezoidal tips (Ref. 10) resulted in a reduction of broadband or 'vortex' noise by approximately 7 dB compared with a square tip at low thrust levels, but at normal thrusts, the improvement was less impressive.

Other investigations have shown the square tip to exhibit a small advantage over the trapezoidal one (see Pollard and Leverton, Ref. 11), but either of these tips appears to generate lower broadband noise than almost any other configuration evaluated. However, the latter test was conducted on a 10-foot diameter rotor at 8° collective pitch and for a $V_t$ no greater than 367 fps. It appears, hence, that tip shape does exhibit some control over broadband noise, but has little influence on the rotational noise established by the basic airloading on the blade.

Blade Planform

With the exception of tip speed, number of blades and thrust level, all other noise control techniques which can be incorporated in the rotor system appear to have only secondary effects on rotational noise. The influence of blade planform on rotor noise for
example, was investigated by Boeing-Vertol on a whirl tower during two test programs. For one program, a set of blades was constructed with constant chord from the cutout to the 59 percent radius, and a 3:1 linear taper from 59 percent radius to the tip was incorporated. A NACA 0012 airfoil was maintained for the entire blade. The noise produced by these blades was compared to standard 0012 CH-47A blades. Plan views of these blades are shown in Figure 2-11.

Noise spectra from this test are illustrated in Figures 2-12 and 2-13. From these figures, it can be seen that little or no change in amplitude in the very low harmonics results from blade planform and corresponding solidity modifications. On the other hand, at 722 fps, the broadband noise of the tapered blade was constantly higher than for the reference blade and this probably resulted from the higher collectives required to achieve the same thrust.

In a second program, another experimental blade (designated the Advanced Geometry Blade (AGB)) was designed to demonstrate the practicality of advanced composite materials. It included both spanwise airfoil variation as well as planform taper. The airfoil section varied linearly from a V23012 at the cutout to a V23010 at the 70 percent radius, and then to a 13006 at the tip. The planform geometry also is illustrated in Figure 2-11. Although the effects of variation of airfoil sections and planform geometry cannot be separated, a noise comparison of this blade with the CH-47B blade is nevertheless interesting. The AGB spectra are compared to those
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<td>23010–1.58</td>
<td>VARIES (SEE SKETCH)</td>
</tr>
<tr>
<td>RADIUS</td>
<td>29.5 FT</td>
<td>30 FT</td>
<td>31 FT</td>
</tr>
<tr>
<td>CHORD</td>
<td>23”</td>
<td>25.25”</td>
<td>VARIES</td>
</tr>
<tr>
<td>TWIST</td>
<td>−9.0°</td>
<td>−9.147°</td>
<td>−6°</td>
</tr>
<tr>
<td>CUTOUT</td>
<td>5.7 FT</td>
<td>5.8 FT</td>
<td></td>
</tr>
<tr>
<td>SOLIDITY</td>
<td>.062</td>
<td>.067</td>
<td>.069</td>
</tr>
</tbody>
</table>

| ROTOR CENTERLINE | - | - | - | - |

---

**FIGURE 2-11 BLADE DESIGNS**
WHIRL TOWER DATA
R = 30 FT
3 BLADES
3 DIA. MIC.

- - - 0012 CH-47A
- - - TAPERED

FIGURE 2-12 EFFECT OF PLANFORM
WHIRL TOWER DATA

R = 30 FT
3 BLADES
3 DIA. MIC.

\( V_T = 722 \text{ FPS} \)
\( T = 13,000 \text{ LB} \)

\( V_T = 722 \text{ FPS} \)
\( T = 18,000 \text{ LB} \)

\( V_T = 722 \text{ FPS} \)
\( T = 22,000 \text{ LB} \)

FIGURE 2-13 EFFECT OF PLANFORM
of the standard 23010 CH-47B rotor in Figures 2-14 through 2-16. There is no conclusive advantage for either blade. At a thrust of 9000 pounds and tip speed of 750 fps, the standard blade spectrum is approximately 10 dB higher at frequencies of about 30 cps. However, the standard blade has a lower spectrum than the AGB at a tip speed of 650 fps, and shows little difference at higher thrusts at 750 fps.

Note again that blade comparisons at the same tip speed and thrust show very similar sound pressure levels for the low harmonics. Differences in the spectra appear only in the high harmonics and in broadband noise. For the AGB/CH-47B rotor comparison, the change in higher harmonics which results from the blade variations is probably less significant than that due to changing ambient conditions between test runs.

Ambient wind has a substantial effect on the noise produced by a rotor (Ref. 12) and this is one reason why whirl tower measurements frequently result in data scatter of as much as 6 dB for repeated points. Consequently, the effects of such blade design parameters as planform and airfoil section (which represent second-order influences) are frequently within the scatter of repeated test points and thus, are difficult to accurately measure.

With the limited amount of data available on the noise of rotors with various planform, radii or airfoil sections, there is no strong evidence to recommend one configuration over another. Furthermore, it is also clear that a particular blade is not
FIGURE 2-14 EFFECT OF COMBINED PLANFORM AND AIRFOIL
FIGURE 2-15 EFFECT OF COMBINED PLANFORM AND AIRFOIL
WHIRL TOWER DATA
R = 30 FT
3 BLADES
3 DIA. MIC.

- - - 23010 CH-47C
- - - AGB

\[
\begin{align*}
T &= 27,000 \text{ LB} \\
V_T &= 700 \text{ FPS}
\end{align*}
\]

\[
\begin{align*}
T &= 27,000 \text{ LB} \\
V_T &= 800 \text{ FPS}
\end{align*}
\]

FIGURE 2-16 EFFECT OF COMBINED PLANFORM AND AIRFOIL
consistently better or worse than any other in terms of its acoustic specturm. Based on this evidence, it appears that a rotor should be designed for its optimum performance at the lowest possible tip speed.

Airfoil Section

The airfoil sections typically used for rotor blades have little effect on the noise generated by the blades at local Mach numbers below 0.85. Above this, airfoil thickness becomes important and the inception of compressibility effects produce an undesirable impulsive noise. Below \( M = 0.8 \), spanwise air-loading determines the pressure amplitude fluctuation of a passing blade. Waveforms of different airfoils sometimes display different time-history characteristics, but there is little or no change in the audible characteristic.

Figure 2-17 compares the spectrum of a model with V23010-1.58 airfoil sections with another made up of VR-7 sections inboard and VR-8 sections outboard of 85 percent \( R \). (The VR-7 and VR-8 are "high-lift" airfoils developed at Vertol for application to the HLH-ATC rotor). There is no significant difference over the entire frequency range at the thrust levels tested, which are all below stall.

Figures 2-18 through 2-20 provide a comparison between the symmetrical 0012 airfoil incorporated in CH-47A rotors and the cambered V23010-1.58 used in the CH-47B and C. These data were
WIND TUNNEL DATA
ROTOR DIA. = 6 FT
TIP SPEED = 725 FPS
3 BLADED ROTOR
MIC. LOC - 1.0 DIA. FROM SHAFT
.3 DIA. BELOW ROTOR

C'\tau/\alpha = .07

SOUND PRESSURE LEVEL - dB re 2 \times 10^{-5} N/m^2

C'\tau/\alpha = .09

C'\tau/\alpha = .11

FREQUENCY - Hz

FIGURE 2-17 AIRFOIL SPECTRA IN HOVER
FIGURE 2-18 EFFECT OF AIRFOIL SECTION
WHIRL TOWER DATA
3 CH-47 BLADES
ROTOR DIA. = 60 FT
MIC. LOC. = 3 DIA.

THRUST = 18,000 LB
V_T = 650 FPS

SOUND PRESSURE LEVEL - dB re 2\times 10^{-6} N/m^2

FIGURE 2-19 EFFECT OF AIRFOIL SECTION

37.
WHIRL TOWER DATA
3 CH-47 BLADES
ROTOR DIA. = 60 FT

--- 23010 CH-47C
--- 0012 CH-47A

MIC. LOC. – 3 DIA.
Vₜ = 650 FT/SEC

SOUND PRESSURE LEVEL - dB re 2 x 10⁻⁶ N/m²

THRUST = 22,000 LB
Vₜ = 750 FT/SEC

THRUST = 22,000 LB
Vₜ = 850 FT/SEC

FIGURE 2-20 EFFECT OF AIRFOIL SECTION
obtained from whirl tower tests under the same conditions for each set of blades. At the lower tip speed tested (650 fps), the V23010 tends to have a lower SPL, but the reverse is true at 850 fps.

At higher thrusts and/or blade tip speeds, the airfoil section has two significant effects not apparent in the test data described above. One is related to the airfoil stall inception point. A section with the greater stall inception angle would delay the noise increase which accompanies blade stall. The importance of this can be seen from Figure 2-21, taken from Ref. 13, which compares a stalled with an unstalled propeller. The stalled propeller spectrum is approximately 5 dB greater at the mid and high frequencies, although $V_t$ of the stalled propeller is lower.

Another effect is the reduction of "Mach bang" in the transitional Mach number range by the use of thin airfoils. Figure 2-22, from Ref. 4 depicts the advantage of the 8-percent thick VR-7 airfoil over the V23010 at high Mach numbers. At $M = .92$, the difference is 3 dB.

It may be concluded, hence, that very limited control over noise may be exercised by airfoils below the transonic and stall regions since, in those regimes, the effect which the airfoil may produce is not readily discernible to the listener. It is only when the local velocity is greater than $M = 0.85$ or when blade stall is approached, that airfoil characteristics become effective in noise control.
FIGURE 2-21 COMPARISON OF 1/3 OCTAVE BAND SPECTRA FROM STALLED AND UNSTALLED PROPELLERS FROM REFERENCE 13
FIGURE 2-22 EXAMPLE OF THE INFLUENCE OF AIRFOIL SECTION ON HIGH-SPEED BAND FROM REFERENCE 4
Blade Twist

Blades with large spanwise twists such as those of tilting rotors and propellers have lower spanwise blade loadings near the tip than flat blades and display, for the same thrust and tip speed, reduced noise levels when compared to helicopter rotors.

Results of a 6-foot diameter model rotor test of two helicopter rotors with different twist are depicted in Figure 2-23. The Option I rotor incorporated a $-9^\circ$ linear aerodynamic twist; the Option II had a $-7.65^\circ$ linear twist from the root cutout to the 85 percent radius station which increased linearly to $-13^\circ$ at the tip. The Option II configuration is approximately 2 dB quieter at the lower thrust level ($C_T/\sigma = .07$), but at higher thrusts, there is no measurable difference.

The effect of twist on near-field noise of the Boeing-Vertol 160 tilt-rotor is illustrated in Figure 2-24. (This data was measured on the ASD indoor propeller whirl rig; no free-field correction is available). An increase of twist resulted in a decrease in noise at tip speeds of 550 and 750 fps, the decrease being more apparent at the lower collective pitch values. At 900 fps, the $41^\circ$ twist maintains a significant advantage at the lower collective pitch only.

Although the blade twists of Figure 2-24 are too great for helicopter rotors, they are of the magnitude being considered for tilt-rotor aircraft. Examining the plot of $10^\circ$ collective
WIND TUNNEL HOVER
3 BLADED ROTORS
6 FT DIA.
VR-7 AND VR-8 AIRFOILS
TIP SPEED = 725 FPS

\[ C'_{T/\alpha} = 0.07 \]

-9° LINEAR

\[ C'_{T/\alpha} = 0.09 \]

-7.65 LINEAR TO 85% R, LINEAR TO -13° AT TIP

\[ C'_{T/\alpha} = 0.11 \]

SOUND PRESSURE LEVEL - dB re 2 x 10^-5 N/m^2
FREQUENCY - Hz

FIGURE 2-23 BLADE TWIST COMPARISONS
13 FT ROTOR STATIC WHIRL  UNCORRECTED FOR REFLECTION

\[ V_T = 550 \text{ FT/SEC} \]

\[ V_T = 750 \text{ FT/SEC} \]

\[ V_T = 900 \text{ FT/SEC} \]

SOUND PRESSURE LEVEL – dBr 2 x 10^5 N/m^2

TWIST
- 29 DEG
- 36
- 41

FREQUENCY – Hz X 10^-2

MIC. LOC. – 1.6 DIA. FROM AXIS,
14 DEG DOWNSTREAM FROM PLANE OF ROT.

FIGURE 2-24  BLADE TWIST COMPARISON
and tip speed of 750 fps, design tip speed of the Model 222, it is seen that a difference in twist has little effect on the lower four harmonics (first harmonic frequency = 55Hz), but can have a significant effect on the higher harmonics. The 41° twist rotor, for example, shows sound pressure level 3-5dB lower than that with a 36° twist over the frequency range of 400-800 Hz. Therefore, while not affecting the overall sound pressure level, increased twist could appreciably lower the perceived noise level of a rotor with uniform flow.

Other Noise Reduction Techniques

Some research is currently underway in the areas of differential azimuth spacing between blades in the same rotor. Although the results are not definitive at this time, it appears that the overall sound levels generated by rotors with different blade spacings remain constant, with harmonic levels varying for each configuration. For example, a four-bladed rotor with equal spacing between blades displays a strong harmonic at four times the fundamental rotational speed. As the blade spacing approaches an X configuration, the harmonic frequency becomes associated with two times rotational speed. This changes the subjective quality of the sound but does not affect the amplitude of the waveform.

Other devices for reducing noise such as serrated leading edges for airfoils have demonstrated only limited success in specific frequency ranges and at low velocities. However, they have not contributed to significant reductions in the range of Reynolds number where rotors typically operate.
Expanding the tip vortex by additional fluid mass injected into the core region is currently being investigated and these tests show promise of substantial velocity reductions in rotational velocities of the core fluid. Full-scale evaluation of this concept will be conducted on the NASA Langley Whirl Tower in 1973.

III. TURBOSHAFT ENGINE NOISE

There are four sources of turboshaft engine noise: (1) compressor, (2) combustion, (3) turbine, and (4) exhaust jet. The noise emitted from the engine inlet is primarily due to the compressor, while that coming from the exhaust pipe is a combination of combustion, turbine and jet noise. Engine noise has a directivity pattern at the inlet and exhaust originating from the above four sources. Typically, the maximum acoustic output of a front-drive turboshaft engine occurs at 30° from the front of the engine for inlet and at 160° for exhaust. This directivity pattern changes, to a small extent, with power setting because the relative magnitude of the acoustic emissions from the sources are changing.

Using the trends for uninstalled engine noise shown in Figure 3-1, it is obvious that the design objective of minimization of installed power also produces a quieter aircraft.
FIGURE SUMMARY CURVE - BARE VS. SHAFT ENGINE NOISE
Figure 3-2 shows the sound pressure level spectra in .oveL of the Model 222 rotor and Lycoming T53-L13 engine inlet and exhaust. The maximum inlet noise intensity occurs at 30° azimuth, 500 feet radially and 12° below the aircraft. The engine inlet noise, as shown on this figure is lower than the rotor spectrum level over most of the frequency range. The high frequency spike occurring in the 12,500 Hz one-third octave band is caused by fundamental blade passage frequency of the first stage of the engine compressor.

Since the engine inlet only exceeds the rotor spectrum at low frequencies (less than 160 Hz one-third octave band), and at high frequencies (12,500 Hz one-third octave band), the aircraft perceived noise level (95.0 PNdB) does not differ significantly from that of the rotors alone (93.6 PNdB).

The exhaust spectrum shown on Figure 3-2 is lower than the inlet spectrum. Thus, the exhaust is less of a factor in the aircraft perceived noise and overall sound pressure levels than the engine inlet. This is due to the relatively low exit and the resulting jet mixing velocities.

If the rotor acoustic signature (either OASPL or PNL) is significantly reduced, then further reduction of the aircraft acoustic signature may require some treatment of the engine installation. The two engine inlets can be treated by application of a sound absorptive lining to the inner surfaces. The typical attenuation characteristics of these linings are shown in Figure 3-3 (Ref. 14).
Figure 3-2 Comparison of Model 222 Rotor and T53-L13 Inlet and Exhaust Spectra in Hover
As can be seen from this figure, the sound attenuation of these linings are frequency sensitive. Therefore, to obtain an effective inlet lining, the frequency bandwidth in the engine inlet spectra requiring the most attenuation in dB and a lining configuration giving an optimum bandwidth match are selected.

Designing a lining having this optimum bandwidth match can be accomplished by two different methods: one is multiple lining layers having different attenuation spectra and peak attenuation frequencies (Figure 3-4) and the other is a longitudinal series of linings with differing properties either in parallel or in series as shown in Figure 3-5.

These attenuation characteristics are a strong function of the parameters shown in Figure 3-6. The lining geometry parameter length/height (L/H) is changed to obtain the proper attenuation over a specified frequency range. Once the liner material and configuration have been selected.
FIGURE 3-4 ATTENUATION BANDWIDTH COMPARISON

FIGURE 3-5 METHODS OF INCREASING SUPPRESSION BANDWIDTH
To determine the proper geometry, a curve such as shown in Figure 3-7 (Ref. 14) is used for a particular lining material. Refer to Appendix B for a sample calculation of the geometry of an absorptive lining installed in the engine inlets of the Model 222.
The exhaust noise presents no problems for perceived noise level or overall sound pressure level, because the exhaust noise is more than 10 dB below that of the rotor throughout the major portion of the spectrum. If treatment of the exhaust noise is desired, a broadband absorptive muffler may be used.

There may also be a different approach to changing the engine acoustic signature other than the previously discussed treatment of the engine installation. The engine manufacturer could undertake a development program for redesign of the internal components of the engine to reduce the inlet and exhaust noise. This would result in changing the rotor/stator spacing of the compressor and/or redesign combustors to reduce the combustion rumble. Redesign of the engine components is an expensive process compared to sound suppression treatment of the aircraft engine installation. However, this initial expense would be somewhat offset by a reduction in aircraft operating costs since weight penalties for engine component redesign should not be as severe as those for the modification of the engine installation.

IV. ROTOR DESIGN PARAMETER TRADEOFFS

General Discussion

This section of the report deals with one aspect of the design problem: defining tradeoffs in performance and weight associated with the noise reduction. This is done by investigating the sensitivity of the far-field noise levels as well as performance and
weight to variations of the following selected design parameters. The acoustic importance of these parameters have been identified in Section II.

(1) tip speed  
(2) rotor solidity ratio  
(3) number of blades per rotor of constant area per blade  
(4) number of blades per rotor of constant total blade area  
(5) hover disc loading.

It should be realized that although this study is directed toward basically low disc-loading rotary-wing aircraft, the results may not be directly applicable to such other low disc-loading configurations as helicopters. This is due to unique rotor design criteria for a compromise of performance optimization between hover and forward flight.

Sensitivity Study Approach

The establishment of a mission profile, aircraft common performance ground rules, drag trends, and prop-rotor hovering cruise performance are necessary for a comprehensive and systematic study. Aircraft performance, weight and acoustic signature resulting from changing rotor design parameters can now be compared on a common basis.

Acoustic Performance Design Study Mission. - The Boeing-Vertol Model 222 will be presented as a transport aircraft in this report. The mission shown in Figure 4-1 is derived from the above approach.
ACOUSTIC–PERFORMANCE DESIGN STUDY MISSION

1. 5 MINUTES HOVER AT 2500 FEET, 93°F, T/W = 1.1
2. CLIMB AT MIL RATED POWER FROM 2500 FEET TO 10,000 FEET, STANDARD DAY
3. CRUISE .99 BEST RANGE SPEED AT 10,000 FEET, STANDARD DAY
4. TRANSFER ALTITUDE
5. 5 MINUTES HOVER AT 2500 FEET, 93°F, T/W = 1.1
6. CLIMB AT MIL RATED POWER FROM 2500 FEET TO 10,000 FEET, STANDARD DAY
7. CRUISE .99 BEST RANGE SPEED AT 10,000 FEET, STANDARD DAY
8. TRANSFER ALTITUDE TO 2500 FEET, 93°F
9. END MISSION WITH 10% OF INITIAL FUEL.

FIGURE 4-1 ACOUSTIC PERFORMANCE DESIGN STUDY MISSION
and thus, is not meant to show its military capability. In this section, the gross weight of the aircraft will not change from its current 12,000 pounds. Therefore, the payload will vary as the Model 222 is affected by the parameter changes.

Common Performance Ground Rules. - According to the intent of this study, the Boeing Vertol Model 222, as defined in Ref. 1, is selected as the baseline aircraft. All of the aircraft resulting from the modification of the baseline model should fulfill the following requirements:

1. All aircraft shall have a gross weight of 12,000 pounds.

2. Engine maximum rating will be equal to hover power required at a thrust-to-weight ratio = 1, IGE Sea Level Standard Day, with one engine inoperative.

3. Transmission torque limit is to be sized for rotor hover rpm and one engine maximum power at 2,500 feet, 93°F, static condition.

The above requirements do not necessarily result in a specific maximum level flight speed of the Model 222 such as 300 knots at 10,000 feet, Standard Day, at hover rpm. Instead, the maximum level flight speed will vary from changes in drag, transmission torque limit and propulsive efficiency.

The manufacturer's engine fuel flow shall be increased by 5 percent in accordance with MIL C-5011A.
The equivalent flat plate trend shown in Figure 4-2 was derived from the "Minimum Parasite Drag Breakdown" of the Model 222 (Ref. 1, Table 7, p. 126). The parasite drag is presented as a function of wing area only, as moment arms and tail volume coefficients are assumed constant.

This study investigates the influence of the 5 rotor design parameters.

1. Hover tip speed, $V_{th}$, at constant $C_T/\sigma$, $B$, $W/A$
2. Rotor solidity ratio, $\sigma$, at constant $V_{th}$, $B$, $W/A$
3. Number of blades per rotor, $B$, at constant $\sigma/B$, $V_{th}$, $W/A$
4. Number of blades per rotor, $B$, at constant $\sigma$, $V_{th}$, $W/A$
5. Hover disc loading, $W/A$, at constant $C_T/\sigma$, $V_{th}$, $B$.

To isolate the effects of each parameter, sensitivity was studied by changing one parameter while the others remained constant. Two off-baseline values for the five parameters were selected, with the baseline providing the third point. In this way, the sensitivity curve was defined in each case. The off-baseline values were expected to reduce the acoustic signature of the aircraft hovering out-of-ground effect.

The wing chord and thickness do not change and thus, wing area becomes a function of rotor diameter, since fuselage width and rotor-fuselage clearance remain constant. This approach is acceptable, since there is no need to maintain particular wing chord-to-diameter or rotor area-to-wing area ratios. For tilting-rotor aircraft, the wing does not have to support a significant portion
FIGURE 4-2 PARASITE DRAG TREND FOR ACOUSTIC-PERFORMANCE DESIGN STUDY
of the aircraft weight in low speed (near-hovering) flight.
This permits the designer to optimize the wing geometry for airplane configuration cruise flight, while complying with the following constraints: the need for placing the rotors at the wing tips and to provide sufficient wing thickness to support the aircraft in the helicopter mode.

The rotor hover and cruise performance trends were established by the following calculation procedure. At each condition, a performance evaluation for a change in twist was calculated using the Boeing Rotor Performance Computer Program. This computer program uses a vortex wake analysis plus an empirical slipstream correction factor (see Ref. 15). The procedure outlined below assured that all rotor designs would have the same design tradeoff between hover and cruise performance. The original Model 222 blade twist increment varying linearly along the blade span was added. The manner in which the optimum twist was obtained is illustrated in the sketch below.

**OPTIMUM TWIST**

[Diagram showing maximum figure of merit, design point, varying design parameter, etc.]
The other blade characteristics such as blade chord, thickness ratio and airfoil section distribution were kept the same as the baseline rotor (Ref. 1). Also, the rotor design operating conditions were selected to be the same as those for the Model 222 aircraft; i.e., hover at 2500 feet, 93°F, 12,000 pounds gross weight and thrust-to-weight equal to 1.05; and cruise at 300 knots, 10,000 feet standard day. Cruise thrust was based on the Model 222 drag in the cruise condition, while cruise tip speed was assumed to be 70 percent of the tip speed of the modified aircraft in hover.

The above procedure avoided excessive rotor design iterations which would have been necessary to define the optimum hover-cruise compromise twist. Thus, the rotor parameters in the sensitivity matrix were defined. Cruise performance was calculated using the same Boeing rotor performance computer program.

Discussion of Results of Tilt-Rotor Sensitivity to the Five Design Parameters

**Structural Flight Envelope Limits.** - The weight empty of the aircraft in this section will be dependent on the following structural limits which are the same as for the baseline aircraft.

1. \( V_{MO} = 350 \text{ knots, EAS} \)
2. \( V_{dive} = 350 \text{ knots, EAS} \)
3. \( M_{MO} = 0.569 \)
4. Wing Design Maneuver Load Factor = 3g.
In general, when the parameters were changed in a manner indicating an improvement in acoustic characteristics, the result reduced aircraft performance and increased its weight empty. The exceptions to this generalization were the following parameter variations: (1) rotor hover disc loading, and (2) number of blades with constant rotor solidity. The aircraft design points establishing the sensitivity lines have their summary weight statements and configurations tabulated in Appendix D.

**Rotor Performance.** - The sensitivity of the rotor performance to the variation of the design parameters is shown in Figure 4-3 for hover and Figure 4-4 for cruise performance. The changes in rotor performance shown in these figures result from the following: increased profile power for changes in rotor solidity whether or not the number of blades were changed from the baseline; and twist, changing the L/D of the rotor blade by altering the spanwise lift distribution for tip speed and hover disc loading.

**Aircraft Performance.** - Figures 4-5 and 4-6 show the rate of change of shaft horsepower required to hover out-of-ground effect at a T/W = 1.05 and 2500 feet altitude, 93°F, with respect to the five design parameters.

Tip speed variation at a constant C_T/α causes the hover power required to increase as V_T is reduced. This is due to the compromise in spanwise lift distribution necessary to maintain the desired
NOTES:
1. HOVER @ 2500 FT / 93°F
2. THRUST HELD CONSTANT.

EFFECT OF TIP SPEED
CONSTANT CT/o, B, & W/A

EFFECT OF SOLIDITY RATIO
CONSTANT VTIP, B, W/A

EFFECT OF NUMBER OF BLADES
CONSTANT VTIP > W/A
AND | OR |/B

EFFECT OF DISC LOADING
CONSTANT VTIP, B, & CT/o

FIGURE 4-3 CHANGE IN HOVER FIGURE OF MERIT WITH VARIOUS ROTOR DESIGN PARAMETERS
NOTES: 1. CRUISE @ 10,000 FT/STD & 300 KT
2. THRUST HELD CONSTANT.

![Diagram of Effect of Tip Speed](image)

**EFFECT OF TIP SPEED**

- **CONSTANT CT/α, B, & W/A**
- **M222 DES. PT.**
- **3 BLADES D = 26 FT. CRUISE V_TIP IS 70% HOVER V_TIP**
- **EFFECT ON SOLIDITY RATIO**
- **CONST. V_{TIP}, B, W/A**
- **M222 DES. PT.**
- **V_{TIP} = 525 FPS**
- **C_T = .0247**

![Diagram of Effect of Number of Blades](image)

**EFFECT OF NUMBER OF BLADES**

- **CONST. V_{TIP}, W/A, AND α OR α/B**
- **M222 DES. PT.**
- **CONST. α**
  - **(α = .1154)**
- **CONST. α/B**
  - **(α/B = .0385)**

![Diagram of Effect of Hover Disk Loading](image)

**EFFECT OF HOVER DISK LOADING**

- **CONST. V_{TIP}, B, C_{T/α}**
- **M222 DES. PT.**

**FIGURE 4-4** CHANGE IN CRUISE PROPULSIVE EFFICIENCY WITH VARIOUS ROTOR DESIGN PARAMETERS
NOTES: 1. GW = 12,000 LBS. 2. T/W = 1.05. 3. HOVER @ 2,500 FT. 93°F 4. HOVER O.G.E.

FIGURE 4-5 EFFECT OF HOVER TIPSPEED, ROTOR SOLIDITY AND DISC LOADING ON HOVER SHP REQUIRED.
NOTES: 1. GW = 12,000 LBS.
2. T/W = 1.05
3. HOVER @ 2,500 FT, 93°F
4. ROTOR DIAMETER = 26 FT
5. HOVER O.G.E.

![Graph showing effect of number of blades at constant σ and σ/B on hover SHP required.](image)

FIGURE 4-6 EFFECT OF NUMBER OF BLADES AT CONSTANT σ AND σ/B ON HOVER SHP REQUIRED
The need to change total twist is explained by the use of the following sketch of generalized prop-rotor performance.

This sketch represents generalized performance for small changes in rotor solidity, while the following characteristics are constant: (a) airfoil section, (b) twist distribution, and (c) planform. A hover tip speed reduction at constant $C_T/\alpha$ resulting in a reduced cruise $V_t(0.7V_{th})$ causes the cruise design point to move horizontally (arrow in chart) as design cruise speed is fixed. The degraded cruise performance is caused by an increased inflow ratio altering the angles of attack along the blade; therefore, changing the lift distribution. This indicates
a deviation from the hover-cruise performance trade slope corresponding to a hover payload/mission fuel tradeoff. Thus, the twist must be altered to maintain the study's prop-rotor performance ground rules. If the rotor were to be designed for hover only, the twist could have been selected to maximize the Figure of Merit. However, cruise efficiency at the design cruise speed would be so poor that unacceptable aircraft performance would result.

Increasing rotor blade solidity ratio at a constant tip speed, either with a constant or increasing number of blades per rotor, requires a larger hover power caused by increasing rotor profile power in direct proportion to solidity (Figures 4-5 and 4-6).

The two parameters reducing the hover power required are: increasing the number of blades per rotor at constant solidity, and reduction of disc loading at constant \( C_T/\sigma \). When the number of blades is increased, a small change occurs (\(-8 \text{ HP/blade}\)) which is less than 1 percent of the total power required to hover. This can be attributed to a change in the blade chord-to-radius ratio and its associated induced power because of the improved tip loss factor.

With respect to disc loading, the hover power required changes at a rate of 47.5 HP p.r lb/ft\(^2\) as the induced power becomes smaller. The induced power has the following relationship to disc loading:
\[
\frac{\delta \text{RHPind}_h}{\delta (W/A)} = \frac{W}{560} \left( \frac{T}{W} \right)^{3/2} \left\{ \text{kind}_h + \frac{\partial^2 \text{kind}_h}{\partial (W/A)^2} \right\}
\]

where \( \text{kind}_h \) is the nonuniform inflow and tip-induced power loss factor.

A further discussion of the calculation of hovering performance is contained in Appendix C.

In this study, payload is calculated on the basis of mission fuel required for 100 n.mi. radius and a takeoff gross weight. The sensitivity of payload to changes in the five design parameters is shown in Figures 4-7 through 4-11. From Figure 4-7, it appears that payload is directly proportional to tip speed. The mission fuel is increased because of reduced cruise efficiency (Figure 4-4) and Figure-of-Merit (Figure 4-3). Weight empty is also growing (Table I, Appendix D). The same factors result in the inverse proportion of payload to rotor solidity whether payload changes as a result of the varying number of blades per rotor, or the blade area at the same number of blades.

Changing the number of blades at constant solidity has no significant impact on mission payload. Increasing the blade number from 3 to 4 causes the payload to grow by 26 pounds, or 2 percent of the original payload; i.e., .22 percent of the gross weight. Another increase from 4 to 5 blades decreases the payload by 4 pounds, or .31 percent of the payload. For details, see Table I, Appendix D.
FIGURE 4-7 EFFECT OF ROTOR TIP SPEED ON PAYLOAD RADIUS
NOTES:  
1. GW = 12,000 LBS  
2. ACOUSTIC-PERFORMANCE DESIGN STUDY MISSION  
3. HOVER V_TIP = 750 FPS  
4. NO. OF BLADES/ROTOR = 3  
5. W/A = 11.3 LB/FT^2

FIGURE 4-8 EFFECT OF ROTOR SOLIDITY ON PAYLOAD-RADIUS
EFFECT OF NUMBER OF BLADES OF CONSTANT TOTAL BLADE AREA ON PAYLOAD - RADIUS

NOTES:
1. GW = 12,000 LBS
2. ACOUSTIC-PERFORMANCE DESIGN STUDY MISSION
3. HOVER V_TIP = 750 FPS
4. \( c = 0.1154 \)
5. W/A = 11.3 LB/FT^2

FIGURE 4-9 EFFECT OF NUMBER OF BLADES OF CONSTANT AREA PER BLADE ON PAYLOAD RADIUS
NOTES: 1. GW = 12,000 LBS
2. ACOUSTIC - PERFORMANCE DESIGN STUDY MISSION
3.  \(\alpha/BLADE = 0.0385\)
4. HOVER \(V_{\text{TIP}} = 750\) FPS
5.  \(W/A = 11.3\) LB/FT\(^2\)

FIGURE 4-10 EFFECT OF NUMBER OF BLADES OF CONSTANT AREA PER BLADE ON PAYLOAD-RADIUS
FIGURE 4-11 EFFECT OF DISC LOADING ON PAYLOAD-RADIUS
From Figures 4-11 and 4-15, it can be seen that the variation of payload with disc loading reaches a maximum at \((W/A) = 9 \text{ lbs/sq ft}\). This maximum is caused by a decrease and then an increase in weight empty while mission fuel is becoming smaller in a linear manner.

The hover rpm transmission limit determines the maximum level flight speed of the Boeing Vertol Model 222 at 10,000 feet Standard day (Figure 63, Ref. 1). This is due to the fact that the engine power available at cruise speed, 10,000 feet Standard Day is greater than the power that can be safely input for extended periods of time to the gearbox at the hover rpm. Even though a reduced tip speed in cruise increases the propulsive efficiency of the prop-rotor because of a better rotor blade L/D, the transmission power limit at cruise rpm (70 percent of hover rpm) becomes lower. The transmission limit is set by the torque transmitted and thus, for constant torque, as rpm is reduced the power transmitted becomes smaller (30 percent). The corresponding improvement in propulsive efficiency resulting from the lower tip speed is approximately 20 percentage points. However, this does not overcome the 30 percent loss of power available. Therefore, the maximum propulsive power available occurs at the hover tip speed.

Figures 4-12 and 4-13 show the impact of the 5 design parameters on the following characteristics which influence maximum
NOTES: 1. GW = 12,000 LB  2. 10,000 FT STANDARD DAY
3. ROTOR DIAMETER = 20'ft

FIGURE 4-13 VARIATION OF NO. OF BLADES WITH .99 BEST RANGE SPEED AND TRANSMISSION LIMIT SPEED
level flight speed: (a) equivalent flat plate drag, (b) transmission power limit (determined from common performance ground rules), and (c) propulsive efficiency. The equivalent flat plate drag and wing span are constant, but the transmission limit undergoes small changes with the following four parameters (Table 2, Appendix D):

- tip speed
- rotor solidity
- number of blades of constant area per blade
- number of blades of constant total area.

However, their influence on the $V_{\text{max}}$ is considerable due to the variation in propulsive efficiency.

Hover rpm transmission limit speed is directly proportional to disc loading (Figure 4-12). This results from the fact that common performance ground rules state that the transmission limit is determined by hover power required. As shown previously, hover power becomes smaller as disc loading decreases. Parasite drag increases as a function of the increasing wing area which, in turn, is inversely proportional to the square root of the disc loading.

The wing geometry changes because the chord is constant while the wing span varies so that the wing tip rotors maintain a constant fuselage clearance. Both the transmission power limit and parasite drag deviations from the baseline values (Table 2, Appendix D) have a greater influence on the $V_{\text{max}}$ level than changes of propulsive efficiency and induced drag.
The .99 best range speed is dependent on engine fuel flow which, in turn, is a function of sfc and power. The latter depends on the tilt-rotor's speed and equivalent lift-to-drag ratio \( (L/D_e) \). The specific range performance of the Model 222 aircraft is shown in Figure 70 of Ref. 1. The changes from the baseline performance of .99 best range speed are shown in Figures 4-12 and 4-13.

The performance sensitivity of the .99 best range speed to all of the design parameters except disc loading results primarily from variations of propulsive efficiency and installed power (Table 2, Appendix D). Therefore, the magnitude of the changes in the .99 best range speed from the baseline level, as caused by varying tip speed, rotor solidity at constant blade number, and increased number of blades are relatively small. The largest change (12 knots) from the baseline performance occurs when hover tip speed is reduced from 750 fps to 550 fps.

However, .99 best range speed is a very strong function of disc loading as shown in Figure 4-12. This is due to wing geometry changes as a function of disc loading. The wing loading and span loading are directly proportional and parasite drag is inversely proportional to disc loading. These factors determine the lift/drag ratio of an aircraft and in turn, the best range speed. The above factors affect the maximum \( L/D \) speed as shown below:

\[
\begin{align*}
wf &= \text{gross weight}/fe \quad \text{equivalent flat plate area loading in lb/ft}^2 \\
wb &= \text{gross weight}/b^2 \quad \text{span loading in lb/ft}^2
\end{align*}
\]
AirCraft Weight SensitivitY. - Sensitivity of weight empty to the five design parameters are shown in Figures 4-14 through 4-16. In addition, the summary weight statements for all the aircraft used to define the sensitivity trends are in Table I, Appendix D. The weights of the baseline aircraft subsystems were either determined from vendor information, statistical trends, or analysis of engineering drawings. The variation of weight empty from that of the baseline is determined from the weight trends created for the VASCOMP computer program (Ref. 16). These trends are substantiated in a separate document.

Except for disc loading and number of blades with constant rotor solidity, large changes in weight empty occurred as a function of the design parameter variation.

In the case of disc loading variation, the trend indicated that a weak minimum for weight empty occurs at W/A ≈ 9 lbs/sq.ft. (Figure 4-15). The unusual sensitivity relationship of weight empty to disc loading is caused by the variation of the drive system weight, which is a function of the torque limit calculated in the following way:

\[ \text{Torque} = \frac{(HP \times 550 \times R)}{V_t} \]

So, for the decreasing disc loading at a constant tip speed used in this study, the rotor radius is increasing but the horsepower limit decreases rapidly, then levels off. This causes the
Figure 4-14  EFFECT OF HOVER TIP SPEED AND ROTOR SOLIDITY RATIO ON WEIGHT OF TILT-ROTOR SUBSYSTEMS
FIGURE 4-15 EFFECT OF DISC LOADING AND NO. OF BLADES WITH CONSTANT TOTAL AREA ON WEIGHT OF TILT-ROTOR SUBSYSTEMS
FIGURE 4-16 EFFECT OF NO. OF BLADES WITH CONSTANT AREA/BLADE ON WEIGHT OF TILT-ROTOR SUBSYSTEMS
transmission weight trend shown in Figure 4-15. The importance of this trend is that the increasing weight of the wing and tail group is balanced by the declining weight of the rotor system flight controls and engines (Table I, Appendix D).

With respect to the variation in the number of blades at constant solidity, the weight empty remains almost constant (Figure 4-15). This is due to the fact that when the number of blades increases while keeping rotor solidity constant, the weights of the rotor and flight control systems are increasing at a faster rate than the rate at which the wing group, engines and drive system weights are declining. However, the magnitude of the variation, ranging from -.208 to .0584 percent of gross weight per blade, is so small that one could neglect it and say that there was no change in weight empty as a function of number of blades at constant rotor solidity.

Weight empty is inversely related to hover tip speed. This is caused by the increasing weight of the engines, rotor system, drive system and flight controls which is only partially offset by a decrease in wing weight due to increased weight relief of wing bending moments.

Increasing rotor solidity for either constant or varying number of blades per rotor causes weight empty to grow. The rate of increase in this case is a result of a large growth in rotor system and flight control system weight and to a smaller extent due to an increase in engine and drive system weight. As in the
previous case, the component weight increases have been somewhat offset by a decrease in wing group weight. Increasing blade number with constant area per blade increases the weight empty by 395 pounds for each additional blade over three.

The airframe weight shown in Figures 4-14 through 4-16 is that defined in the AMPR (Aircraft Manufacturers Planning Report) which is one section of the Cost Information Report.

**Far-Field Acoustic Signature.** - The acoustic signature discussed in this section occurs under the following conditions:

a. hover OGE
b. noise source (rotors) 100 feet above and 500 feet away from the observer
c. sea level, standard day.

The prediction method used in this section for rotor acoustic signature is based on that used in a previous Boeing-Vertol study (Ref. 2). However, the rotational noise loading law used in the above study has been modified by wind-tunnel test results (Ref. 17). The new loading law is shown in Figure 4-17. A discussion of the accuracy of the theoretical tilt-rotor acoustic model is contained in Appendix E.

The tradeoff slope or sensitivity of overall sound pressure level to the variation of the 5 design parameters is shown in Figures 4-18 and 4-19. The first harmonic of blade passage frequency is a component of the acoustic signature determining, to a large extent, the overall sound pressure level. The magnitude of this major component is a function of:
\[ n = X_{P_{\phi}}^{.347} \]

**Figure 4-17** Rotor Force Harmonic Decay \((n)\) as a function of operating condition.
Figure 4-18 Effect of 3 Design Parameters on Perceived Noise Level and Overall Sound Pressure Level

Notes: 1. GW = 12,000 lb
2. Hover O.G.E. @ 30 ft
3. Distance from aircraft = 500 ft

Pressure Level: dB RE: 2 x 10^-5 N/m²; Perceived Noise Level; PNdB Overall Sound
NOTES: 1. GW = 12,000 LB
2. HOVER OGE @ 30 FT
3. DISTANCE FROM AIRCRAFT = 500 FT

FIGURE 4.19 EFFECT OF NO. OF BLADES ON PERCEIVED NOISE LEVEL AND OVERALL SOUND PRESSURE LEVEL
(a) tip speed
(b) thrust
(c) number of blades per rotor
(d) rotor power
(e) rotor radius
(f) height and distance of source from the observer.

In this study, all acoustic aspects were investigated for constant height (100 feet) and distance (500 feet). However, it should be realized that the relative influence of power and thrust change with the height of the source over the observer for a constant horizontal distance. The rotor power becomes a more important factor as the observer approaches the tip path plane. By contrast, the sensitivity of overall sound pressure level would not be affected by variations in hover power required if the observer was 500 feet directly beneath the rotor.

Broadband noise of the rotor acoustic signature is approximately 15-20 dB below the sound pressure level of the first harmonic of blade passage frequency. Thus, the overall sound pressure level is less sensitive to variation of the broadband noise sound pressure level than that of the first harmonic. Overall sound pressure level shows the smallest sensitivity to the variation of rotor solidity at a constant number of blades. Solidity varying from .1154 to .15 causes a .1 dB decrease; then, a further increase in solidity from .15 to .20 results in an increase of .4 dB.

The trend for tip speed reduction shown on Figure 4-18 indicates that the overall sound pressure level decreases with a reduced tip speed, but at smaller increments as the tip speed becomes lower.
Increasing the number of blades per rotor either at a constant or changing rotor solidity, reduced the overall sound pressure level (Figure 4-19).

Figures 4-18 and 4-19 show the effects of the variation of 5 design parameters on perceived noise level. It can be seen that the tip speed at constant CT/\(\rho\) is the most effective parameter for reducing the perceived noise level from that of the baseline aircraft. A 37.5 percent reduction in tip speed decreases the perceived noise level by 6.6 PNdB. Increasing the number of blades of constant area per blade is the second most effective parameter for reducing noise.

The effectiveness of the tip speed reduction is due to the spectrum shift to a lower frequency combined with lowering of the spectrum sound pressure level. The frequency weighting of perceived noise level requires that the broadband noise component be weighted more than that of the rotational noise. This depends on the magnitude of the fundamental blade passage frequency. Therefore, as the rotor becomes larger in diameter at a constant tip speed, the rpm drops, and thus, the dependence of PNL on the broadband component grows.

The relative tolerances of the tradeoffs discussed in this section are included in Appendix F.
Additional Design Change. - A promising design parameter other than those analyzed in the previous sections is to be selected for study. The selection process required the study and comparison of unusual or new rotor noise reduction design concepts, and determination of the most promising.

The special devices studied for possible inclusion in this report were:

1. Vertical rotor blade spacing in various planes of rotation and unequal blade azimuth spacing.

2. Tip blowing (linear mass injection causing the rapid tip vortex decay).

3. Owl Wing (leading edge serrations to favorably affect the boundary layer).

4. Special blade tips (discussed in Section II).

The investigation led to the following information about the above-listed concepts. There are some indications that a more favorable tip vortex separation can be achieved by vertical and unequal azimuth blade spacing (Ref. 18). However, no acoustic test data has been published on these concepts as of this writing. John Ward, Principle Investigator, NASA Langley Research Center, indicated that recent acoustic measurements on a small model did not indicate significant changes from the acoustic signature of a conventional arrangement.
Test data on tip blowing has been published (Ref. 19), but the test results refer to a fixed-wing at relatively low tunnel speeds. The tests were performed primarily to determine the tip vortex behavior and very little acoustic information was presented. This concept of tip blowing is scheduled to be tested on a full-scale rotor (UH-1) at Langley Research Center later in 1973.

The owl wing showed very little noise reduction potential at conventional rotor Reynolds Numbers although some benefits were derived at very low Reynolds Numbers (Ref. 20). The maximum reduction obtained was 4 dB OASPL at a very large collective pitch setting of 18° on a zero twist rotor and a Reynolds Number at the tip of 1.59 x 10^6.

Tip shapes have been extensively discussed in published literature and a large amount of test data has been taken. However, when test data from various sources (Refs 10, 11, and 13) are compared objectively, there does not seem to be any clear-cut improvement over the conventional square tip with a thin airfoil section.

Hence, from the investigation of unconventional rotor designs, it may be concluded that the technical data available were not sufficient to permit ranking of the devices discussed above, nor could an evaluation and comparison of their relative acoustic effectiveness be performed.
V. PARTIAL DERIVATIVES WITH RESPECT TO OPERATIONAL AND DESIGN PARAMETERS

General Remarks

A linearized approach in establishing trends in performance and/or weight tradeoffs vs noise is quite attractive, since it would permit one to superimpose the influences of various operational and/or design parameters. In order to apply this approach, one must know, at the point of interest, values of partial derivatives of various significant quantities with respect to either operational or design parameters. Seven tables of such derivatives in nondimensional form, are presented in this section.

At this point, one must be cautioned that due to strong nonlinearities which may be associated with large excursions in the parameter values, validity of trends established by the linearized approach may be limited to the immediate neighborhood of the point of interest. However, large design parameter excursions from the baseline were necessary to achieve acoustically significant reductions (3dB, 3PNdB) as shown in Figures 4-18 and 4-19. One should also realize that although the derivatives are presented in nondimensional form, the values cannot be considered as general. They were established on the basis of one particular aircraft only, and consequently, they must be applied with caution to other rotary-wing configurations and even to other tilt-wing aircraft.
A comparison of OASPL nondimensional derivatives between test and prediction (Table 5-6) can be made for thrust and tip speed. The test data slopes are taken from Figures 2-5 and 2-10, and the resulting nondimensional derivatives are .28 for tip speed and .084 for thrust. The predicted derivatives (Table 5-6) agree fairly well with measured ones, considering that the measured SPL of the fundamental blade passage frequency was assumed to be directly proportional to the OASPL in the calculation of the measured data derivatives.
<table>
<thead>
<tr>
<th>NONDIMENSIONAL PARTIAL DERIVATIVES</th>
<th>VALUE</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \frac{\Delta \text{HOVER SHP REQ}}{\Delta V_{th}} ) ( \frac{\Delta V_{th}}{V_{th}} )</td>
<td>-.122</td>
</tr>
<tr>
<td>( \frac{\Delta \text{HOVER SHP REQ}}{\Delta c} ) ( \frac{\Delta c}{c} )</td>
<td>.1049</td>
</tr>
<tr>
<td>( \frac{\Delta \text{HOVER SHP REQ}}{\Delta B} ) ( \frac{\Delta B}{B} ) ( \sigma = .1154 )</td>
<td>-.027</td>
</tr>
<tr>
<td>( \frac{\Delta \text{HOVER SHP REQ}}{\Delta B} ) ( \frac{\Delta B}{B} ) ( \sigma/B = .0385 )</td>
<td>.0803</td>
</tr>
<tr>
<td>( \frac{\Delta \text{HOVER SHP REQ}}{\Delta T} ) ( \frac{\Delta T}{T} )</td>
<td>1.31</td>
</tr>
<tr>
<td>( \frac{\Delta \text{HOVER SHP REQ}}{\Delta V_{th}} ) ( \frac{\Delta V_{th}}{V_{th}} ) ( c_T/c = .0896 )</td>
<td>-.118</td>
</tr>
<tr>
<td>( \frac{\Delta \text{HOVER SHP REQ}}{\Delta (\Delta W/A)} ) ( \frac{\Delta (\Delta W/A)}{(\Delta W/A)} ) ( c_T/c = .0896 )</td>
<td>.149</td>
</tr>
</tbody>
</table>

**TABLE 5-1**

SENSITIVITY OF HOVER SHP \( \text{REQ} \) @ 2500'93°F
TO DESIGN PARAMETER CHANGES

94.
<table>
<thead>
<tr>
<th>NonDimensional Partial Derivatives</th>
<th>Value</th>
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<tbody>
<tr>
<td>( \frac{\Delta v_{th}}{v_{th}} )</td>
<td>( \frac{\Delta c}{c} )</td>
</tr>
<tr>
<td>( \frac{\Delta c}{c} )</td>
<td>( \frac{\Delta e}{e} )</td>
</tr>
<tr>
<td>( \Delta \frac{\Delta c}{c} )</td>
<td>( \Delta \frac{\Delta e}{e} )</td>
</tr>
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<td>( \Delta \frac{\Delta c}{c} )</td>
<td>( \Delta \frac{\Delta e}{e} )</td>
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<tr>
<td>( \Delta \frac{\Delta c}{c} )</td>
<td>( \Delta \frac{\Delta e}{e} )</td>
</tr>
<tr>
<td>( \Delta \frac{\Delta c}{c} )</td>
<td>( \Delta \frac{\Delta e}{e} )</td>
</tr>
</tbody>
</table>

Table 5-2

Sensitivity of .99 Best Range Speed to Design Parameter Changes
<table>
<thead>
<tr>
<th>NONDIMENSIONAL PARTIAL DERIVATIVES</th>
<th>VALUE</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\frac{\Delta TLS}{TLS} / \frac{\Delta V_{th}}{V_{th}}$</td>
<td>-.0369</td>
</tr>
<tr>
<td>$\frac{\Delta TLS}{TLS} / \frac{\Delta \sigma}{\sigma}$</td>
<td>-.153</td>
</tr>
<tr>
<td>$\frac{\Delta TLS}{TLS} / \frac{\Delta B}{B} \big</td>
<td>_{\sigma=.1154}$</td>
</tr>
<tr>
<td>$\frac{\Delta TLS}{TLS} / \frac{\Delta B}{B} \big</td>
<td>_{\sigma/B=.0385}$</td>
</tr>
<tr>
<td>$\frac{\Delta TLS}{TLS} / \frac{\Delta T}{T}$</td>
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</tr>
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<td>$\frac{\Delta TLS}{TLS} / \frac{\Delta V_{th}}{V_{th}} \big</td>
<td>_{C_T/\sigma=.0896}$</td>
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<tr>
<td>$\frac{\Delta TLS}{TLS} / \frac{(\Delta W/A)}{(W/A)} \big</td>
<td>_{C_T/\sigma=.0896}$</td>
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**TABLE 5-3**

SENSITIVITY OF HOVER RPM TRANSMISSION LIMIT SPEED TO DESIGN PARAMETER CHANGES
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<th>NONDIMENSIONAL PARTIAL DERIVATIVES</th>
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<td>$\frac{\Delta W}{W_e} / \frac{\Delta B}{B} \mid \sigma = .1154$</td>
<td>-.00816</td>
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<tr>
<td>$\frac{\Delta W}{W_e} / \frac{\Delta B}{B} \mid \sigma/B = .0385$</td>
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<tr>
<td>$\frac{\Delta W}{W_e} / \frac{\Delta T}{T}$</td>
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</tr>
<tr>
<td>$\frac{\Delta W}{W_e} / \frac{\Delta V_{th}}{V_{th}} \mid C_T/\sigma = .0896$</td>
<td>-.248</td>
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<tr>
<td>$\frac{\Delta W}{W_e} / \frac{(\Delta W/A)}{(W/A)} \mid C_T/\sigma = .0896$</td>
<td>.00007</td>
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**TABLE 5-4**

SENSITIVITY OF WEIGHT EMPTY TO DESIGN PARAMETER CHANGES
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<th>NONDIMENSIONAL PARTIAL DERIVATIVES</th>
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<td>( \frac{\Delta \text{PAYLOAD}}{\text{PAYLOAD}/V_{th}} )</td>
<td>.184</td>
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<tr>
<td>( \frac{\Delta \text{PAYLOAD}}{\text{PAYLOAD}/\sigma} )</td>
<td>-1.078</td>
</tr>
<tr>
<td>( \frac{\Delta \text{PAYLOAD}}{\text{PAYLOAD}/B} \bigg</td>
<td>_{o=.1154} )</td>
</tr>
<tr>
<td>( \frac{\Delta \text{PAYLOAD}}{\text{PAYLOAD}/B} \bigg</td>
<td>_{o/B=.0385} )</td>
</tr>
<tr>
<td>( \frac{\Delta \text{PAYLOAD}}{\text{PAYLOAD}/T} )</td>
<td>.804</td>
</tr>
<tr>
<td>( \frac{\Delta V_{th}}{V_{th}} \bigg</td>
<td>_{C_T/o=.0896} )</td>
</tr>
<tr>
<td>( \frac{\Delta (W/A)}{\text{PAYLOAD}/(W/A)} \bigg</td>
<td>_{C_T/o=.0896} )</td>
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</tbody>
</table>

**TABLE 5-5**

SENSITIVITY OF MISSION PAYLOAD TO DESIGN PARAMETER CHANGES

98.
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<th><strong>NONDIMENSIONAL PARTIAL DERIVATIVES</strong></th>
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<tr>
<td>( \frac{\Delta \text{OASPL}}{\text{OASPL}} \left/ \frac{\Delta V_{th}}{V_{th}} \right. )</td>
<td>.243</td>
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<td>( \frac{\Delta \text{OASPL}}{\Delta \sigma} \left/ \frac{\text{OASPL}}{\sigma} \right. )</td>
<td>-.00721</td>
</tr>
<tr>
<td>( \frac{\Delta \text{OASPL}}{\Delta B} \left/ \frac{\text{OASPL}}{B} \right</td>
<td>_{\sigma=.1154} )</td>
</tr>
<tr>
<td>( \frac{\Delta \text{OASPL}}{\Delta B} \left/ \frac{\text{OASPL}}{B} \right</td>
<td>_{\sigma/B=.0385} )</td>
</tr>
<tr>
<td>( \frac{\Delta \text{OASPL}}{\Delta T} \left/ \frac{\text{OASPL}}{T} \right. )</td>
<td>.1038</td>
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<td>( \frac{\Delta \text{OASPL}}{\Delta V_{th}} \left/ \frac{\text{OASPL}}{V_{th}} \right</td>
<td><em>{C</em>{T}/\sigma=.0896} )</td>
</tr>
<tr>
<td>( \frac{\Delta \text{OASPL}}{\Delta (W/A)} \left/ \frac{\text{OASPL}}{(W/A)} \right</td>
<td><em>{C</em>{T}/\sigma=.0896} )</td>
</tr>
</tbody>
</table>

**TABLE 5-6**

SENSITIVITY OF OVERALL SOUND PRESSURE LEVEL TO DESIGN PARAMETER CHANGES
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<th>NONDIMENSIONAL PARTIAL DERIVATIVES</th>
<th>VALUE</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \frac{\Delta PNL}{PNL} / \frac{\Delta V_t}{V_{th}} )</td>
<td>.176</td>
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<tr>
<td>( \frac{\Delta PNL}{PNL} / \frac{\Delta \sigma}{\sigma} )</td>
<td>-.0213</td>
</tr>
<tr>
<td>( \frac{\Delta PNL}{PNL} / \frac{\Delta B}{B} \bigg</td>
<td>_{\sigma = .1154} )</td>
</tr>
<tr>
<td>( \frac{\Delta PNL}{PNL} / \frac{\Delta B}{B} \bigg</td>
<td>_{\sigma/B = .0385} )</td>
</tr>
<tr>
<td>( \frac{\Delta PNL}{PNL} / \frac{\Delta T}{T} )</td>
<td>.1026</td>
</tr>
<tr>
<td>( \frac{\Delta PNL}{PNL} / \frac{\Delta V_{th}}{V_{th}} \bigg</td>
<td>_{CT/\sigma = .0896} )</td>
</tr>
<tr>
<td>( \frac{\Delta PNL}{PNL} / \frac{(\Delta W/A)}{(W/A)} \bigg</td>
<td>_{CT/\sigma = .0896} )</td>
</tr>
</tbody>
</table>

**TABLE 5-7**

SENSITIVITY OF PERCEIVED NOISE LEVEL TO DESIGN PARAMETER CHANGES
VI. CONCEPTUAL DESIGN OF QUIET TILT-ROTOR AIRCRAFT

Approach

This section covers the problems of reducing the signature of the baseline rotor design by: (1) -10 dB OASPL and (2) -10 PNdB, while maintaining a specified payload over a radius of 100 n.mi. and performing the mission defined in Section IV. The three parameters shown in Section IV to be most effective in changing the rotor acoustic signature are: (1) number of blades at constant solidity, (2) tip speed ($V_t$) at constant $C_T/\sigma$, and (3) hover disc loading ($W/A$) at constant $C_T/\sigma$. When these parameters are varied, the gross weight will change from that of the Model 222.

In this section, an effort is presented of designing for a specified noise level while pursuing the traditional design goals of performance, flying qualities, and structural flight envelope.

Criteria

All aircraft shall carry the reference aircraft payload (1298 pounds) for the design study mission as shown in Figure 4-1. One design shall have a PNL 10 PNdB lower than that of the Model 222 and the other one, an OASPL 10 dB less than that of the baseline aircraft. For convenience, the design shall be designated throughout the rest of the report as M222-10 PNdB and M222-10 dB OASPL. The two aircraft shall have
tail volume coefficients and overall fuselage dimensions identical to that of the Model 222. The engine and transmission sizing requirements as stated in Section IV shall be met by all aircraft. The landing gear shall be able to withstand a vertical sink speed of 14 fps. The study structural limits which remain the same as for the baseline aircraft are: (1) $V_{MO} = 350$ knots EAS, (2) $V_{dive} = 350$ knots EAS, (3) $M_{MO} = .569$, and (4) wing design maneuver load factor of 3 g's.

The wing chord shall be a function of gross weight only. The rotor diameter plus fuselage and rotor fuselage clearance will determine the wing span. The wing sizing criteria stated above is identical to that used in Section IV. The tilt-rotor has no need to maintain a specific wing chord to rotor diameter ratio as other convertible aircraft, because the wing does not contribute a significant portion of lift in low-speed flight. Thus, the wing loading may be designed for cruise only, except as limited by the placement of the rotors at the wing tips and the requirement of sufficient thickness to support the fuselage in near-hovering flight.

The engine specific fuel consumption shall be increased by 5 percent as required in MIL-C-5011A. The engine performance shall be scalable from that of the Lycoming T53-L13.

The end result of the above criteria is that the two new aircraft (M222-10 PNdB and M222-10 dB OASPL) will be similar in external appearance; however, rotors and wings as well as performance characteristics differ from the M222 model as defined in Ref. 1.
Quiet Aircraft Selection

The three rotor parameters were varied one at a time in the direction shown in Section IV to be most effective. The disc loading of the baseline was reduced to one-half its original value in two equal steps. The hover tip speed was reduced by 100 fps increments from the baseline. Then, the above matrix of rotor design was repeated for 4 and 5 blades per rotor. Thus, 27 designs were evaluated. The final selection of an aircraft meeting each acoustic criteria was made on the basis of minimum design gross weight.

The process of selecting the minimum design gross weight was facilitated by the three figures (Figures 6-1, -2 and -3) showing the variation of design gross weight as a function of hover tip speed and disc loading (W/A) for 3, 4 and 5 blades per rotor. Also, shown on these figures are the lines of constant PNL and OASPL which are 10 dB lower than that of the baseline aircraft. In Figure 6-1, however, the OASPL - 10 dB line is missing because it does not occur in the range of parameters investigated. This graphical optimization method not only shows the minimum, but also the parameter values about the minimum, thus determining the penalty involved in a nonoptimum design.

Description of New Designs

The selected aircraft have a lower disc loading, a larger number of blades per rotor, and higher design gross weight than the reference model. The aircraft with a reduced overall sound
NOTE:
1. NO. OF BLADES/ROTOR = 3
2. $C_{L/o} = 0.0896$
3. DESIGN STUDY MISSION - 100 NM RADIUS
4. MISSION PAYLOAD - 1,296 LB
5. PNLo = 92.5
6. OASPLo = 93.6

FIGURE 6-1 TILT-ROTOR DESIGN SELECTION CHART
(3 BLADES PER ROTOR)
NOTE:
1. NO. OF BLADES/ROTOR = 4
2. CT/o = .0896
3. DESIGN STUDY MISSION - 100 NM RADIUS
4. MISSION PAYLOAD = 1298 LB
5. PNLo = 92.5
6. OASPLo = 93.6

FIGURE 6-2 TILT-ROTOR DESIGN SELECTION CHART
(4 BLADES PER ROTOR)
NOTE:
1. NO. OF BLADES/ROTOR = 5
2. $C_{T/o} = .0896$
3. DESIGN STUDY MISSION - 100 N M RADIUS
4. MISSION PAYLOAD = 1298 LB
5. OASPL = 93.6

FIGURE 6-3 TILT-ROTOR DESIGN SELECTION CHART
(5-BLADES PER ROTOR)
pressure level M222-10 dB OASPL has increased by 2 db and at 5
1,000 feet pae 3 blades per rotor and a rotor speed of 150
to 170 pounds. The aircraft have a reduced weight of 11 to
level M222-10 PNdB has a new weight reduction of 1 db to 5
blades per rotor and a rotor speed of 170 to 200 pounds.
The two aircraft are depicted in photographs sketches 10-2178
and 10-2181, respectively.

A comparison of the summary weight statements for the baseline aircraft and the two new aircraft is presented in Table 6-1. The greatest relative increase in percentage of component weight of M222-10 db OASPL and M222-10 PNdB occur in the rotor (21 and 58 percent, respectively), wing (16 and 7 percent), empennage (43, 70 percent), flight controls and the drive system (15, 82.5 percent). This is offset by a decrease in engine weight (43, 4 percent), and mission fuel for the M222-10 db OASPL (13 percent).

Tables 6-2 and 6-3 show the mass properties of M222-10 db OASPL and M222-10 PNdB, respectively. In both aircraft, because of larger and heavier rotors, wing and nacelles, the roll and yaw inertias increased from the M222 mass properties shown in Table 6-1, Appendix G. In the case of M222-10 db OASPL, the nacelle horizontal roll and yaw inertias are 182 and 172 percent of the baseline aircraft. The nacelle vertical (helicopter configuration) roll and yaw inertias undergo similar increases. The much greater inertias about the two axes of the 2 quiet aircraft will result in lower acceleration
<table>
<thead>
<tr>
<th>MODEL 222</th>
<th>MODEL 222</th>
<th>MODEL 222</th>
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<tr>
<td>- 10 dB</td>
<td>- 10 PNdB</td>
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<td>718</td>
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<td>*DRIVE SYSTEM</td>
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<td>MENG. VARIATION</td>
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<td></td>
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<td>WEIGHT EMPTY</td>
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<td>(400)</td>
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<td>CREW (2)</td>
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<td>360</td>
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<tr>
<td>FUEL</td>
<td>1072</td>
<td>945</td>
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<tr>
<td>CARC. and/or</td>
<td>1298</td>
<td>1298</td>
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<td>PASSENGERS/TEAPERS</td>
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**TABLE 6-1 SUMMARY WEIGHT STATEMENT COMPARISON**

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<th>BALANCE</th>
<th>INERTIA - SLUG FT^2</th>
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<tr>
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<td>X (F.S.)</td>
<td>Y (N.L.)</td>
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<tr>
<td>FUSE &amp; CONTENTS</td>
<td></td>
<td></td>
</tr>
<tr>
<td>PWD</td>
<td>164</td>
<td>14.2</td>
</tr>
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<td>CENTER</td>
<td>2723</td>
<td>160.4</td>
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<td>AFT</td>
<td>376</td>
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<td>413.3</td>
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<td>190</td>
<td>395.3</td>
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<td>M/NELLE &amp; CONTENTS</td>
<td>2091</td>
<td>171.2</td>
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<td>OPER. WT. EMPTY</td>
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<td>181.4</td>
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<td>FUEL - INBOARD</td>
<td>946</td>
<td>171.3</td>
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<td>CABIN</td>
<td>1290</td>
<td>171.3</td>
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<td>DESIGN GROSS WEIGHT</td>
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NOTE: ROTOR BLADE INERTIAS (I_q)
BLADE WT. EA. (WITH CUFF) I_{XX} (ROLL) I_{YY} (PITCH) I_{ZZ} (YAW) SLUG FT^2

TABLE 6-2 M222-10 dB OASPL MASS PROPERTIES
<table>
<thead>
<tr>
<th>SUB-GROUPS</th>
<th>WEIGHT</th>
<th>X (F.S.)</th>
<th>Y (B.L.)</th>
<th>X (M.L.)</th>
<th>Y (ROLL)</th>
<th>Y (PITCH)</th>
<th>Y (YAW)</th>
<th>X (F.S.)</th>
<th>Y (B.L.)</th>
<th>X (M.L.)</th>
<th>Y (ROLL)</th>
<th>Y (PITCH)</th>
<th>Y (YAW)</th>
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<tr>
<td>FUSE &amp; CONTENTS</td>
<td>166</td>
<td>14.2</td>
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<td>PD</td>
<td>2949</td>
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<tr>
<td>CENTER</td>
<td>380</td>
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<td>0.6</td>
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<td>264</td>
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<tr>
<td>APT</td>
<td>257</td>
<td>430.8</td>
<td>55.0</td>
<td>15.0</td>
<td>23</td>
<td>76</td>
<td>38</td>
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<td>66</td>
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<td>HORIZ TAIL</td>
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<td>1956</td>
<td>260</td>
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<td>WING/CONTENTS</td>
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<td>140.5</td>
<td>278.6</td>
<td>44.8</td>
<td>363</td>
<td>1507</td>
<td>1730</td>
<td>178.6</td>
<td>278.6</td>
<td>79.5</td>
<td>1730</td>
<td>1507</td>
<td>363</td>
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<td>12507</td>
<td>159.8</td>
<td>29.6</td>
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<td>17131</td>
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<td>179.3</td>
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<td>124529</td>
<td>18280</td>
<td>131584</td>
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<td>OPER. WT. EMPTY</td>
<td>1181</td>
<td>171.3</td>
<td>93.0</td>
<td>40.5</td>
<td>482</td>
<td>70</td>
<td>666</td>
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<td>FULL - INBOARD</td>
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<td>171.3</td>
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<td>600</td>
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<td>315</td>
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<td>CARGO</td>
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<td>19302</td>
<td>134778</td>
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NOTE: ROTOR BLADE INERTIAS (I_0) (WITH CUFF)  I_{XX} (ROLL)  I_{YY} (PITCH)  I_{ZZ} (YAW)  SLUG FT^2

TABLE 6-3 M222-10 PNDB MASS PROPERTIES
than for the Model 222. Flying qualities have not been investigated in this study. Thus, the pilot acceptance is unknown. However, an effort was made to keep the flying qualities similar to the original model by the sizing constraint of constant tail control volumes. Nevertheless, each aircraft has different characteristics.

Performance

The performance estimates for the reference aircraft and the two designs with acoustic constraints were made in the following areas:

1. Rotor hover performance (Figure 6-4)
2. Level flight power required (Figure 6-5)
3. Mission payload – radius (Figure 6-6)
4. Mission productivity as a function of range (Figure 6-7).

The design hover Figure-of-Merit for the three aircraft are:

1. M222
   \[ FM = 0.757 \]
2. M222 - 10 dB OASPL
   \[ FM = 0.796 \]
3. M222 - 10 PNdB
   \[ FM = 0.751 \]

The above Figures of Merit correspond to the design points shown in Figure 6-4. This figure indicates that no large hover/performance penalty is encountered by a quiet design. However, the same cannot be said for cruise performance, as the aircraft propeller efficiencies decrease from the baseline values for the design condition of 300 knots, 10,000 feet Standard Day.
FIGURE 6-4 COMPARISON OF ROTOR HOVER PERFORMANCE
FIGURE 6-5 EFFECT OF AIRCRAFT DESIGN FOR NOISE REDUCTION ON POWER REQUIRED
FIGURE 6-6  PAYLOAD VS RADIUS COMPARISONS OF STANDARD AND NOISE REDUCTION DESIGNS
FIGURE 6-7 COMPARISON OF MISSION PRODUCTIVITY

MISSION RANGE, NM

V BLOCK X PAYLOAD / WEIGHT EMPTY, LB KN / LB

MODEL 222
MODEL 222 - 10 DB OASPL
MODEL 222 - 10 PNdB
The comparison of level flight power required (Figure 6-5) shows that the common performance ground rules chosen for this study result in balanced design, as all three aircraft have a .99 best range speed (cruise condition) very close to the cruise rpm transmission limit. Taking into account the differences in parasite drag, wing span loading, rotor area to wing area ratio and hover disc loading, the only unusual item in the comparison is the relative increase in power required through conversion with respect to that in hover, as the design hover tip speed is decreased (M222, $V_{th} = 750$ fps; M222 - 10 dB OASPL, $V_{th} = 680$ fps; and M22 - 10 PN dB, $V_{th} = 505$ fps). The transition power required increase occurs at advance ratios ($V/V_t$) $\approx$ .20 and a nacelle angle of $\approx$ 45 degrees. The increasing power is attributable to re-treating blade stall despite the rotor blade area increasing as the square of the reduction of design hover tip speed.

Design approaches for reducing power required through conversion are: (1) lower the $C_T/\sigma$, (2) decrease wing loading, and/or (3) incorporate sophisticated high-lift devices on the wing. The above design alternatives penalize the aircraft performance and weight. The two new aircraft have had their wing area increased over that of the baseline (Table 6-4), but not enough to permit conversion to the airplane mode at sufficiently low speeds to
<table>
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<th>ITEM</th>
<th>BASELINE</th>
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<td>No. of Blades</td>
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<td>&quot;H&quot; Speed, FFS</td>
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<td>Moment Arm, Ft.</td>
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<td>XMSN HP LIMIT/ROTOR</td>
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<td>820</td>
</tr>
<tr>
<td>fe, Ft²</td>
<td>6.279</td>
<td>7.404</td>
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<table>
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<th>HOVER OGE, 100 FT, ALT</th>
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<td>OASPL @ 500 Ft.</td>
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<td>79.4</td>
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<tr>
<td>PNL @ 500 Ft.</td>
<td>93.6</td>
<td>83.6</td>
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**TABLE 6-4**
avoid an increase in power required. It is left to a more detailed study to determine the optimum design of the wing and rotor in the conversion flight regime.

The payload vs mission radius comparison (Figure 6-6) illustrates that as a result of the common performance ground rules adopted in this study, the capability of all three aircraft does not vary widely. The same cannot be said of the comparison of mission productivity (Figure 6-7) where the productivity of the noise constrained designs is lower than that of the Model 222.

Far Field Acoustic Signatures

The two noise constrained designs have sound absorptive linings in the inlets of the engines. Thus, the rotor spectrum represents the major influence in determining either OASPL or PNL. Acoustic signature resulting from engine inlet treatment and rotor design in hover is shown in Figures 6-8 through 6-10. As a result of the design changes, the aircraft acoustic directivity pattern also changes from that of the baseline aircraft (Figure 6-11). A comparison of the relative aural detectability of the baseline and the two noise constrained designs in airplane configuration level flight is shown in Figure 6-12. The baseline aircraft (Model 222) is shown to be less detectable (shorter detection distance) at some airspeeds than the design with reduced OASPL. The aural detection distance for the tilt-rotor aircraft in this study is set by the broadband component of the rotor spectrum (typically, the 315 Hz, 1/3-octave...
FIGURE 6-8 MODEL 222 HOVER OGE SPECTRUM
FIGURE 6-9  MODEL 222 - 10 dB OASPL HOVER OGE SPECTRUM
FIGURE 6-10  MODEL 222 - 10 PNDB HOVER OGE SPECTRUM
FIGURE 6-11 HOVER PNL DIRECTIVITY COMPARISON
NOTES:
1. OBSERVER DIRECTLY UNDERNEATH FLIGHT PATH
2. LINE OF SIGHT DETECTION
3. DETECTION BASED ON THRESHOLD OF HEARING
4. ZERO AMBIENT NOISE LEVEL
5. ZERO TERRAIN ABSORPTION
6. OBSERVER AT SEA LEVEL

FIGURE 6-12 COMPARISON OF TILT-ROTOR AIRCRAFT AURAL DETECTABILITY FOR STANDARD AND NOISE REDUCTION DESIGNS
band). Broadband noise is a function of thrust, tip speed and blade area. However, there is not much difference in rotor blade areas between the baseline and the M222-10 dB OASPL. The difference in aural detectability of the new design with reduced OASPL is attributable to the larger thrust (drag) at the higher speeds shown in Figure 6-12. The equivalent flat plate drag is shown in Table 6-4 for all three designs.

Takeoff and Landing Trajectories

The trajectories performed by these designs show their off-design point performance and perceived noise level. An aircraft designed to improve the acoustic signature in hover, but acoustically unacceptable in takeoff or landing, would not represent much improvement over an aircraft with no claims of being designed with such a constraint. The three types of trajectories show the tradeoff between acoustically constrained designs and performance capability defining the trajectory and thus the distance between aircraft and observer. The three types of trajectories performed are:

1. Conventional Takeoff (nacelle tilts from vertical to horizontal while climbing)

2. Conventional Landing (nacelle tilts from horizontal to vertical while descending)

3. Helicopter-Type Takeoff (nacelle remains at an angle $\theta_n \approx 80^\circ$ such that the wing contributes no lift while climbing).
The spectra of the maximum PNL of each aircraft in both the helicopter and airplane modes, if occurring in the trajectory, are also shown. A review of Figures 6-13 through 6-27 reveals that the peaks in the PNL time histories for the two climbing trajectories of the M222 - 10 dB OASPL and M222 - 10 PNdB aircraft are lower than the baseline Model 222. However, the duration of the peak PNL is longer for the two quiet designs. The longer durations are the result of both quiet designs having less excess power than the baseline; hence, the result is a lower rate of climb. The landing trajectory information shown in Figures 6-28 through 6-36 indicates that as in the takeoffs, the two quiet designs have lower peak PNL than the baseline. However, the durations of the peak levels do not change very much from the Model 222 to M222 - 10 dB OASPL and M222 - 10 PNdB.
NOTE:
AIRCRAFT POSITION CORRESPONDING TO MAX PNL AT EACH OF THE PREDICTION LOCATION IS INDICATED BY PRIME LETTERS ON TRAJECTORY.

FIGURE 6-13 MODEL 222 STANDARD TAKEOFF AND CLIMBOUT TO 3,000-FOOT TRAJECTORY AND TIME HISTORY
FIGURE 6-14

FIGURE 6-15
FIGURE 6-16 MODEL 222 - 10 dB QASPL STANDARD TAKEOFF AND CLIMBOUT TO 3,000-FOOT TRAJECTORY AND PNL TIME HISTORY
FIGURE 6-17

FIGURE 6-18
FIGURE 6-19 MODEL 222 - 10 PNdB STANDARD TAKEOFF AND CLIMBOUT TO 3,000-FOOT TRAJECTORY AND PNL TIME HISTORY
FIGURE 6-20

FIGURE 6-21
NOTE:
AIRCRAFT POSITION CORRESPONDING TO MAX PNL AT EACH OF THE PREDICTION LOCATION IS INDICATED BY PRIME LETTER ON TRAJECTORY

FIGURE 6-22 MODEL 222 HELICOPTER TAKEOFF AND CLIMBOUT TO 3,000-FOOT TRAJECTORY AND PNL TIME HISTORY
FIGURE 6-23  MODEL 222 HELICOPTER TAKEOFF
AND CLIMBOUT TO 3,000 FEET
FIGURE 6-24 MODEL 222 - 10 dB OASPL HELICOPTER TAKEOFF AND CLIMBOUT TO 3,000-FOOT TRAJECTORY AND PNL TIME HISTORY
MAX PNL SPECTRUM IN THE HELICOPTER MODE
PNL = 101.9
TIME = 8 SEC

FIGURE 6-25 MODEL 222 - 10 dB OASPL HELICOPTER TAKEOFF AND CLIMBOUT TO 3,000 FEET
FIGURE 6-26 MODEL 222 - 10 PNdB HELICOPTER TAKEOFF AND CLIMBOUT TO 3,000-FOOT TRAJECTORY AND PNL TIME HISTORY
MAX PNL SPECTRUM IN THE HELICOPTER MODE
PNL = 106.8
TIME = 7 SEC

FIGURE 6-27 MODEL 222 - 10 PNDB HELICOPTER TAKEOFF
AND CLIMBOUT TO 3,000 FT
Figure 6-28 Model 222 - Standard Landing From 3,000-Foot Trajectory and PNL Time History
FIGURE 6-29

FIGURE 6-30
FIGURE 6-31 10 dB OASPL STANDARD LANDING FROM 3,000-FOOT TRAJECTORY AND PNL TIME HISTORY
FIGURE 6-32

FIGURE 6-33
Aircraft position corresponding to max PNL at each of the prediction location is indicated by prime letters on trajectory.

**Figure 6-34** Model 222 - 10 PNdB standard landing from 3,000-foot trajectory and PNL time history.
FIGURE 6-35

FIGURE 6-36
VII. CONCLUSIONS AND RECOMMENDATIONS

CONCLUSIONS:

A. The only design parameter change producing a significant reduction of perceived noise level at constant gross weight is lowering the hover tip speed at a constant $C_T/\sigma$. However, the following performance and weight penalties are associated with this approach.

1. A lower payload due to increases in weight empty and mission fuel.

2. Small increase in power required to hover.

3. A small decline in both hover rpm transmission limit and .99 best range speed.

   Example: A reduction in hover tip speed from 750 fps to 550 fps caused a 7.6 PNdB reduction in PNL, while

   a. Weight empty increased 8.7 percent, Mission Fuel increased 4.3 percent, and Payload declined 57.7 percent.

   b. Hover Power Required increased 2.6 percent.

   c. Hover rpm Transmission Limit and .99 Best Range Speed both declined 4.3 percent.

B. The design parameter variations most effective in decreasing the overall sound pressure level are as follows:
1. Reducing tip speed.

2. Lowering Disc Loading.

3. Increasing number of blades with constant rotor solidity.

Lowering the hover tip speed from 750 fps to 550 fps reduces the overall sound pressure level by 5.9 dB. The performance and weight penalties are the same as those identified for perceived noise level reduction.

Reducing the disc loading changes the performance and weight as follows:

a. A large decrease in power required to hover.

b. A large decline in both hover rpm transmission limit speed and .99 best range speed.

c. Very small variation of weight empty* and payload.

d. A lower mission fuel consumption.

Example: A reduction in disc loading from 11.3 to 5.6 lb/ft causes the overall sound pressure level to decrease from 92.5 to 89.1 dB, while:

*The very small variation in weight empty as a function of disc loading is the result of the common performance ground rule requiring that the engines supply sufficient power to meet the hovering condition only.
Hover power required was reduced by 26.6 percent.

Hover rpm transmission limit speed declined 19.4 percent and .99 best range speed decreased by 20.2 percent.

Weight empty increased 1 percent and payload increased 1.9 percent.

Mission fuel declined 11 percent.

Increasing the number of blades at constant rotor solidity from 3 to 4 blades results in the following:

a. 2.1 dB decrease in OASPL.

b. No noticeable change in weight empty, mission fuel and payload.

c. A small decline in power required to hover.

d. No noticeable change in .99 best range speed.

e. A small decrease in hover rpm transmission limit speed.

Example: An increase in the number of blades at constant solidity from 3 to 5 blades reduces the overall sound pressure level by 1.8 dB per blade or 3.6 dB, while:

- Hover power required declined 1.6 percent.

- Hover rpm transmission limit speed decreased 1 percent.
C. Design of tilt-rotors with significantly lower OASPL (-10dB) in hover than an unconstrained design may be accomplished with a moderate increase in design gross weight of 5.84 percent. However, the increase in design gross weight for a design constraint of a significant reduction in PNL (-10 PNdB) increases the DGW by 25.4 percent over that of the unconstrained design.

D. Mission productivity \((V_{block} \times \text{Payload}/WE)\) significantly declines from that of the reference.

E. A tradeoff exists between an acoustic design constraint in hover and maximum rate of climb capability on the resulting duration corrected annoyance (EPNL) underneath the takeoff trajectory. The same tradeoff is not applicable to landing.

F. The tradeoffs shown in Section IV and used in Section VI are dependent on the criteria used to size the engines and transmissions. If the criteria were based on a high-speed dash capability (300 knots) rather than hover, the weight penalties would be much larger than those used. It should be realized, hence, that many of the above conclusions are particular to the selected design and performance ground rules.

RECOMMENDATIONS:

A. The 57.7 percent decrease of payload at constant gross weight caused by a tip speed reduction of 200 fps is too severe a penalty to be accepted as a normal design practice. An effort to reduce this penalty through advanced drive systems and
advanced composite materials in the rotor system is necessary
if tip speed is to be the route used for reduction of far-
field acoustic signature.

B. Airfoils should be developed which have a higher L/D over a
wider range of Reynolds Number and angles of attack than
present ones. This would make possible a better compromise
between hover and cruise performance when the hover tip speed
is reduced in order to improve acoustic far-field characteristics.

C. The influence of (1) rotor blade span loading distribution,
(2) number of blades per rotor with constant total area, and
(3) airfoil characteristics, on the far-field rotor acoustic
signature is not well understood. An investigation should be
conducted to establish a consistent data base and empirical
relationships for the above design parameters.

D. Continuing theoretical and experimental research should be
carried out to improve the quantitative predictions of prop-
rotor noise. If the measure of acoustic acceptability is fre-
quency weighted, then emphasis should be placed on prediction
of the rotor acoustic signature above 250 Hz.

Acknowledgements

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Soule for rotor performance calculations, J.S. Wisniewski for
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The following material on tip speed and blade tips is supplementary to that presented in Section II. It is included for information purposes.

Tip Speed

The relationships presented in Section II on Tip Speed were found to be supported in most of the literature. However, the data reported by Stuckey and Goddard (Ref. 7) and presented in Figure A-1, show a lesser effect of tip speed on noise than those reported elsewhere. The sound pressure levels shown are for constant nominal thrust of a full-scale 3-bladed rotor. The first harmonic noise levels increase (with a few exceptions) with tip speed as expected. However, the higher 10th and 15th harmonics are essentially constant below a tip speed of 1000 fps.

Other available data show an increase in noise with tip speed even in the higher harmonics. Figure A-2, for example, shows the results of a whirl-tower test of CH-47B/C rotor blades operated at three different nominal thrusts (Ref. 5). The microphone located at a three-diameter distance registered an average 6 dB increase in first harmonic SPL for each 100 fps of tip speed increase. The data gathered by the ground microphone indicates
FIGURE A-1 EFFECT OF TIP SPEED ON ROTATIONAL NOISE HARMONICS FROM REFERENCE 7
FIGURE A-2 EFFECT OF TIP SPEEL ON ROTATIONAL NOISE HARMONICS
a somewhat lower rate of increase, and does not display the same uniformity with increasing tip speed as the aerial microphone.

The higher harmonic amplitudes generally increase with tip speed, but the trends are not as well defined as for the lower harmonics of sound pressure.

A cross-plot of Hubbard and Maglieri full-scale rotor data (Ref. 6) illustrates rotor noise in terms of tip speed and is given in Figure A-3. The overall SPL is approximately linear, increasing at 5 dB per 100 fps. This agrees well with the Vertol data of Figure A-2.

Figure A-4 illustrates the sound pressure level for several harmonics of the 13-foot diameter Model 160 propeller (-41° twist). For this configuration, the increase in sound pressure is approximately linear for all harmonics shown up to 850 fps tip speed, with a 6 dB increase for each 100 fps of tip speed between 550 and 850 fps for the lower two harmonics. Above 850 fps, the higher harmonics increase at a diverging rate.
ASD STATIC PROPELLER WHIRL RIG
13 FT DIA., 3 BLADES
-41 DEG TWIST
θ/7.5 = 12°
UNCORRECTED FOR REFLECTION
MICROPHONE LOCATION – 1.6 DIA FROM AXIS
14° DOWNSTREAM FROM PLANE OF ROT.

FIGURE A-4 EFFECT OF TIP SPEED ON HARMONIC SOUND LEVELS
Blade Tip Modifications

Blade tip modifications can have a moderate effect on far-field noise. A Boeing-Vertol investigation in 1961 on a tied-down CH-46 revealed that several modifications to the tip could be made to achieve amplitude reductions in certain frequency ranges relative to the unmodified, revolved airfoil tip. These tests were conducted on an overlapped tandem-rotor aircraft and the results shown in Figure A-5 would be expected to be less pronounced than for a single isolated rotor. The greatest reduction in noise at all frequencies was displayed by the square tip which averaged 5 dB over 6 octave bands.

A recent test reported by Pollard and Leverton (Ref. 11) also confirms that the square tip is as quiet as any tip yet tested, at least at the low tip speeds. Figure A-6 from their report shows the results for a 10-foot diameter rotor at 8° pitch. By contrast, Sikorsky data (Figure A-7), taken from Ref. 10 show a contradictory trend for trapezoidal tips when compared with that of Pollard and Leverton. However, the maximum tip speed in Figure A-5 is only 367 fps.

A double-swept tip was installed by Bell on a UH-1 and the results reported in Ref. 21. Figure A-8, reproduced from this reference, shows a very small benefit (in the 75-150 Hz band) except at an advancing tip Mach number of .8 where the difference is 6 dB.
MIC. LOC.: 50 FT FROM AIRCRAFT AT $\psi = 135^\circ$

SHADED DATA = EXPERIMENTAL TIP
UNSHADED DATA = AIRFOIL OF REVOLUTION TIP

FIGURE A-5  NOISE LEVELS OF EXPERIMENTAL TIP SHAPES ON TIEDOWN CH-46 AIRCRAFT
FIGURE A-6  TIP SHAPE COMPARISON FROM REFERENCE 11
\[ \begin{align*}
\triangle 183 \text{ RPM} / 591 \text{ FPS} \\
\bigcirc 203 \text{ RPM} / 661 \text{ FPS} \\
\times 213 \text{ RPM} / 692 \text{ FPS}
\end{align*} \]

ROTOR DIA. = 61.7 FT

\text{OVERALL VORTEX SOUND PRESSURE LEVEL} \\
\text{dB re } 2 \times 10^{-5} \text{ N/m}^2

\text{THrust in Thousands of Pounds}

\text{FIGURE A-7 COMPARISON OF SQUARE AND TRAPEZOIDAL TIPS FROM REFERENCE 10}
48 FT DIAMETER
27 IN. CHORD
100 FT ALTITUDE

7TH – 14TH NOISE HARMONIC
(75 – 150 Hz BAND)

FIGURE A-8 EFFECT OF SWEPT-TIP MAIN ROTOR BLADES ON HELICOPTER FLYOVER NOISE (REF. 21)
APPENDIX B

SAMPLE CALCULATION OF TURBO-SHAFT ENGINE INLET NOISE SUPPRESSION

The following sample calculation is included to familiarize the reader with design parameters controlling the installation of sound attenuating linings. The calculation procedure will utilize figures from Section III which are representative of the current state-of-the-art sound absorption lining design. The parameters represent the Model 222 and typical noise attenuation requirements.

\[
\text{Peak attenuation frequency (f) } = 4000 \text{ Hz}
\]

\[
\text{Distance between inside inlet surface & transmission fairing (H) } = 0.5 \text{ ft.}
\]

\[
\text{Speed of sound (c) } = 1117 \text{ ft/sec}
\]

\[
\text{Desired attenuation } = -10 \text{ dB}
\]

Therefore, the frequency parameter \( (fH/c) = \frac{4000 \times 0.5}{1117} = 1.79 \) (non-dimensional)

Then, using Figure 3-7, the resulting peak attenuation per \( L/H \) is \(-7.5 \text{dB}/(L/H) \) at \( M = 0 \).

Since the desired peak attenuation is \( 10 \text{dB} \), the \( L/H \) ratio needed is

\[
\left( \frac{7.5 \text{dB}}{L/H} \right) \left( \frac{L}{H} \right)_N = 10 \text{dB}
\]

\[
(L/H)_N = \frac{10 \text{dB}}{7.5 \text{dB}/(L/H)} = 1.333
\]
Thus, the length of inlet lining needed is

\[ L_N = (L/H) H = 1.333 \times 0.5 = 0.667 \text{ ft.} \]

This procedure ignores the modifications that might be needed to de-ice the new noise suppression inlet.
APPENDIX C

HOVER OGE PERFORMANCE PREDICTION

Hover and axial flow rotor performance presented in this report is calculated by a computer program consisting of vortex theory with empirical corrections for the wake structure. The Boeing Vertol Company uses this program as a primary design tool in the optimization of aerodynamic performance of rotors in axial flow. Confidence in the prediction capability of this method was gained by comparing test results and predictions for many rotors, propellers, and convertible proprotors. Examples of such comparisons may be seen in References 22 and 23.

In hover, the interaction of the wake vortex structure and the induced velocity at the rotor plane caused by the wake determine the wake vorticity. The calculation procedure uses momentum theory to establish the reference or normal vortex structure. A new blade loading is calculated from the reference wake vortex structure which is then updated to account for the new blade loading. The procedure is repeated until a solution is found.

Wake structure reflects the vortex law requiring vortex filaments to travel at the same velocity as the flow in the wake. The wake flow is the sum of the axial speed of the airscrew \( V \) and the three components of flow induced by the vortex filaments. Thus, the axial variation of the slipstream velocities are a prime factor.
in the vortex arrangement of the wake. Glauert's approximation (Ref. 24, pg. 367) of the downstream variation of the slipstream velocity with distance from the tip path plane was used.

This finally led to the approximate formula

\[ \frac{v_x}{v_{\infty}} = \left(\frac{1}{2}\right) \left[ 2 - (1 - AF) e^{-Nz/R} \right], \tag{C-1} \]

where the slipstream acceleration factor \( AF \) is about 0.19 and falls asymptotically to zero as \( v \) increases.

The slipstream model based on the above relationship worked quite well for low disc loadings, but gave too optimistic results for propellers as used in the tilt-wing configurations. Assuming that the general form of Equation (C-1) is correct, values of the so-called contraction rate parameter \( N \) were selected to match theoretical results against experimental data. In this way, a curve of "correct" values of \( N \) versus aircrew thrust coefficient values \( (C_T = T/\rho_d V_t^2) \) was obtained. The thus established relationship of \( N = f(C_T) \) was used in the computer program.

Combined blade element and momentum theory may be used for rapid calculation of rotor hover performance. Fairly accurate results (i.e. 5% Figure of Merit) should be attained by suitable correction factors based on the more detailed description of a hovering rotor contained in the vortex theory computer program.

Thus:

\[ C_p = C_p^{\eta_h} \sqrt{\frac{\eta}{\eta}} k_{\text{ind}} + \frac{\sigma_d \sigma}{8} \]

(C-2)

164.
where:

\[ \text{kind}_h \equiv \text{nonideal induced power correction} \]

\[ \overline{d}_o \equiv \text{average blade profile drag} \]

A reasonable average profile drag for the spanwise airfoil section distribution (Reference 1) of the Model 222 rotor is .00831 at the design operating condition. The nonideal induced power correction factor is shown to be a function of thrust coefficient \( C_T \) in Figure C-1. The dependence of the nonideal induced power correction on \( C_T \) is consistent with the explicit vortex interference theory because the wake structure of the vortex theory was defined empirically as a function of \( C_T \). The trend of the nonideal induced power correction shown in Figure C-1 is only good for the assumed hover figure-of-merit to propeller cruise efficiency tradeoff and can be represented over the range \( .006 \leq C_T \leq .088 \) by the following equation:

\[ \text{kind}_h = .8733 + 43.30C_T - 1774.9C_T^2 + 28664C_T^3 \quad \text{(C-4)} \]

If \( \text{kind}_h \), \( C_T/\sigma \) and disc loading \((W/A)\) are assumed to be constant, it can be shown that the figure-of-merit would increase as hover tip speed decreases.

\[ C_T/\sigma = .0886 \]

\[ C_T |_{W/A = \text{constant}}/V_t^2 \equiv K/V_t^2 \quad \text{(C-5)} \]

\[ \text{FM} |_{C_T/\sigma, W/A} = \frac{K^{1/2}/\sqrt{3}}{K^{1/2}\text{kind}_h/\sqrt{3} + .0116V_t} \quad \text{(C-6)} \]
Equation (C-6) contradicts the trend shown in Figure 4.3. However, $k_{indh}$ is not a constant as assumed in Equation (C-6), but has been shown to be a function of $C_T$. Therefore, Equation (C-6) is consistent with the methodology used for performance calculations and supports the rotor performance trends shown in Figure 4.3 if $k_{indh}$ varies.

FIGURE C-1. HOVER INDUCED POWER CORRECTION FOR TILT-ROTOR AIRCRAFT AND HELICOPTERS
APPENDIX D

TABULATION OF SUMMARY WEIGHT STATEMENTS
AND CONFIGURATION CHARACTERISTICS
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**TABLE D-1** LISTING OF SUMMARY WEIGHT STATEMENTS FOR ALL SENSITIVITY LINE AIRCRAFT
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<td>978.0</td>
<td>826.0</td>
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<td>NOVER O.E.E. IDEAL ALT</td>
<td>OASPL @ 500 FT</td>
<td>92.5</td>
<td>99.6</td>
<td>96.6</td>
<td>92.7</td>
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<td>88.4</td>
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<td>88.2</td>
<td>91.1</td>
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<tr>
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<td>93.8</td>
<td>90.3</td>
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</table>

*MAXIMUM STATIC HORSEPOWER @ S.L. STD DAY

**TABLE D-2 CHANGES IN AIRCRAFT CONFIGURATION CAUSED BY DESIGN PARAMETER CHANGES**
The acoustical mathematical model used in this report to predict the aircraft signature is the same as that in Reference 2, with a minor change in the loading law as shown in Figure 4-17.

The accuracy of the acoustic model in predicting the signature of a tilt-rotor has not been defined in depth due primarily to a lack of data on full-scale rotors. However, a valid comparison of prediction and experimental rotor data has been made with a CH-47B/C for a range of rotor tip speed and thrust sweeps on the Boeing Vertol experimental whirl tower. This comparison is shown in Table E-1 and E-2. These blades have less than 9° twist, however, tilt-rotor blades have 30-40 degrees. For correlation, harmonically related data is analyzed with narrow band filtering (2 Hz bandwidth) and averaged over approximately 30 rotor cycles or 3 seconds. For comparison with broadband theory, the data is analyzed with 1/3 or 1/1 octave filters and the average of 3 seconds of data is also read.

The only comparisons of prediction and measurements to be incorporated into this report are test data published in Reference 5, pages 47, 48, and 49. To accurately interpret the comparisons shown in Tables E-1 and E-2 for OASPL and PNL respectively, the following must be taken into account.
(1) Whirl Tower Data may vary by as much as 6 dB for identical test conditions.

(2) The rotor height was approximately one diameter above the ground. To be out of aerodynamic effects of the ground plane, the rotor should be 1.5 diameters, or higher, above the ground.

(3) The acoustic model assumes 1/3-octave bands with a: infinite roll-off of 1/3 octave; the rotor to be out-of-ground effect with zero forward speed.

(4) The acoustic model prediction has not been corrected for the reflections of the rotor acoustic signature from the ground. See Ref. 12, pgs. 96 - 100 for pure tone correction.

(5) The octave band filters used are 3 dB down at bandwidth limits and have a rolloff of 25 dB/octave.

(6) The prediction of the rotor fundamental blade passage frequency is not included in the OASPL or PNL presented in Tables E-1 and E-2.

The theoretical acoustic model for harmonically related noise consistently predicts the first five harmonics of the rotor signature with good accuracy (Ref. 5). The empirical theory or algorithm for broadband noise, however, does not achieve the same amount of

172.
precision and consistently underpredicts data (Ref. 5). Although a
generalized empirical correction has been made to this theory to
improve the agreement with rotor data, it cannot be verified with
any accuracy at this time due to a lack of available data for com-
parison. When compared with CH-47B rotor data from whirl tower
experiment, the acoustic model consistently underpredicts both
OASPL and PNL by an average of 6.9 dB and 6.6 PNdB, respectively.
The OASPL deviations vary from a minimum of 4.2 dB and a maximum of
11.5 dB. The PNL deviations have a minimum of 1.5 PNdB and a maximum
of 13.5 PNdB. It cannot be determined from this small data sample
whether the theoretical model is correctly predicting the change of
the acoustic signature as a function of either tip speed or thrust.
The authors caution the reader that the trends of the small number
of data points presented in Tables E-1 and E-2 are not sufficient
to be conclusive evidence of the accuracy of the acoustic model.
For statistical accuracy, a minimum of 8 data points per design
parameter variation (while maintaining other design parameters as
constants) would be desirable, including repeated measurements to
define the data variance. It is recommended that this be pursued
so as to create a data base against which any mathematic representa-
tion of rotor acoustics can be compared. This would be useful to
the whole rotary-wing industry, as the rotor acoustic far-field
signature becomes more important to operators of rotary-wing
aircraft.
### Measured and Predicted Overall Sound Pressure Levels for a CH-47B/C Rotor on a Whirl Tower

<table>
<thead>
<tr>
<th>Tip Speed ( \sim \upsilon_T )</th>
<th>Thrust ( \sim LB )</th>
<th>Calculated ( \sim dB )</th>
<th>Measured ( \sim dB )</th>
<th>( \Delta ) ( \sim dB )</th>
</tr>
</thead>
<tbody>
<tr>
<td>650</td>
<td>8,600</td>
<td>85.3</td>
<td>96.8</td>
<td>11.5</td>
</tr>
<tr>
<td></td>
<td>17,500</td>
<td>90.9</td>
<td>96.1</td>
<td>5.2</td>
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<tr>
<td></td>
<td>23,000</td>
<td>93.4</td>
<td>97.6</td>
<td>4.2</td>
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<tr>
<td>750</td>
<td>9,700</td>
<td>94.0</td>
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<td>19,000</td>
<td>103.2</td>
<td>109.5</td>
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<td>102.3</td>
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<tr>
<td></td>
<td>17,600</td>
<td>98.7</td>
<td>106.9</td>
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<td></td>
<td>26,600</td>
<td>102.5</td>
<td>107.4</td>
<td>4.9</td>
</tr>
</tbody>
</table>

**Table E-1**

Average \( \Delta dB \) \( 6.87 \)

### Measured and Predicted Perceived Noise Levels for a CH47E/C Rotor on a Whirl Tower

<table>
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<th>Tip Speed ( \sim \upsilon_T )</th>
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<th>Calculated ( \sim dB )</th>
<th>Measured ( \sim dB )</th>
<th>( \Delta ) ( \sim dB )</th>
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<tbody>
<tr>
<td>650</td>
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<td>107.0</td>
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<tr>
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<td></td>
<td>23,000</td>
<td>101.9</td>
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<td>750</td>
<td>9,700</td>
<td>97.2</td>
<td>108.3</td>
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<tr>
<td>850</td>
<td>8,300</td>
<td>98.9</td>
<td>106.9</td>
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<td></td>
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<td></td>
<td>26,600</td>
<td>108.6</td>
<td>110.1</td>
<td>1.5</td>
</tr>
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</table>

**Table E-2**

Average \( \Delta dB \) \( 6.6 \)
APPENDIX F

AIRCRAFT SENSITIVITY PREDICTION TOLERANCES

This section contains sensitivity prediction tolerance charts. To provide further insight into this assessment, an Evaluation Diagram is presented which summarizes the analytical process used to calculate the most probable predicted value. In addition, optimistic (low cumulative probability) and pessimistic (high cumulative probability) values are estimated and plotted as a probability curve to indicate the range of possible deviations.

Evaluation diagrams and prediction ranges for the following characteristics are contained in this Appendix:

1. Weight Empty    Figures F-1 and F-2
2. Hover RPM Transmission Limit Speed    F-3 and F-4
3. Mission Payload    F-5 and F-6
4. 99% Best Range Speed    F-7 and F-8
5. Overall Sound Pressure Level    F-9 and F-10
6. Perceived Noise Level    F-11 and F-12
FIGURE F-2 PREDICTION RANGE FOR WEIGHT EMPTY
DGW (10,000 FT, STD DAY)

- Airfoil Characteristics
- Parasite Drag
- Rotor Analysis
- Power Analysis Methodology
- Engine Installation Losses
- Transmission Losses
- Accessory Power
- Aircraft RHP Required
- Aircraft RHP Available
- Determination of Hover Limit Speed at DGW

**Figure F-3 Evaluation Diagram**

(Hover RPM Transmission Speed)

**Figure F-4** Prediction Range for Hover RPM XMSN Limit Speed
FIGURE P-5 EVALUATION DIAGRAM ACOUSTIC DESIGN STUDY
MISSION PAYLOAD
FIGURE F-6 PREDICTION RANGE FOR MISSION PAYLOAD
**FIGURE F-7** EVALUATION DIAGRAM
(.99 BEST RANGE SPEED)

**FIGURE F-8** PREDICTION RANGE FOR .99 BEST RANGE SPEED
EXTERNAL NOISE IN HOVER – DGW @ 500 FT DISTANCE

OVERALL SOUND PRESSURE LEVEL

FIGURE F-9 EVALUATION DIAGRAM
FIGURE F-10 PREDICTION RANGE FOR OVERALL SOUND PRESSURE LEVEL
EXTERNAL NOISE IN HOVER – DOW @ 500 FT DISTANCE

PERCEIVED NOISE LEVEL

- THRUST REQUIRED
  - THET
  - AIRFOIL
  - NUMBER OF BLADES
  - ROTOR RADIUS

- PLANEFORM
- ROTOR POWER
- ROTOR CONING ANGLE
- TIP MACH NUMBER
- BLADE SPAN LOADING
- BLADE NOISE

- ROTOR BROAD BAND NOISE
- ROTOR TONAL NOISE

- AIRCRAFT PERCEIVED NOISE LEVEL

FIGURE F-11 EVALUATION DIAGRAM
FIGURE F-12 PREDICTION RANGE FOR PERCEIVED NOISE LEVEL
APPENDIX G

WEIGHTS PREDICTION METHODOLOGY

This appendix contains the summary, development and validation of the mass properties (weight, balance and moments of inertia) for the Model 222, 26-foot diameter tilt-rotor aircraft (baseline). Preliminary estimates indicate only minor changes to the quantitative data presented in this report.

Summary and Development

The significant weights developed for the Model 222 are:

- Weight Empty: 9,230 lbs
- Operating Weight Empty: 9,630 lbs
- Design Gross Weight: 12,000 lbs
- Alternate Gross Weight: 14,400 lbs
- Airframe Weight*: 7,499 lbs

*DCPR (or AMPR)

The aircraft weight empty was determined using a combination of methods, including:

- Statistical Weight Trend Equations: 21%
- Actual Weights of Existing Aircraft and/or Components: 22%
- Vendor Information: 16%
- Calculated Weights (Layout and tail drawings): 31%
Similar Components of Existing Aircraft

(Percentages pertain to the weight empty of the aircraft.)

A summary weight statement for the aircraft is presented in Table G-1. Balance and mass moments of inertia for the configuration are included in Table G-2. The data in this table is distributed by sections of the aircraft to facilitate mass properties studies. Balance reference datums (X, Y and Z) defined in the table correspond to those used on the Mitsubishi MU-2 aircraft. Balance arms were determined by scaling the various layout drawings.

The group weights in Table G-1 consider current technology and the use of existing materials and manufacturing techniques.

Validation of Weights

The weight trends were developed around the aircraft geometry, design parameters, materials and structural criteria. A discussion of the various groups and the methods used to determine their weights follows:
## MODEL 222 TILT ROTOR AIRCRAFT - NASA RESEARCH AIRCRAFT

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<th>COMPONENTS</th>
<th>DESIGN GROSS WEIGHT</th>
<th>ALT. GROSS WEIGHT</th>
<th>AMPR WEIGHTS</th>
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### OPERATING WEIGHT EMPTY

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TABLE G-1 SUMMARY WEIGHT STATEMENT
(Weight, Balance & Moments of Inertia)
NASA RESEARCH AIRCRAFT

NOTES
71.8 IN.
L.E. MAC F.S. 144.6
ROTOR PLANE F.S. 114.1
WING & CHORD F.S. 162.5
PIVOT POINT F.S. 173.3

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<td>AFT</td>
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<tr>
<td>WING CONTENTS</td>
<td>190</td>
</tr>
<tr>
<td>NACELLE &amp; CONTENTS</td>
<td>1410</td>
</tr>
<tr>
<td>OPER. WT. EMPTY</td>
<td>4732</td>
</tr>
<tr>
<td>FUEL - INBOARD</td>
<td>9630</td>
</tr>
<tr>
<td>CARGO</td>
<td>2000</td>
</tr>
<tr>
<td>DESIGN GROSS WEIGHT</td>
<td>370</td>
</tr>
</tbody>
</table>

NOTE: ROTOR BLADE INERTIAS \(I_{s}\)
BLADE WT. EA = 107 EB (WITH CUFF)
\(I_{xx}\) (ROLL) 46, \(I_{yy}\) (PITCH) 5, \(I_{zz}\) (YAW) 50 SLUG FT.

| TABLE G-2 MASS PROPERTIES |
(1) **Wing Group**

\[ W_w = 220(K)^{0.585} \]

where

\[ W_w = \text{weight of wing (lbs)} \]

and

\[ K = \left( \frac{R_m W_x S_w}{100} \right) \left( \log \frac{b}{B} \right) \left( \sqrt[1/2]{1 + \lambda} \right) \sqrt{N} \]

\[ \log V_D \log AR \]

**LEGEND:**

- \( R_m \) = relief term = 1.0
- \( W_x \) = gross weight less tip pod = 7000
- \( S_w \) = planform area of wing = 200 (sq.ft)
- \( b \) = wing span = 33.42 (ft)
- \( B \) = maximum fuselage width = 5.6 (ft)
- \( \lambda \) = taper ratio = 1.0
- \( k_r \) = relative wing root thickness = .21
- \( N \) = ultimate load factor = 4.0
- \( V_D \) = dive velocity = 350 (kts)
- \( AR \) = aspect ratio = 5.61

The wing weight equation predicted the weight of the Model 222 tilt-rotor wing. For conventional wings, designed primarily for airloads resulting from forward flight, the term \( R_m W_x \) indicates the magnitude of the resultant wing shear and bending loads located at the semispan center of lift in forward flight.
Figure G-1 represents the results of wings analyzed in this manner. In the tilt-rotor, the wing design requirements results from vertical flight and transitional modes and the term $R_m W_x$ is reinterpreted by locating the center of lift at the thrust line of the rotor and defining $W_x$ as the aircraft gross weight less the weight of the nacelle and contents. The trend weight represents the total wing structure as defined in AN-9103D MIL-STD weight specification.

The wing weight was determined from layout drawings. Honeycomb construction torque-box was stress-checked to the available loads. The remaining wing structure ribs, fittings, leading and trailing edges, etc., were calculated from scale drawings. The calculated weights* are as follows:

<table>
<thead>
<tr>
<th>Component</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Torque Box</td>
<td>436 lbs</td>
</tr>
<tr>
<td>Nacelle Carry-Through Structure</td>
<td>50</td>
</tr>
<tr>
<td>Ribs, Doublers, Hardware</td>
<td>100</td>
</tr>
<tr>
<td>Leading and Trailing Edges</td>
<td>250</td>
</tr>
<tr>
<td>Fittings and Miscellaneous</td>
<td>50</td>
</tr>
<tr>
<td><strong>TOTAL</strong></td>
<td><strong>886 lbs</strong></td>
</tr>
</tbody>
</table>

*Stress-Checked

Wing structure weight review meetings are currently in progress for the purpose of reducing the wing weight below the predicted trend value of 800 pounds.
WING GROUP WEIGHT TREND

\[ W_W = 220 \times (K)^{0.585} \]

FIGURE G-1 WING GROUP WEIGHT TREND
(2) Tails

The weights of the horizontal and vertical tails are determined from the weight trend equations presented below.

**Horizontal Tail**

\[ W_h = 360(K)^{0.54} \]

where \( W_h \) = weight of horizontal tail (lbs)

\[ K = F_h \left( \frac{S_h}{1000} \right) \left( \frac{\log V_D}{TMA(t)} \right) \]

and \( F_h = \left( \frac{W g k b h}{10^6} \right) \left( \frac{1 + 2\lambda h}{1 + \lambda h} \right) \)

**Vertical Tail**

\[ W_v = 380(K)^{0.54} \]

where \( W_v \) = weight of vertical tail (lbs)

\[ K = \left( F_v + \frac{a(F_h)}{2(b_v)} \right) \left( \frac{S_v}{100} \right) \left( \frac{\log V_D}{TMA(t)} \right) \]

and \( F_v = \left( \frac{W g k b v}{10^6} \right) \left( \frac{1 + 2\lambda v}{1 + \lambda v} \right) \)

**Legend:**

- \( F \) = tail load parameter
- \( S \) = planform area (sq.ft.)
- \( V_D \) = dive velocity (kts)
- \( TMA \) = tail moment arm (measured from wing 1/4 chord to tail 1/4 chord) (ft)
- \( t \) = root thickness (ft)
= design gross weight (lbs)
\( k_y \) = pitch radius of gyration (ft)
\( k_z \) = yaw radius of gyration (ft)
\( b \) = tail span (ft)
\( \lambda \) = taper ratio; (chord at tip)/(chord at root)
\( a \) = height of horizontal tail attachment to vertical tail (measured from root of vertical tail)

Subscripts:

- \( h \) denotes horizontal tail
- \( v \) denotes vertical tail

The trends consider the tail loads which are a function of the gross weight, span, radius of gyration and point of load application (distance of the mean aerodynamic chord from the point of support). The "a" term in the vertical tail equation accounts for "T" tail configurations. Figures G-2 and G-3 present the aircraft used to develop the trends.

Refer to Figures G-4 and G-5 to determine the values of \( k_y \) and \( k_z \) to be placed in the structural box of the weight equations for \( F_H \) and \( F_V \), respectively. The weight trend equations over-predict the weights of both horizontal and vertical tails (Figures G-2 and G-3) of the Mitsubishi MU-2w aircraft as received from Mitsubishi Aircraft International, Inc., San Angelo, Texas.
WEIGHT OF HORIZONTAL TAIL ($W_{HT}$)

$W_{HT} = 350K^{0.54}$

1. MODEL 76  21. 727
2. APACHE  22. C-130E
3. HELIO  23. C-133A
4. AZTEC
5. AT-9
6. T37A
7. T38A
8. FH-1
9. T39A
10. S2F
11. F2H
12. F3H
13. F101
14. F105
15. XC-142A
16. F100
17. DHC-5
18. F89D
19. 440
20. C-130A

○ STANDARD TAIL WITH ELEVATOR
□ FULL FLYING TAIL
△ HORIZ. TAIL MOUNTED ON VERT. TAIL

FIGURE G-2 HORIZONTAL TAIL WEIGHT TREND
FIGURE G-4  RADIUS OF GYRATION - PITCH
**FIGURE G-5  RADIUS OF GYRATION - YAW**

**NOTES:**
1. SELECT APPROPRIATE VALUE FOR $k_z$ FROM ABOVE. INCLUDE DECIMAL ONLY ON WEIGHT INPUT SHEET.
2. MASS MOMENTS OF INERTIA = $\frac{W}{g} (k_z)^2$

**LEGEND:**
- $W$ = GROSS WEIGHT
- $g$ = MASS 32.17
- $L_f$ = LENGTH OF FUSELAGE
- $L_c$ = LENGTH OF CABIN
- $b$ = WING SPAN
- $e$ = SEE BELOW

**FORMULAS:**
- $e = \frac{b + l}{2}$ (AIRPLANES)
- $e = L_f + L_c$ (HELIICOPTERS)

**X-Axis:** LENGTH IN FEET ($e$)
**Y-Axis:** $k_z$ YAW RADIUS OF GYRATION - FEET

**MODEL 222 TREND**
Therefore, the constants 360 for the horizontal tail and 380 for the vertical tail were changed to 305 and 256, respectively. This assures that the aircraft in the Sensitivity Study and the new quiet design tail weights are all based on the same design criteria as represented by the Weight Trend Line.

(3) **Body and Alighting Gear**

The weights of the body and landing gear are actual weights of the Mitsubishi MU-2J aircraft.

(4) **Flight Controls**

The weights of the flight controls were determined from the following:

- **Cockpit Controls**
  \[ W_{CC} = 26(GW/10)^{0.41} \]
  \[ = \text{71 lbs} \]

- **Upper Controls**
  \[ W_{UC} = 0.35(W_R - W_{spin}) \]
  \[ = \text{360 lbs} \]

- **Hydraulics**
  \[ W_H = 25(W_R - W_{spin}/100)^{0.64} \]
  \[ = \text{178 lbs} \]

- **Fixed-Wing Controls**
  \[ W_{FW} = 0.012(GW) \]
  \[ = \text{144 lbs} \]

- **SAS and Mix Box**
  \[ \text{= 75 lbs} \]

- **Tilting Mechanism**
  \[ \cdot029(GW) \]
  \[ = \text{355 lbs} \]

where:

- \( GW \) = gross weight
- \( W_R \) = propeller weight
- \( W_{spin} \) = spinncr weight

Miscellaneous flight control components have been calculated and are in general agreement with the trend weights.
(5) **Engine Section**

(a) **Internal Structure**

The weights comprising the engine section were determined from layout drawings. The internal structure supporting the engine and transmissions is as follows:

<table>
<thead>
<tr>
<th>Item</th>
<th>Weight (lbs)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Internal Structure</td>
<td>200</td>
</tr>
<tr>
<td>Fairing</td>
<td>140</td>
</tr>
<tr>
<td>Fire Walls</td>
<td>40</td>
</tr>
<tr>
<td>Engine Mounts</td>
<td>15</td>
</tr>
<tr>
<td>Miscellaneous</td>
<td>5</td>
</tr>
</tbody>
</table>

**TOTAL** 400 lbs

(b) **Engines**

The engine weight was obtained from the manufacturer. The engines (2) are Lycoming Turboshaft T53-L-13B. The engine was modified by removing the speed decreaser gearing (engine gearbox). Vertol is designing its own drive system for the Model 222. The engine weight, including residual fluids (fuel and oil), is 513 pounds each.

(c) **Engine Installation**

The items comprising the engine installation package were calculated and estimated from layout drawings. The weights are as follows:

200.
Air Induction (no foreign object separator) 35 lbs
Exhaust 40
Cooling System (includes core, fan and drive unit) 60
Lubrication 20
Engine Controls 20
Starting System (cables, etc.) 25

TOTAL 260 lbs

(6) Fuel System 200 lbs

The weight of the fuel system is based on a fuel capacity of 308 gallons carried internally in the wing. A statistically-derived weight factor of .65 pounds per gallon was used to determine the fuel system weight of 200 pounds. The weight includes crash-resistant fuel bladders, pumps, valves, filters, plumbing and installation hardware.

(7) Rotor Installation 1100 lbs

The rotor installation weight was determined from detail drawings of the individual components of the rotor assembly. The details represent the rotor system designed and fabricated at Vertol for NASA under Contract NAS2-6598. A summary of the items and weights comprising the rotor installation are as follows:

Hub and Hardware (2) 300 lbs
Blade Retention 88
Spinners (2) 60
Blades (6) 625

TOTAL 1100 lbs

The rotor installation weight was also checked using the weight equation shown below. The weight of the spinners must be added to the end result to compare it to the calculated values.

\[ W_R = 14.2 \ a(K)^{0.67} \]

where

\[ W_R = \text{weight of rotor installation (lbs)} \]
\[ a = \text{propeller group adjustment factor (1.10)} \]
\[ (\text{rigid, articulated, etc.}) \]

\[ K = r^{-2} \left( \frac{HP_r}{100} \right)^{0.5} \left( \frac{V_{tt} R_{bo}}{10} \right) \left( \frac{R^{1.6}}{100 K^2 t} \right) \]

NOTE: The last term is a droop factor. It is used only if the result is greater than 1.

LEGEND:

\[ r = \text{centerline of rotation to average blade attachment point} \]
\[ HP_r = \text{horsepower (Xmen limit 1264 per prop)} \]
\[ V_{tt} = \text{design limit tip speed (750 x 1.15)} \]
\[ R = \text{prop radius} \]
\[ b = \text{number of blades per prop} \]

202.
\[ c = \text{blade chord (average)} \quad 1.57 \text{ ft} \]

\[ K_d = \text{droop constant} \]

\[ t = \text{blade thickness at 0.25 ft} \]

In the trend equation, the \((14.2)\) constant is the average for the articulated rotor system presented in Figure G-6. The \((16.0)\) constant is the estimated average line for rigid or hingeless systems based on the limited number of points shown. The "a" factor for the Model 222 is 1.10. The trend weight for the rotor is 515 pounds, plus 30 pounds for the spinner.

(8) Drive System

1107 lbs

The weight of the drive system was determined from design layout drawings. A second method of checking the weight was with the weight trend equation shown below:

\[ W_{\text{BOX}} = 150(QPUA/N^B)\]

where

\[ W_{\text{BOX}} = \text{weight of the individual gearbox} \]

and

\[ Q = \text{nondimensional weight factor for gear set or planetary stage} \]

\[ P = \text{design horsepower} \]

\[ U = \text{function of use factor} \]

\[ A = \text{gearbox support factor} \]

\[ N = \text{rpm} \]

\[ \bar{S} = \text{average Hertz factor} \]

\[ B = \text{bearing support factor} \]
FIGURE G-6  ROTOR/PROP WEIGHT TREND

LEGEND:
- ○ TANDEM ARTICULATED
- ♦ SINGLE ARTICULATED
- □ RIGID ROTOR
- ▲ TILT WING

\[ W_R = 16 (K)^{0.67} \]

\[ W_R = 14.2(k)^{0.67} \]
The trend permits a box-by-box building block approach to determine the drive system weight. It allows for actual design considerations to be used in predicting the weight of the individual gearboxes. The trend includes the weights of the gears, bearings, seals, spacers, case, etc. The weight of the lubrication system and interconnect cross-shafting is not included in the trend values; these must be added separately. Figure G-7 presents a plot of the actual weights of some existing aircraft gearboxes. The trend weights are presented below along with the weights of the various boxes, lubrication system and shafting determined from calculating layout drawings.

<table>
<thead>
<tr>
<th></th>
<th>Calculated Weight</th>
<th>Trend Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Engine Box</td>
<td>174</td>
<td>150</td>
</tr>
<tr>
<td>Rotor Box (includes accessory drive)</td>
<td>624</td>
<td>589</td>
</tr>
<tr>
<td>Bevel Box</td>
<td>65</td>
<td>90</td>
</tr>
<tr>
<td>Cross-Shaft</td>
<td>100</td>
<td>100</td>
</tr>
<tr>
<td>Miscellaneous Shafting</td>
<td>26</td>
<td>26</td>
</tr>
<tr>
<td>Lubrication</td>
<td>18</td>
<td>112</td>
</tr>
<tr>
<td><strong>TOTALS</strong></td>
<td><strong>1107</strong></td>
<td><strong>1067 lbs</strong></td>
</tr>
</tbody>
</table>

(9) **Fixed Equipment**

The fixed-equipment group includes the items beginning with the auxiliary powerplant and ending with the auxiliary gear group on the Summary Weight Statement, Table G-1. The weights were
determined from equipment lists developed around the tilt-rotor research aircraft requirements. A summary of the items and the weights of the individual groups are tabulated below:

(a) Instruments 108 lbs

<table>
<thead>
<tr>
<th>Item</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flight</td>
<td>50</td>
</tr>
<tr>
<td>Engine</td>
<td>25</td>
</tr>
<tr>
<td>Drive/Rotor</td>
<td>26</td>
</tr>
<tr>
<td>Hydraulics</td>
<td>7</td>
</tr>
<tr>
<td><strong>TOTAL</strong></td>
<td><strong>108 lbs</strong></td>
</tr>
</tbody>
</table>

(b) Electrical Group 305 lbs

<table>
<thead>
<tr>
<th>Item</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power Supply (starter/generator, batteries)</td>
<td>133</td>
</tr>
<tr>
<td>Power Conversion</td>
<td>46</td>
</tr>
<tr>
<td>Power Distribution (controls. circuit breakers, junction boxes, connectors, wiring, supports, etc.)</td>
<td>106</td>
</tr>
<tr>
<td>Lights (interior, exterior, landing, taxi, etc.)</td>
<td>20</td>
</tr>
<tr>
<td><strong>TOTAL</strong></td>
<td><strong>305 lbs</strong></td>
</tr>
</tbody>
</table>

(c) Electronics 230 lbs

<table>
<thead>
<tr>
<th>Item</th>
<th>Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>AN/ARC-51A Radio (UHF)</td>
<td>36</td>
</tr>
<tr>
<td>AN/ARC-115 Radio</td>
<td>6</td>
</tr>
<tr>
<td>AN/ARN-52 Radio (TACAN)</td>
<td>47</td>
</tr>
<tr>
<td>AN/ALC-14 Interphone</td>
<td>19</td>
</tr>
<tr>
<td><strong>TOTAL</strong></td>
<td><strong>230 lbs</strong></td>
</tr>
</tbody>
</table>
AN/ASN-73 Attitude and Heading Reference System 49
AN/APN-171(V) Electronic Altimeter Set 20
Shelves, Wiring & Supports 53

TOTAL 230 lbs

(d) Furnishings and Equipment 439 lbs

Accommodations for Personnel: 299
(pilots ejection seats (2)
(No. American Aviation LW-3B), seat rails, relief tubes, litter supports)

Miscellaneous Equipment: 63
(data cases, windshield wiper/washer, instrument boards, consoles)

Furnishings: 35
(floor covering, trim, soundproofing)

Emergency Equipment: 42
(fire detection and extinguishing equipment, portable fire ext., first-aid kit)

TOTAL 439 lbs

(e) Airconditioning: 108 lbs

Environmental Control Unit, Fan, Plumbing, Ducting, Supports and Hardware
(f) Auxiliary Gear 10 lbs

Fittings and Supports for Tiedowns (jacking, leveling, hoisting, etc.)

(g) Useful Load 2770 lbs

The useful load for the 12,000-pound DGW configuration includes:

(a) Pilots (2) - 180 lbs each 360
(b) Trapped Liquid & Engine Oil 40
(c) Mission Fuel for 100 n.mi. radius 1072
(d) Mission Payload 1298

TOTAL 2770 lbs
LIST OF REFERENCES


FOLDOUT FRAME

[Diagram of a vehicle or aircraft with a 527" dimension and a 46° angle indicated]
WING
ASPECT RATIO: 6.57
AREA: 308.3 SQ FT
SPAN: 45.0 FT
GEOM. MEAN CHORD: 8.9 FT
TAPER RATIO: 1.0
ROUT THICKNESS: 0.210
TIP THICKNESS: 0.210
WING LOADING: 48.9 LB/SQ FT

HOR. TAIL
ASPECT RATIO: 4.61
AREA: 103.0 SQ FT
SPAN: 21.8 FT
MEAN CHORD: 4.7 FT
THICKNESS/CHORD: 0.1

VERT. TAIL
ASPECT RATIO: 1.77

ASPECT RATIO: 1.77
AREA: 89.9 SQ FT
SPAN: 12.8 FT
MEAN CHORD: 7.1 FT
THICKNESS/CHORD: 0.090

PROPELLER
DIAMETER: 37.6 FT
SOLIDITY: 0.183
DISC LOADING: 6.8 LB/SQ FT
THRUST COEFF/SOLIDITY: 0.076
BLADES/ROTOR: 4.0

WEIGHTS
WEIGHT EMPTY: 12,167
GROSS WEIGHT: 18,046

ENGINE INLETS AND GEAR BOX FAIRINGS
TREATED WITH SOUND ATTENUATION MATERIAL

PIVOT Q
STA 177.80

PIVOT AXIS