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LEVEL II

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STUDY OF COST/ BENEFIT TRADEOFFS AVAILABLE IN HELICOPTER NOISE TECHNOLOGY APPLICATIONS

R.H. Spencer
H. Sternfeld, Jr.

The Boeing Vertol Company
Philadelphia, PA

JANUARY 1980
FINAL REPORT

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NOTICE

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16. Abstract

This study investigated cost/benefit tradeoffs using the case histories of four helicopters for which design and development were complete, and in three cases, have undergone substantial flight testing. The approach to quieting each helicopter was an incremental reduction of each source as required to obtain reductions in flyover noise with modifications to other secondary systems only as necessary. The methodology used to predict the effects of the design modifications on acquisition, maintenance, and operating costs were typical of those employed by rotorcraft manufacturers. The reduction of helicopter flyover noise generally was achieved through reductions in rotor tip speed. Performance characteristics were maintained to specified minimums for each aircraft in the study.

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METRIC CONVERSION FACTORS

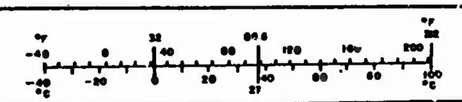
Approximate Conversions to Metric Measures

Symbol	When You Know	Multiply by	To Find	Symbol
		LENGTH		
in	inches	*2.5	centimeters	cm
ft	feet	30	centimeters	cm
yd	yards	0.9	meters	m
mi	miles	1.6	kilometers	km
		AREA		
in^2	square inches	6.5	square centimeters	cm^2
ft^2	square feet	0.09	square meters	m^2
yd^2	square yards	0.8	square meters	m^2
mi^2	square miles	2.6	square kilometers	km^2
	acres	0.4	hectares	ha
		MASS (weight)		
oz	ounces	28	grams	g
lb	pounds	0.45	kilograms	kg
	short tons (2000 lb)	0.9	tonnes	t
		VOLUME		
tsp	teaspoons	5	milliliters	ml
Tbsp	tablespoons	15	milliliters	ml
fl oz	fluid ounces	30	milliliters	ml
c	cups	0.24	liters	l
pt	pints	0.47	liters	l
qt	quarts	0.95	liters	l
gal	gallons	3.8	liters	l
ft^3	cubic feet	0.03	cubic meters	m^3
yd^3	cubic yards	0.76	cubic meters	m^3
		TEMPERATURE (exact)		
°F	Fahrenheit temperature	5/9 (after subtracting 32)	Celsius temperature	°C

*1 in = 2.54 (exactly). For other exact conversions and more detailed tables, see NBS Misc. Publ. 286, Units of Weights and Measures, Price $2.25, SD Catalog No. C13.10:286.

Approximate Conversions from Metric Measures

Symbol	When You Know	Multiply by	To Find	Symbol
		LENGTH		
mm	millimeters	0.04	inches	in
cm	centimeters	0.4	inches	in
m	meters	3.3	feet	ft
m	meters	1.1	yards	yd
km	kilometers	0.6	miles	mi
		AREA		
cm^2	square centimeters	0.16	square inches	in^2
m^2	square meters	1.2	square yards	yd^2
km^2	square kilometers	0.4	square miles	mi^2
ha	hectares (10,000 m^2)	2.5	acres	
		MASS (weight)		
g	grams	0.035	ounces	oz
kg	kilograms	2.2	pounds	lb
t	tonnes (1000 kg)	1.1	short tons	
		VOLUME		
ml	milliliters	0.03	fluid ounces	fl oz
l	liters	2.1	pints	pt
l	liters	1.06	quarts	qt
l	liters	0.26	gallons	gal
m^3	cubic meters	35	cubic feet	ft^3
m^3	cubic meters	1.3	cubic yards	yd^3
		TEMPERATURE (exact)		
°C	Celsius temperature	9/5 (then add 32)	Fahrenheit temperature	°F



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ILLUSTRATIONS

ILLUSTRATIONS (continued)

ILLUSTRATIONS (continued)

TABLES

SYMBOLS

A_b	blade area, ft^2
B	Number of blades
c	blade chord, ft
$\bar{c}_{do}$	profile drag coefficient
c_o	speed of sound in air, ft/sec
c_{λ_C}	radial force harmonic coefficient
c_{λ_D}	drag force harmonic coefficient
c_{λ_T}	thrust force harmonic coefficient
C	lift coefficient
C_n	amplitude of nth sound harmonic at specified field point, lb/ft^2
D/L	disk load, lb/ft^2
E	number of interactions per revolution
EPNL	Effective Perceived Noise Level, EPNdB
f_p	peak frequency, Hz
ft_o	$S \cdot E \cdot /\!\!\!\!\ell_w$, nondimensional parameter
j	summation index
J_i'	complex collection of Bessel function of argument $nM\cos\Theta$
J_n	Bessel function of order n
k	$c/2R_t$, slenderness ratio
k	loading power law exponent
K	constant, rotational noise equation, lb
K_T	thrust constant, lb/ft^2
$\Delta L/L_o$	fractional steady load change per blade

SYMBOLS (continued)

m	rotational harmonic number
M	rotational Mach number
M_{dd}	drag divergence Mach number
M_e	effective Mach number
M_f	forward flight Mach number
M_t	rotational tip Mach number
n	mB, harmonic number x number of blades
N	number of blades
N	rotor speed, RPM
p_m	sound pressure in harmonic mB, lb/ft^2
PNL	Perceived Noise Level, PNdB
PNLT	tone-corrected Perceived Noise Level, PNdB
r	distance from rotor center to field point, ft
R	blade radius, ft
RHP	rotor horsepower, HP
s	blade loading harmonic number
$S_{1/3}$	one-third octave band correction (Fig A-1), dB
SPL	sound pressure level - dB re 2×10^{-5} N/m^2
t	blade thickness
T	rotor thrust, lb
V_t	rotor tip speed, ft/sec
β_j	Fourier coefficients in blade torque loading
θ	angle between disk plane and field point, deg.

SYMBOLS (continued)

λ	air loading harmonic number
ξ	$\frac{R_{TIP}}{R}$
ρ	mass density of air, lb-sec^2/ft^4
ρ_w	load solidity (fraction of the effective disk annulus occupied by the unsteady loading region)
σ	rotor solidity
χ_s	blade loading spectrum function
ψ_o	blade azimuth at intersection, deg.
$\Delta\psi$	incremental azimuth angle where blade section M 0.8, deg.
Ω	rotor rotational speed, rev/sec

I. SUMMARY

This study investigated the effect which exterior noise standards will have on helicopter design and costs. The study investigates case histories of the four helicopters shown in Figure 1, for which design and development were complete, and in three cases, have undergone substantial flight testing. The developmental background, therefore, has been well documented on each aircraft. The approach to quieting each helicopter was an incremental reduction of each source as required to obtain reductions in flyover noise with modifications to other secondary systems only as necessary. The methodology used to predict the effects of the design modifications on acquisition, maintenance, and operating costs were typical of those employed by rotorcraft manufacturers.

The reduction of helicopter flyover noise generally was achieved through reductions in rotor tip speed, and for single rotor aircraft this sometimes was accompanied by modification of the tail rotor placement. Secondary reduction in noise was derived by system modification or acoustic treatment of other aircraft systems (engine inlets, rotor transmissions, advanced airfoils, etc.).

Performance characteristics were maintained to specified minimums for each aircraft in the study. Rotor speed reduction, for example, generally was accompanied by an increase in rotor solidity and strengthened drive train components. Where performance capability was initially substantially above specified requirements, margins over these minimums were reduced without modifying the system.

The major findings of the study are:

> The acquisition and operating costs of new aircraft are substantially less affected by modifications than helicopters already in production.
>
> The impact of reducing flyover noise on helicopter acquisition and operational costs is strongly influenced by the production quantity over which the modification costs are spread.
>
> Each helicopter must be studied as an individual case and generalization of cost trends of noise reduction should be avoided.

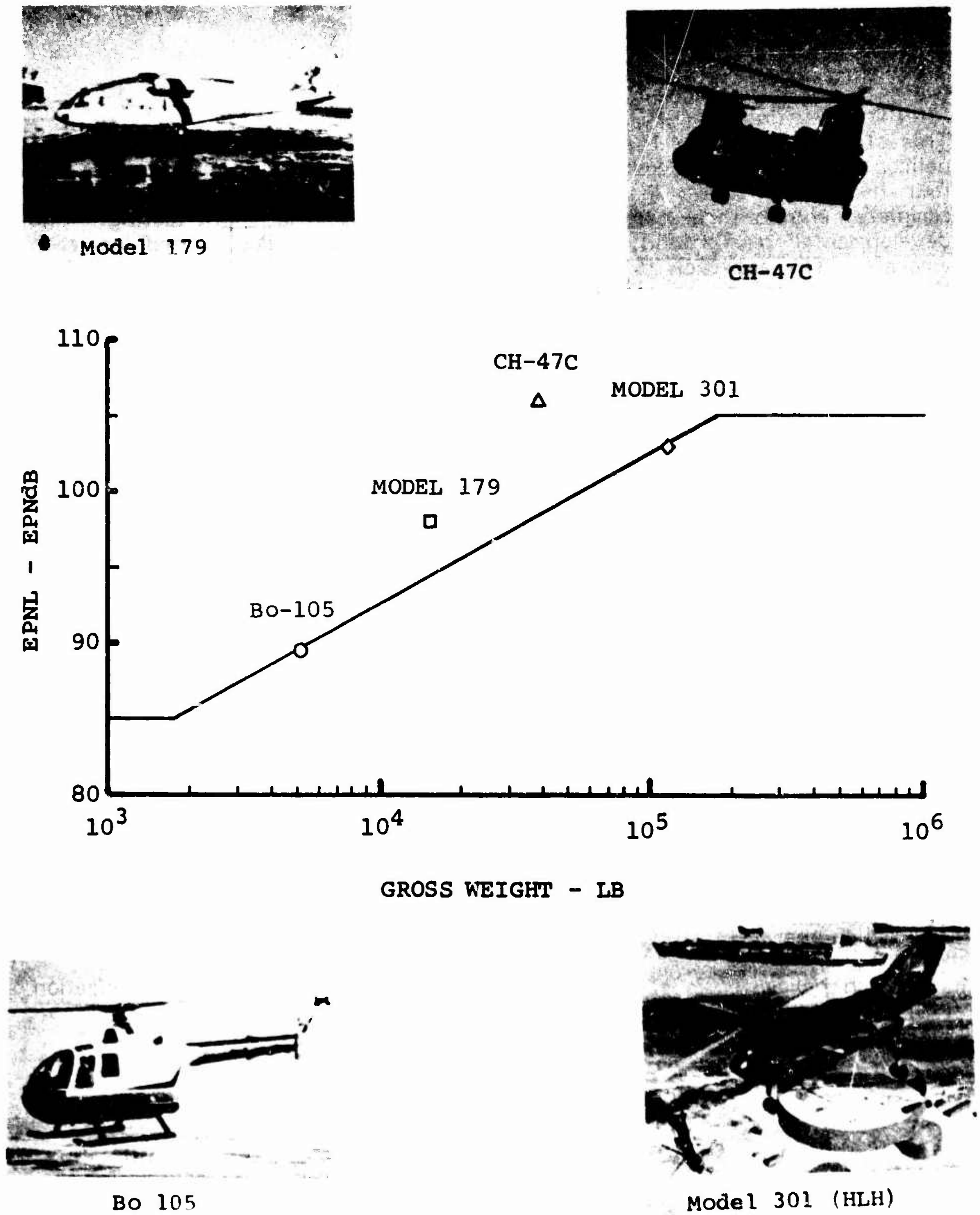


Figure 1. Comparison of Aircraft Noise with Proposed Flyover EPNL Criteria.

II. INTRODUCTION

The past few decades have seen the helicopter industry grow steadily from its infancy primarily as a supplier of military vehicles in the 1940's and 1950's to the present, where the inventory of civil helicopters in the United States and Canada exceeds 8000 vehicles. (1) The helicopter has found widespread use in such diverse applications as emergency transport, logging, executive and business transport, firefighting, heavy equipment installation, transmission line installation, traffic reporting and newscasting. The number of active heliports has increased by an order of magnitude to currently more than 3400 sites. (1) The next decade is expected to see an unprecedented growth in the civil helicopter industry and estimates of one market forecaster (2) indicate that the commercial helicopter inventory of the free world by 1987 will increase 154 percent over the previous ten years to over 25,000 units.

In response to the 1968 amendment to the Federal Aviation Act, the FAA is proposing noise standards for helicopter certification. These rules are intended to assure the orderly control of community noise due to helicopters, commensurate with "economic reasonableness and technical practicability". This study investigates the impact which these noise rules will impose on rotorcraft cost.

In the earlier stages of development, virtually all helicopters were designed for military usage. Some of these subsequently were FAA certified and used in civil applications. Due to the growth of the civil demand for small to mid-sized helicopters, manufacturers have found it worthwhile to design helicopters specifically for commercial applications. Larger sized helicopters, however, are both in less demand and require substantially more capital to develop. It therefore appears that these aircraft will continue to be derived from military models for the foreseeable future. Although it may be possible to design helicopters with acceptably low noise signatures, it is often not possible to do so while also complying with some of the performance requirements which are imposed on military models. Therefore, it can be anticipated that larger civil helicopters will have to be derived by making modifications to their military parents.

The approach taken for this program was to utilize the noise reduction analyses and cost estimating procedures used by a major helicopter manufacturer, using both the methodology and personnel who would normally perform these functions. The helicopters which were selected for study were all ones which had been designed primarily to performance, flying qualities, and strength criteria, with noise as a secondary consideration. In this manner, the 'real world' constraints, which often limit the amount of noise reduction which can actually be achieved, and the changes required to secondary systems, because

(1) "Aerospace Facts and Figures - 1979/1980" published by Aerospace Industries Association of America, Inc., Washington, D.C., July 1979.

(2) Defense Marketing Systems, Inc., "World Helicopter Forecast to 1987", Published by DMS, Inc., 100 Northfield St. Greenwich, Conn. 06830.

because of changes to the noise generating components, will be encountered. It is the above considerations, which are not encountered with 'paper design' aircraft and are usually only recognized by the manufacturer of an actual aircraft which often have major impact on the cost of accommodating noise reduction design changes.

III. METHODOLOGY

General Approach

Figure 2 presents an overview of the steps which were taken to arrive at reduced noise configurations and the associated acquisition cost and direct operating cost for each helicopter. More detailed descriptions of the individual procedures used are contained in the remainder of this section.

Available acoustical data for each baseline helicopter were analyzed (1) and used in conjunction with analytical predictions (2) to identify the individual component noise sources and spectra (3). This information was then used to identify those components of the acoustical signature whose reduction would be required in order to reduce the Effective Perceived Noise Level (EPNL) of the aircraft (4). The required design changes were determined (5) and reviewed for adherence to good design practice (6). The changes in weight (7) and performance (8) were also predicted.

Estimates of the nonrecurring man-hours required for design, design support, and testing were made (9). This information, along with the estimated weight changes (7), become input for determining the production costs of the completed aircraft (10). This information then was utilized as input for computing direct operating costs (11) for the baseline and alternate designs.

Acoustics

Three major sources generally contribute to the exterior acoustical signature of helicopters: main and tail rotors and the powerplants. For some helicopters, dynamic system noise (transmissions, shafting, etc.) and airframe noise also may be significant.

Each of these individual sources creates noise by several acoustical mechanisms which are not all equally influential in determining the flyover noise of the helicopter. Figure 3 illustrates key mechanisms which contribute to the total noise signature of the helicopter. The contribution of each of these components to flyover noise has been determined from measured data and identified from source frequencies based on the prediction methodology described in following sections.

Noise sources on each of the baseline aircraft were determined by analysis of measured data as well as predictions for each of the source components. The flow chart of Figure 4 illustrates the procedure. The magnitude and frequency of each source were identified in 1/3 octave spectra and sound levels were converted to NOY values to determine the contribution of each source to the Perceived Noise Level (PNL) at selected instants in time during flyover. The largest magnitude NOY values were identified and reduced selectively to obtain incremental reductions in PNL. Appropriate configuration changes were developed to achieve the desired noise reduction for each source. In most instances, trends available from test data were used to determine the magnitude change of the operating variable, such as rotor speed effects on flyover noise.

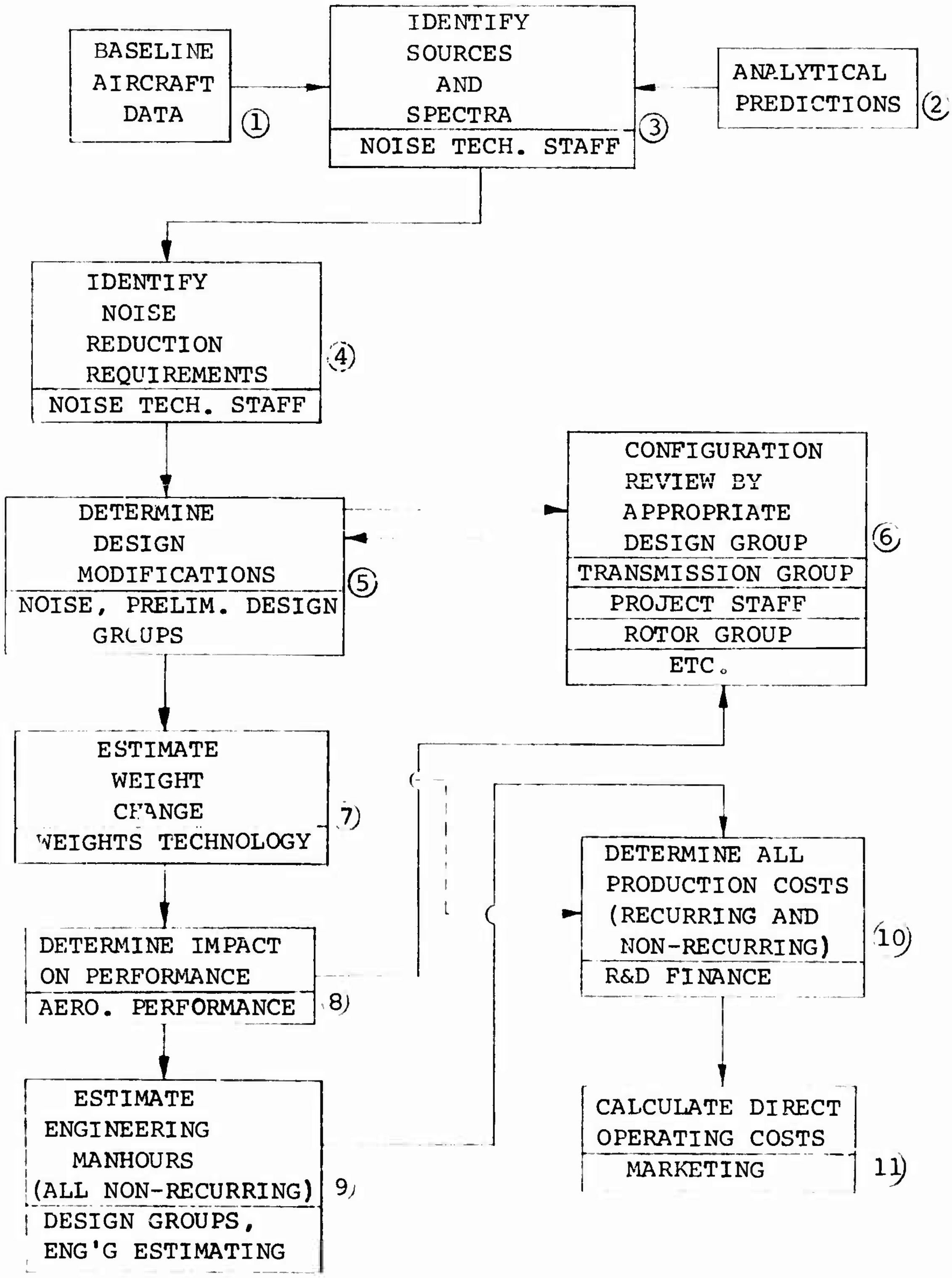


Figure 2. Program Flowchart

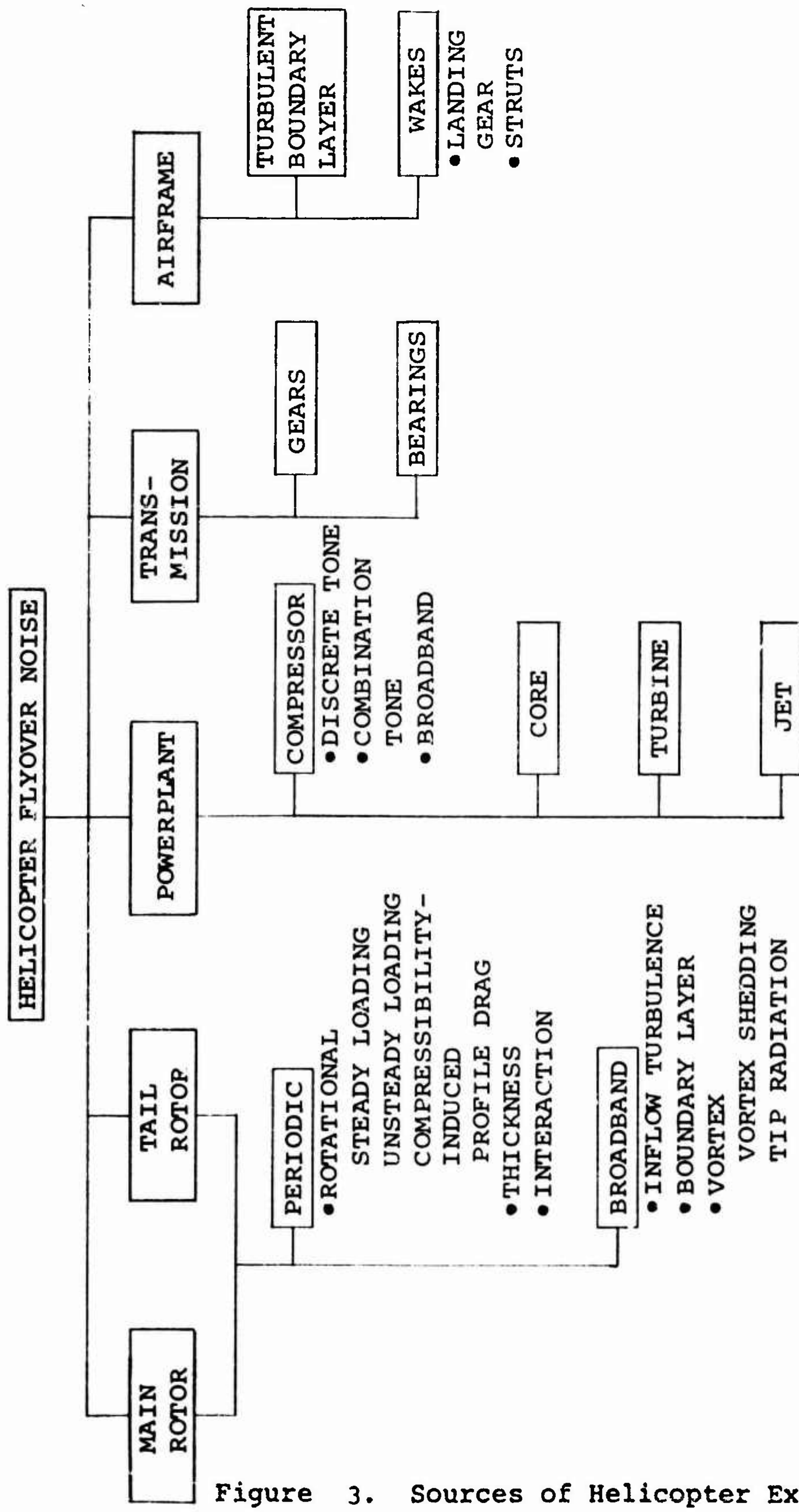
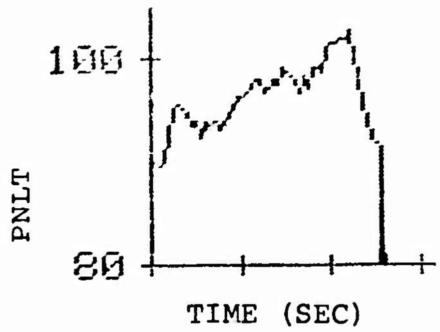
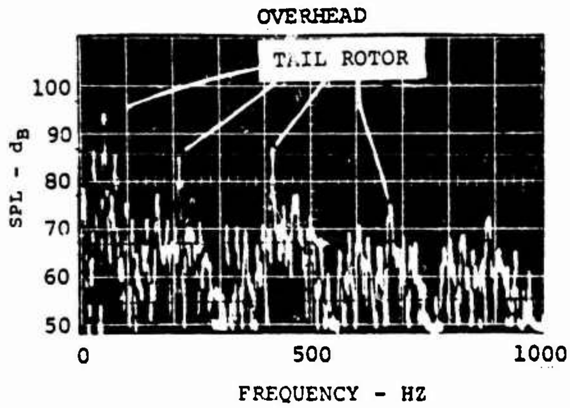


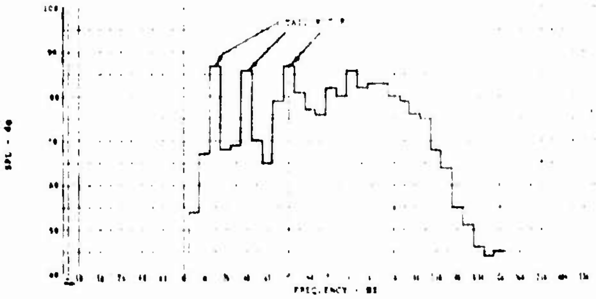
Figure 3. Sources of Helicopter Exterior Noise



PNLT TIME HISTORY



NARROW BAND SPECTRA



ONE-THIRD OCTAVE BAND SPECTRA

NOY VALUES

Figure 4. Identification of Critical Sources

Where flyover noise measurements were available (CH-47C, Model 179, BO-105), a determination of EPNL was made using recorded data input to an analog-to-digital converter interfacing a mini-computer. From this analysis, the magnitude of the flyover EPNL was determined and an assessment of the status of each aircraft with regard to proposed exterior noise standards was made. The general requirement for noise reduction of each aircraft type was defined and the procedure of noise source identification initiated.

The process for identifying noise sources on each aircraft consisted of comparing an analysis of flyover noise both in 1/3 octave bands and narrowband spectra with predicted signatures for each aircraft flyover in order to identify source components. The initial step comprised a determination of the instant of maximum amplitude PNL on the magnetic tapes as an approximation of maximum perceived noise level during flyover. The tapes were then marked at two-second intervals from this PNL-Max point for both approach and departure.

The analyzer used to obtain 1/3 octave band spectra was a General Radio 1921 Realtime Digital Analyzer. A 1/8 second integration was employed to identify flyover spectra at each interval. The 1/8 second integration time was utilized for the source identification process since vehicle angular position changes rapidly particularly near the overhead locations (determination of flyover EPNL was performed using the normally specified interval of 1/2 second). The individual spectra were converted to equivalent NOY values, and these flyover spectra are illustrated for each aircraft in Section V, Aircraft Cost/Benefit Trade Investigation.

Narrowband spectra were developed by a similar procedure using a Federal Scientific (Nicolet) Model 500 Ubiquitous Analyzer. The narrowband spectra were used primarily in identifying sources such as main and tail rotors as well as engine tones that are not discernable using 1/3 octave or wider filters which are also shown in Section V.

Identification of helicopter noise sources from data for separating discrete frequency from broadband sources is of little assistance in separating the several sources of periodic rotor noise listed in Figure 3, and analytical prediction must be used. For example, Figure 5A shows that for the CH-47C at an airspeed of 141 knots, the measured data at frequencies below 250 Hz are due to blade-vortex interaction, while Figure 5B indicates that at 157 knots the thickness noise has increased to values greater than that due to blade-vortex interaction.

The rotor prediction methods used in this study are described in the Appendix, but have been reported in greater detail by Pegg(3) who has adapted several procedures into a convenient format for estimating flyover

(3) Pegg, R. J., "A Summary and Evaluation of Semi-Empirical Methods for the Prediction of Helicopter Rotor Noise," NASA TM 80200, December 1979.

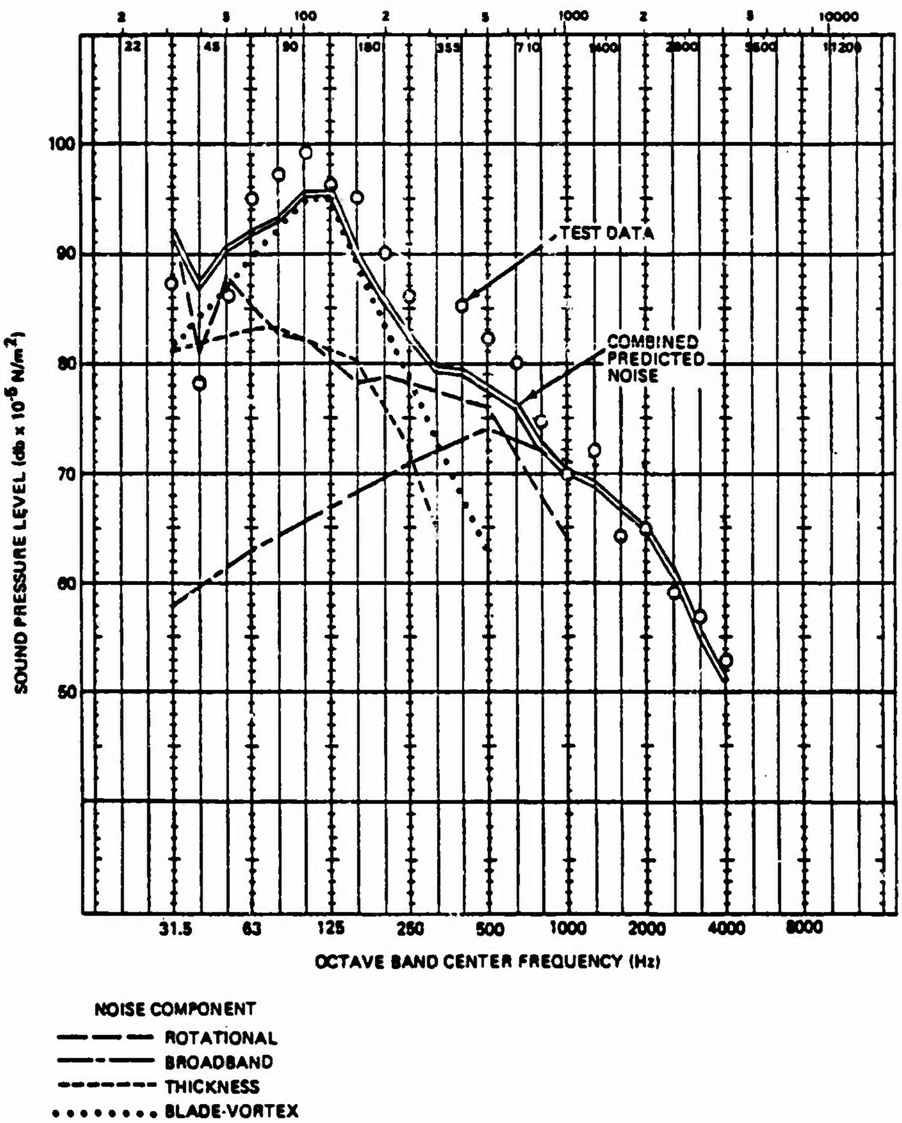


Figure 5a. Comparison of Measured and Predicted CH-47C Rotor Noise - 141 KT (Sheet 1)

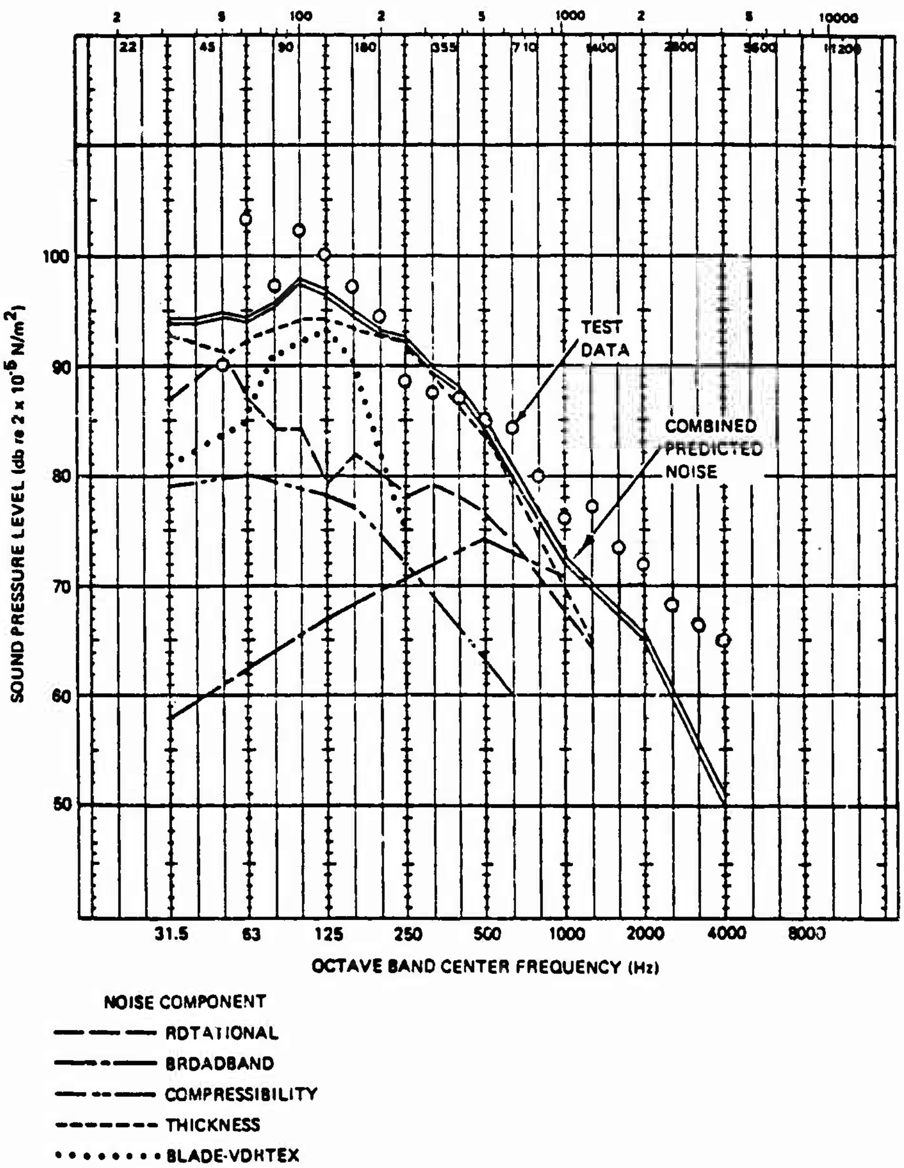


Figure 5b. Comparison of Measured and Predicted CH-47C Rotor Noise - 157 KT (Sheet 2)

noise. For engine noise, the procedures used were as published in three NASA reports treating fans and compressors, (4) core noise, (5) and jet noise (6).

Aerodynamics

The aerodynamic/performance methods used to compute the performance data shown in this report for the BO-105, CH-47, Commercial UH-61A (Model 179) and Heavy Lift Helicopter are described in this section. In general, the information presented is based on power required data measured in flight test and corrected for configuration differences. In areas where flight test data were lacking or insufficient, trim and power analyses, which were correlated with the available test data, were used to extend the data base.

Engine power-available data were based on manufacturers uninstalled power and fuel flow corrected for test-measured installation effects. Transmission power losses were based on test derived values.

Subsequent paragraphs describe the basis for the performance data presented elsewhere in this report for each of the aircraft (and their derivatives) under study. The hover performance computer program uses an Explicit Vortex Influence technique. This technique uses a prescribed wake approach which is basically an extension of fixed wing lifting line theory where each blade is represented by a lifting line and trailing vortex wake. This wake is composed of an infinite number of weak vortex filaments which the theory mathematically approximates by a finite number of vortices streaming from various radial locations. The positioning of the vortices below the rotor is dictated by a semi-empirical prescribed rate of wake contraction since the vortex filaments must travel at the velocity of the surrounding fluid. The contraction rate, specified as a function of the thrust coefficient $C_T = T/\rho\pi R^2 V_t^2$, is determined by analytical studies of finite-core vortex ring flows and by correlation of calculated and measured propeller and rotor static performance.

The strength of the vortices is determined by the section lift (C_ℓ) distribution using the Kutta-Joukowski theorem. The angle-of-attack and, hence, the C_ℓ distribution is determined by the downwash velocity induced by the vortices defined by the Biot-Savart law. An iterative technique is used to obtain a mutually consistent C_ℓ and downwash distribution. Once an agreement is achieved, the C_ℓ and section drag (C_d) distributions are integrated taking into consideration the local downwash angle, thus thrust and power required are obtained. If the computed thrust and the desired thrust do not agree, the collective pitch angle setting is changed and the entire process is repeated.

(4) Heidmann, M. F., "Interim Prediction Method for Fan and Compressor Source Noise", NASA TM X-71763, NASA Lewis Research Center, June 1975.

(5) Huff, R. G., Clark, B. J., and Dorsch, R. G., "Interim Prediction Method for Low Frequency Core Engine Noise", NASA TM X-71627, NASA Lewis Research Center, November 1974.

(6) Stone, J. R., "Interim Prediction Method for Jet Noise", NASA TM X-71618, NASA Lewis Research Center.

The iterative calculations described above require the use of a high-speed computer. The inputs required are:

- o Airfoil section C_ℓ and C_d characteristics
- o Rotor Geometry
- o Ambient condition
- o Required thrust

Figure 6 shows the excellent correlation between the theory and CH-47C flight test data in hover.

Aircraft trim for forward flight conditions is determined by solving the six steady-state equations of motion developed from a force and moment balance about the center-of-gravity. The computer program was formulated in such a way that the flight conditions, gross weight, speed of flight (horizontal), sideslip angle and aircraft geometry are input.

Iterative solution techniques are required because of the complexity of the rotor analysis needed to compute the rotor forces and moments. The rotor analysis is a subroutine in itself and uses a numerical approach for solving the rotor flapping and force equations. Blade stall, reverse flow, and compressibility effects are taken into account by the use of two-dimensional airfoil section data. However, in order to simplify this analysis, the following assumptions were made:

1. Induced velocity distribution is assumed to be uniform.

2. Blade lag and all elastic degrees of freedom are neglected.

3. Unsteady aerodynamic and spanwise flow effects are ignored.

4. Three-dimensional compressibility effects at the blade tip are not considered.

Once the trim has been established, corrections to the power required for nonuniform downwash and parasite powers are added to the basic trim power required predictions. Figure 7 shows the good correlation between the power computed by the trim analysis and CH-47 flight test for various weights and velocities.

Weights

The parametric relationships from Boeing Vertol's Semi Empirical Weight Trend expressions (see Figures 8, 9 and 10) were used to assess the weight effects of changing rotor speed, blade design and number of blades. Weight changes due to redesign of other parts of the aircraft were estimated using standard procedures.

Costs

In the past, civil helicopters were mainly derived from aircraft that already had accumulated substantial flight experience that typically stemmed from an extensive military background. In this study these are referred to as 'in-production' aircraft. More recently there has been a growing trend for

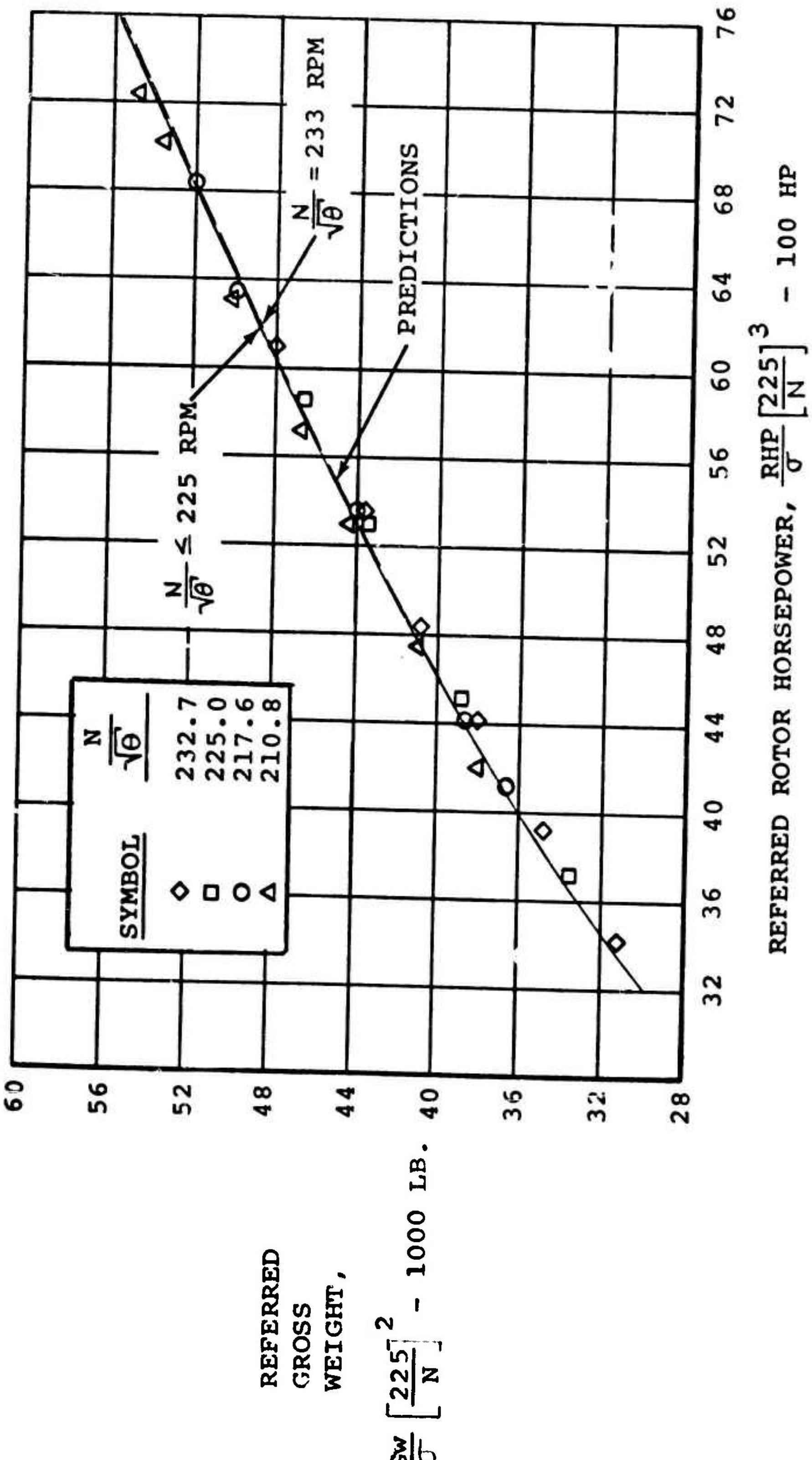


Figure 6. Hover Performance Prediction - Correlation of Test Data with Prediction

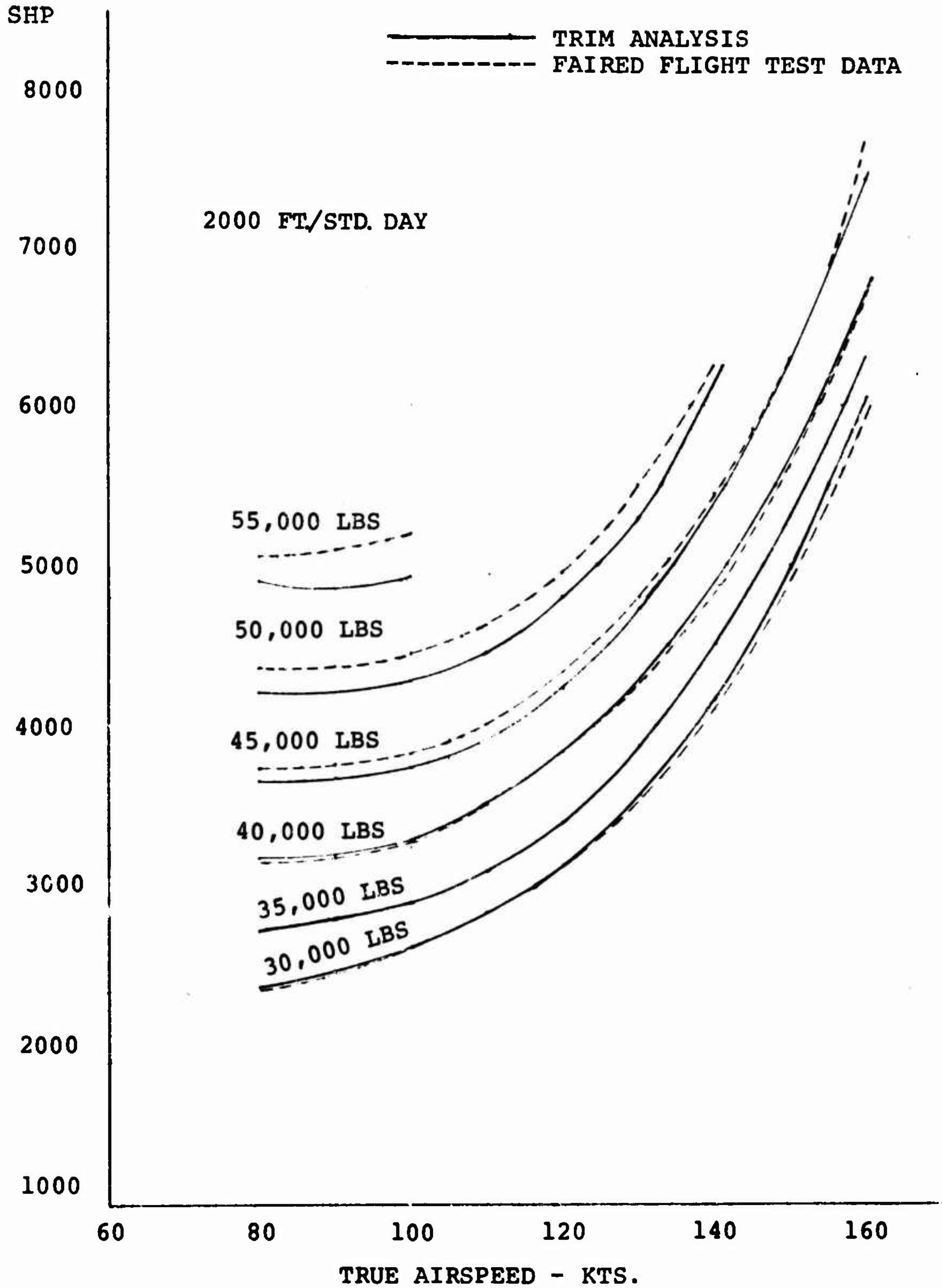


Figure 7. COMPARISON OF CH-47 FLIGHT TEST AND THEORY IN FORWARD FLIGHT

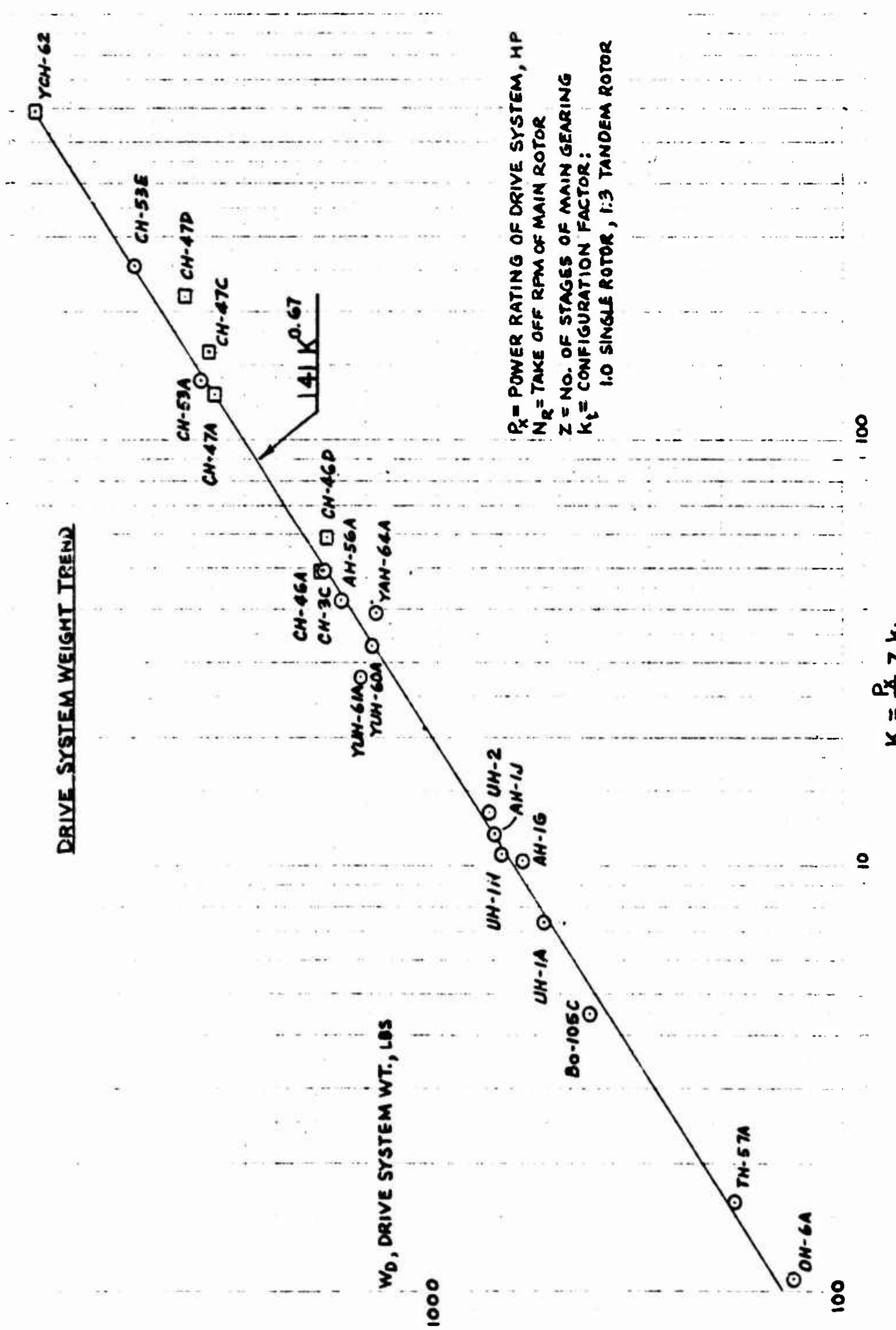


Figure 8 Drive System Weight Trend.

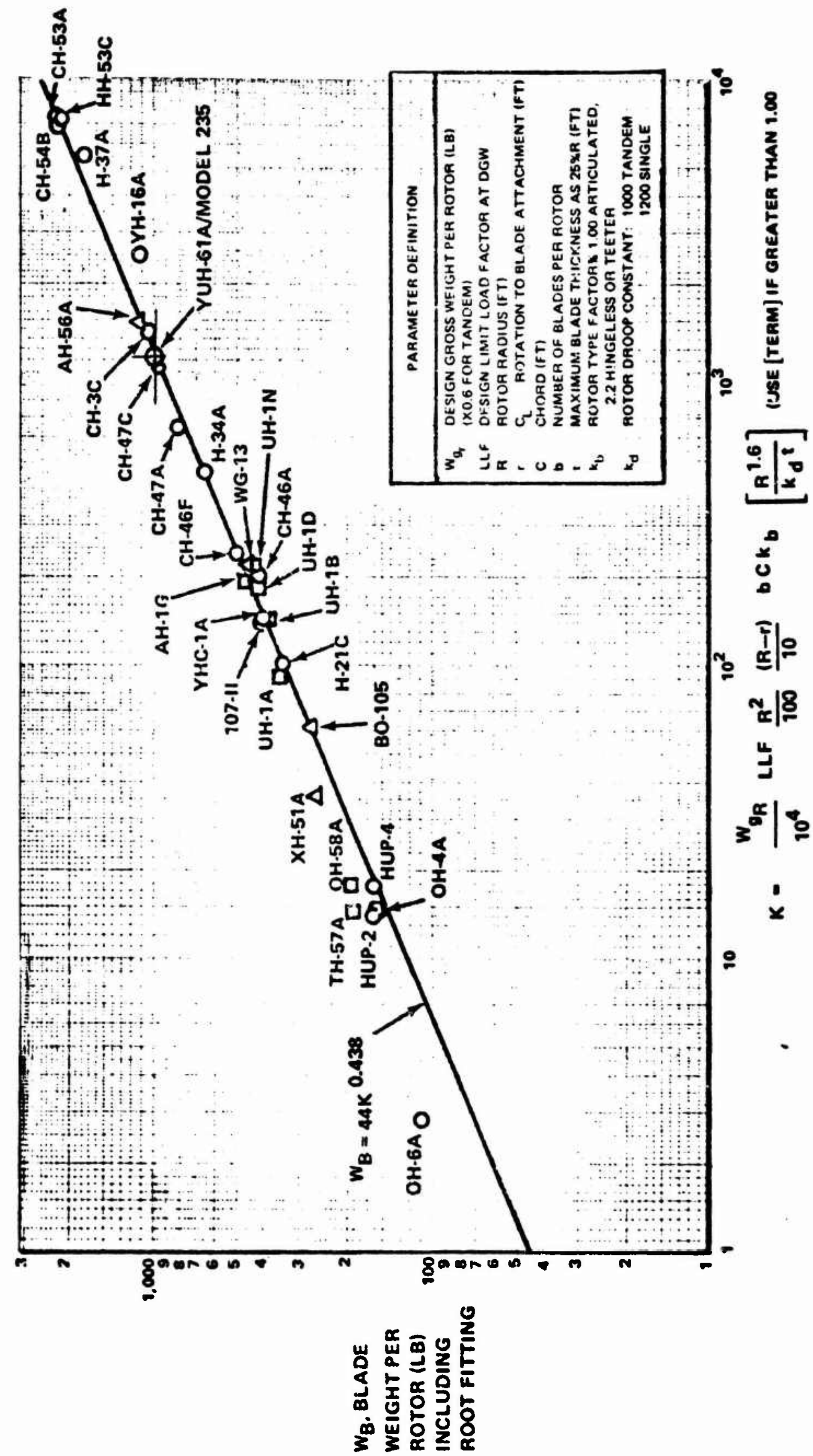


Figure 9 Rotor Blade Weight Trend

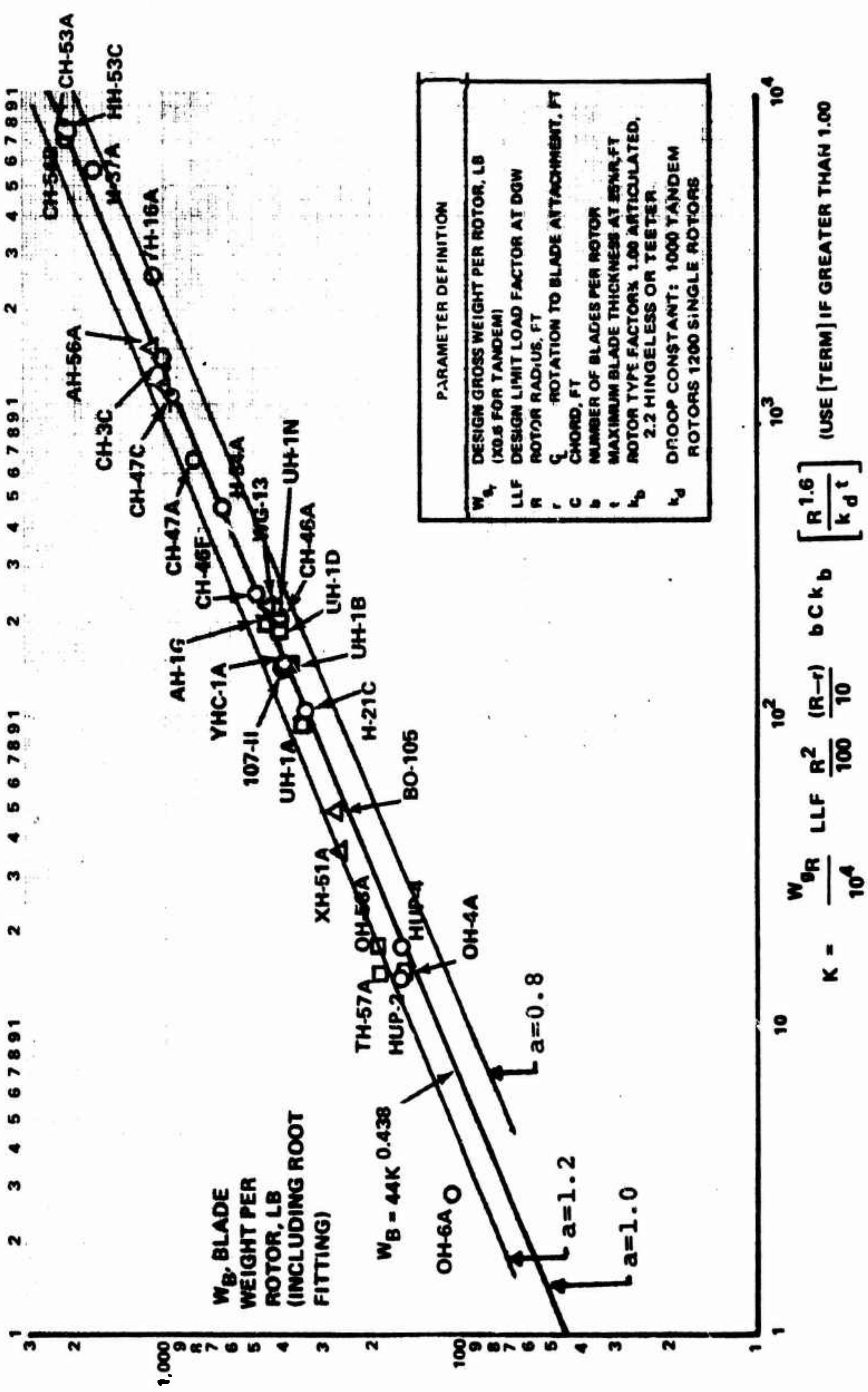


Figure 10. Hub and Hinge Weight Trend

manufacturers to design helicopters specifically to meet the needs of the business and commercial market without a military parent. Examples of this are the Sikorsky S-76, Bell 222, Augusta 109 and smaller single model developers such as Enstrom and Robinson helicopters. For this study helicopters in this category are called 'new' aircraft.

In developing the cost impact of noise reduction for 'in-production' helicopters all development costs not associated with noise reduction are charged to prior production. The costs associated with noise reduction, which may include redesign and retesting of previously qualified components, are written off against the projected quantity of remaining aircraft to be sold after modification.

In the case of 'new' aircraft it is assumed that all low noise design features are identified during the preliminary design stage of development. The only effect which noise constraints will have on the cost of a 'new' helicopter is in the degree to which the aircraft may be more complex to design and test, or more expensive to manufacture than one which did not consider noise.

Helicopter Pricing

Commercial helicopter prices are established by the manufacturer at a level which will provide the manufacturer a reasonable rate of return on his investment commensurate with the risks undertaken. The risks involved are technical risks, cost risks and market risks. Market risks can be defined as the ability to predict the market size and the ability to penetrate the market and achieve the predicated share of the market. Naturally, price has a large influence on the market's acceptance of the product and at times a manufacturer may have to establish a lower price which, in turn, increases his risk and results in a breakeven point further into the production quantity.

The breakeven point or breakeven quantity is the number of aircraft that must be sold at a given price to equal the sum of the non-recurring costs (development, testing, tooling, etc.) plus the recurring cost to produce the quantity of aircraft sold. Generally, the breakeven point should be reached within three to five years into the production cycle.

Since this study investigates a wide range of production quantities and vehicle sizes, it was not feasible to calculate breakeven quantities in the above described manner for each condition, so a simplified method was needed.

The method or procedure adopted for this study set the breakeven point at a percentage of the expected production quantity as an inverse function of helicopter gross weight as shown in Figure 11. Generally, the market for smaller aircraft is larger than for the heavier helicopters and the expected market share or production quantities would be greater and at a higher rate, thus permitting the breakeven point to be a greater percentage of the expected production quantity. Such relationships are shown in Figure 11 and should be representative of pricing practices.

In this study, development costs for noise modifications to an in-production aircraft or for a new development aircraft were spread over the breakeven quantity. It was estimated that a new type helicopter could be designed to meet any of the noise level limits studied herein at the same development cost because the design differences within each type were small.

For production pricing, the helicopter was divided into airframe, dynamic system, avionics and engine. For each baseline configuration, the Unit #1 and learning curve slope was estimated for each of these subsystems using industry average rates. Production cost changes for each noise configuration were also estimated by subsystem based on the detail description of the change and the associated weight estimate. These costs were then used with the assumed production quantities to calculate the cumulative average production price. To this was added the amortized development cost to give the flyaway price.

Direct Operating Costs

Direct operating costs (DOC's) include:

- o Flying Operations

    Flight crew
    Fuel oil
    Insurance

- o Maintenance

    Labor
    Material
    Burden

- o Depreciation

A modified AIAA Method (7) was used in this study to calculate DOC's. The advantages of this method are that the VTOL flight profile is recognized, and that the effects of inflation are calculated for personnel expenses, aircraft price and fuel price before entering the formula. Modifications to the formula adopted for this study were in the areas of flight crew expenses and maintenance burden, where Boeing procedures reflecting more recent air carrier history were applied. In addition, a 0.65 factor on maintenance costs was used to reflect current technology.

A significant share of seat-mile operating costs are represented by aircraft flyaway cost which, in turn, rests in large measure on the actual numbers of aircraft that will be produced. Since this factor is a variable for each aircraft program, the subject study evaluated seat-mile costs over the range of production quantities from 50 to 1000, producing a corresponding range of operating costs.

The costing methodology is based on an annual utilization rate of 1800 flight hours per year. Industry averages were used for flight crew and ground crew salaries and insurance rates. Depreciation was assumed over a 10-year period to zero residual. Fuel costs used were $1.00 per U.S. gallon. All costs were in 1980 dollars.

(7) "Revised Standard Method of Estimating Comparitive Direct Operating Costs of Turbine Powered VTOL Transport Aircraft", Aerospace Industries Association of America, Washington, D.C., December 1968.

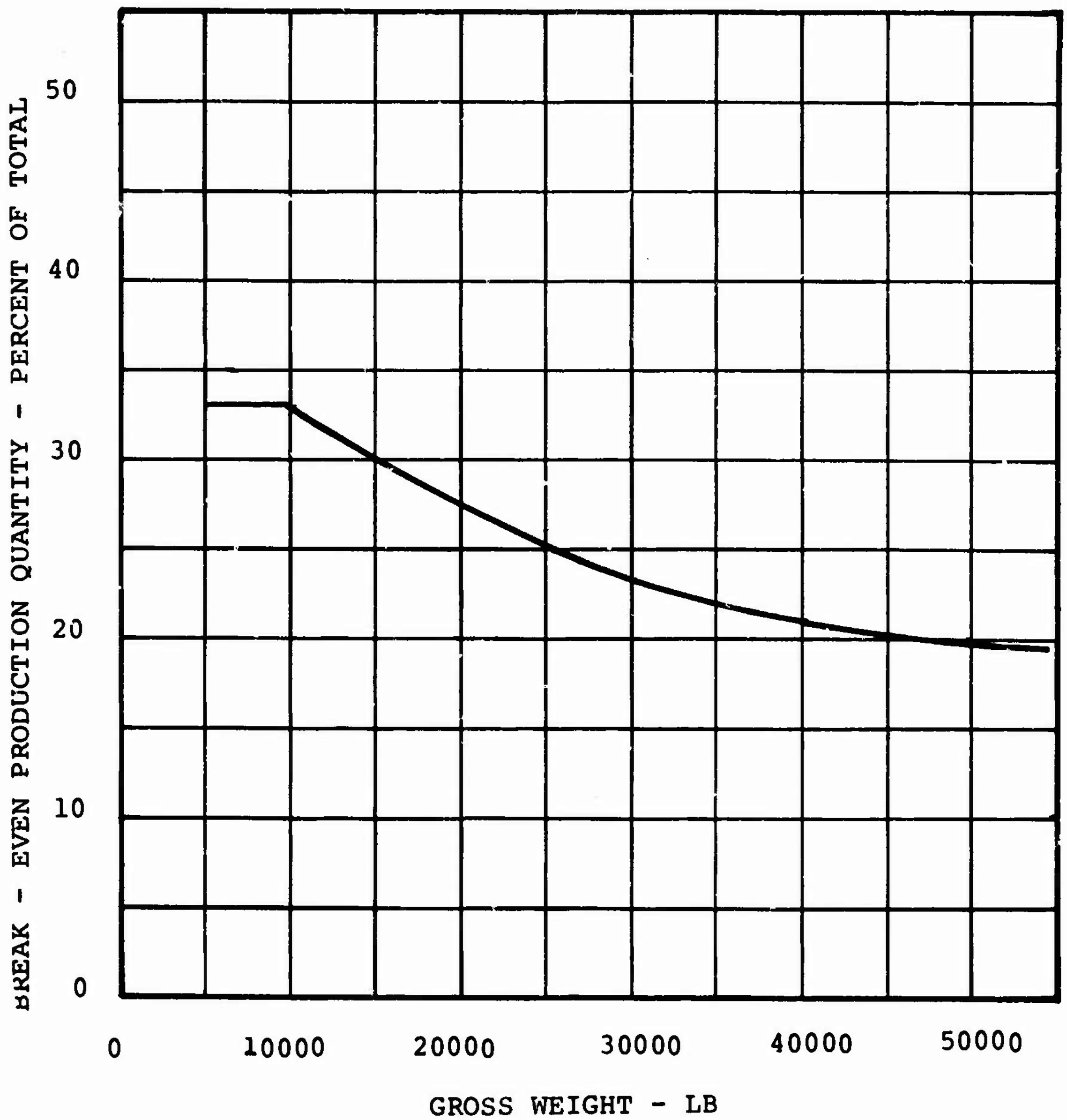


FIGURE 11 BREAKEVEN COST TRENDS

Final results are presented in terms of flyaway cost and available seat-mile operating costs. In each case, the cost impact is shown in percent change from the baseline configuration.

Figure 12 illustrates a sample of the cost program output evaluated for each helicopter and production quantity in the program. Each output sheet contains the following information as noted: (1) the flyaway cost, (2) the flight profile, and (3) the operating costs in terms of air-mile, seat-mile, flight hour and block hour costs.

BO-105 MOC 2 QUANTITY 1000 100 S MI EXIST PGM

1

WEIGHT OF AIRFRAME	1542.	TOTAL COST	830556.	CRUISE SPEED (MPH)	141.00
WEIGHT OF DYNAMIC SYS.	921.	COST OF AIRFRAME	445185.	CRUISE SPEED (KMPH)	226.91
UTILIZATION RATE (FH/YR)	1800.	COST OF DYNAMIC SYS.	266103.	BLOCK SPEED (MPH)	134.29
NUMBER OF SEATS	4.	COST PER ENGINE	59634.	BLOCK SPEED (KMPH)	216.11
BLOCK FUEL (LBS)	281.	$/LB FOR AIRFRAME	289.	ENGINE RATING (SHP)	400.00

2

VTOL FLIGHT PROFILE SEGMENT	TIME REQUIRED (MINUTES)	DISTANCE CREDIT (STATUTE MILES)	DISTANCE CREDIT (KILOMETERS)
TAXI OUT	0.50	0.00	0.00
TAKEOFF	0.50	0.00	0.00
CLIMB	2.00	7.66	12.36
CRUISE	37.68	88.55	142.50
DESCEND	1.00	2.30	3.70
AIR MANEUVER	0.00	0.00	0.00
APPROACH & LANDING	2.50	1.47	2.37
TAXI IN	0.50	0.00	0.00
TOTAL	44.68	100.00	160.93

3

DIRECT OPERATING COST	COST PER AIR MILE	COST PER SEAT MILE	COST PER AIR KM	COST PER SEAT KM	COST PER FLIGHT HR	COST PER BLOCK HR
FLYING OPERATIONS						
FLIGHT CREW	0.395922	0.098980	0.246026	0.061506	54.383800	53.166650
FUEL AND OIL	0.433313	0.108328	0.269260	0.067315	59.519820	58.187710
HULL INSURANCE	0.103083	0.025771	0.064056	0.016014	14.159490	13.842590
TOTAL FLYING OPERATIONS	0.932318	0.233079	0.579342	0.144836	128.063100	125.156900
DIRECT MAINTENANCE - FLIGHT EQUIP.						
AIRFRAME - LABOR	0.166085	0.041521	0.103205	0.025801	22.813430	22.302846
MATERIAL	0.037762	0.009440	0.023465	0.005866	5.186956	5.070868
ENGINES - LABOR	0.175273	0.043818	0.108914	0.027229	24.075420	23.536590
MATERIAL	0.065382	0.016346	0.040629	0.010157	8.980919	8.779914
DYNAMIC SYSTEM - LABOR	0.021156	0.005290	0.013148	0.003287	2.906283	2.841236
MATERIAL	0.044825	0.011207	0.027857	0.006964	6.157778	6.019961
TOTAL DIRECT MAINTENANCE	0.510489	0.127622	0.317218	0.079304	70.120780	68.551420
MAINTENANCE BURDEN	0.543774	0.135943	0.337901	0.084475	74.692700	73.021020
TOTAL MAINTENANCE	1.054262	0.263566	0.655118	0.163780	144.813400	141.572300
DEPRECIATION - FLIGHT EQUIP.	0.405684	0.101401	0.252042	0.063011	55.713790	54.466850
TOTAL DIRECT COST EXCL MAINT BURD	1.848410	0.462102	1.148602	0.287150	253.897400	248.215000
TOTAL DIRECT COST INCL MAINT BURD	2.392182	0.598046	1.486502	0.371625	328.590000	321.235800

Figure 12. Typical Printout of Cost Prediction Program

IV. NOISE REDUCTION TECHNOLOGY

Achievement of targeted noise levels for minimum penalties of performance, weight and cost requires a balanced acoustical design in which the noise source establishing the Perceived Noise Level is reduced initially followed successively by each next more important source. Identification of each of the noise sources combined with an assessment of the flyover time history of the NOY values for each aircraft, described in the previous section, was the initial task.

It was then necessary to define noise reduction trends for each of these sources and to quantify the reduction in noise available for each approach. A discussion of the methods available for reducing helicopter noise is presented in the following sections.

1. Tip Speed Reduction. The most effective method for reducing rotor noise for many helicopters is the reduction of rotor tip speed. Rotor tip speed reduction is effective in controlling rotational noise, broadband noise, thickness noise, and noise due to blade-vortex interaction, on both main and tail rotors. The sensitivity of each of these noise components to tip speed, however, is different and therefore the effect of reducing tip speed is very dependent on the relative levels of the various noise sources for a given helicopter. These effects were evaluated for each helicopter studied.

In order to preserve the performance of a rotor, it is necessary to maintain the design lift coefficient:

$$C_\ell = \frac{6T}{V_t^2 \rho A_b}$$

where

T thrust

V_t Tip Speed

ρ Air Density

A_b Total Blade Area

Hence any reduction in tip speed will require the blade area to increase as a second order function. This can be achieved either by increasing the blade chord or the number of blades.

2. Rotor Design. A second technology area involves the design of the rotor blade, including blade thickness and planform, shape, airfoil, twist and stiffness. Of these, the thickness effect, particularly at high tip speeds, is most important. The methodology for predicting the effect of both blade thickness and compressibility on noise level is presented in the Appendix. Figure 13 (reproduced from Reference (8)) illustrates the effect of rotor operation above drag divergence on PNLT as well as flyover EPNL. When the tip

(8) Sternfeld, H. and Wiedersum, C. W., "Study of Design Constraints on Helicopter Noise", NASA CR 159118, July 1979.

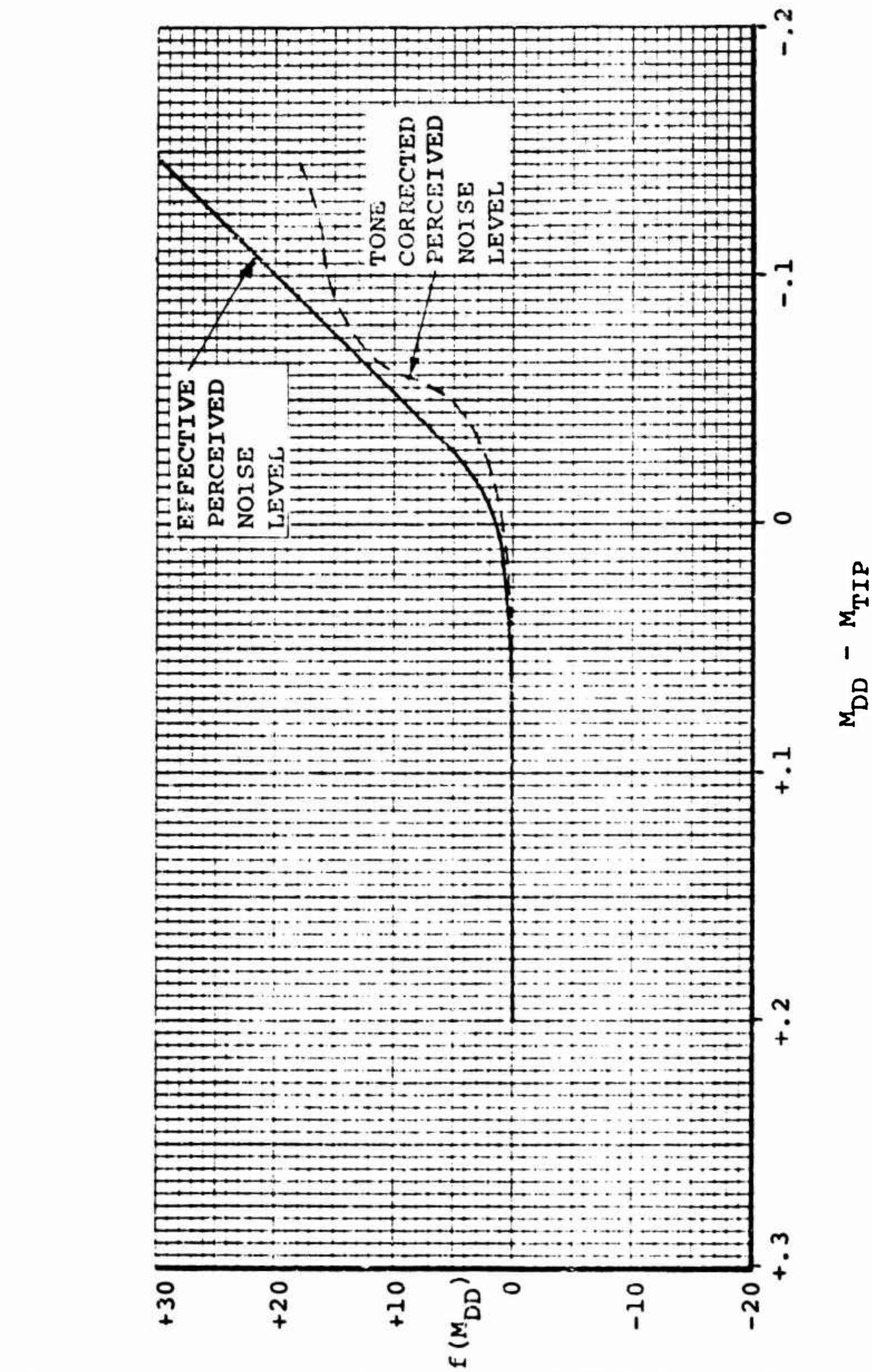


Figure 13. Forward Flight PNLT and EPNL Estimating Function

Mach number becomes greater than the drag divergence Mach number for a given airfoil (M_{DD} - M_{TIP} is negative), rotor noise levels increase substantially. The trend illustrates that an airfoil with higher drag divergence Mach numbers delay the onset of drag divergence effects on rotor noise.

Tip shape changes have been shown to demonstrate reductions in sound pressure levels of 2-3 dB, although these noise reduction are sometimes accompanied by increases in power required resulting from increased tip losses. Advanced airfoils have also demonstrated reductions in noise that can be attributed to their sections and modified chordwise pressure distributions. Recent tests of an advanced airfoil on a tandem rotor configuration have demonstrated noise reductions of a least 5 dB when compared with an older design airfoil.

3. Main/Tail Rotor Separation. A significant source of noise on some helicopters arises from the interaction of the wake shed by the main rotor with the tail rotor. These interference effects due to inflow turbulence and the trailed tip vortex of the main rotor result in an impulsive or buzzing acoustic signature of the tail rotor that is not present when the tail rotor operates in an isolated environment. Tail rotor noise resulting from this disturbed inflow can dominate helicopter flyover noise including Perceived Noise Levels. Investigations into reducing this component of tail rotor noise have been reported in References (9) and (10). Levine (9), has shown analytically that separation of the main and tail rotors by a distance equivalent to 12% of the tail rotor diameter results in a tail rotor noise signature that is equivalent to an isolated, free rotor. It was shown in the same study that main and tail rotor separation by lateral offset of the tail rotor produced minimum interference effects for all flight conditions. Vertical and longitudinal separation were less effective approaches in terms of the number of flight conditions for which noise was reduced. Balcerak (10) reporting the results of a model test program, showed a reduction in tail rotor noise of up to 10 dB when the tail rotor was moved one-tenth of the tail rotor diameter to the left of the fin.

4. Tandem Helicopter Rotor-to-Rotor Separation. Rotor-to-rotor wake interference on an overlapped tandem rotor helicopter can be reduced by separation of the rotor disc planes in two ways. The first approach involves selection of longitudinal cyclic pitch on each rotor such that the tip path planes in the overlapped region are separated by at least two feet. The major disadvantage to this approach is that increased rotor flapping generally increases rotor shaft bending moments and thus reduces component life. In addition, fuselage trim attitude changes and may result in increased drag forces.

(9) Levine, L. S., "Analytic Investigation of Techniques to Reduce Tail Rotor Noise", NASA CR-145014, 1 July 1976.

(10) Balcerak, J. C., "Parametric Study of the Noise Produced by the Interaction of the Main Rotor Wake with the Tail Rotor", RASA Division, Systems Research Laboratories, Inc. RASA Report No. 76-14-01. NASA Contract NAS1-13690.

A second approach involves both a reduction in rotor overlap as well as disc plane separation by extending the length of the fuselage and increasing the height of the aft rotor pylon. This method of rotor disc separation requires an airframe modification which results in increased vehicle weight and drag.

5. Drive System Component Noise Reduction. For some low noise configurations defined in this study, noise levels of main and tail rotors have been reduced to a level where normally inaudible noise sources contribute to a portion of the flyover noise spectra and thus the EPNL for that aircraft. This is frequently found to be noise radiated from drive system components. Such a case involved a main rotor transmission on the BO-105.

6. Engine Noise Reduction. Review of data from the aircraft under study indicates that at the present time the only component of engine noise which occasionally causes problems is compressor inlet noise. This is particularly important if the pure tone level is sufficient to generate a correction term in the PNLT calculation. Combustion and core noise, for the aircraft studies, did not appear to be an important factor, but could become so if further reductions in rotor noise had been achieved. Noise due to the jet is of little concern due to the very low exit velocities encountered with turbo-shaft engines.

The engine noise reduction treatments used for helicopters are similar to those employed on subsonic jet airplanes. Tuned honeycomb absorbative inlets may increase the complexity of anti-icing provisions, however.

Limits of Applicability of Noise Reduction

In many cases, the changes to an 'in-production' aircraft may be more extensive than only the modifications to noise generating components. Any modification to an aircraft component or system increases the risk to the operation of that system and represents a potential limitation to the operation of the helicopter. For example, a reduction in rotor speed reduces the frequency of the rotor forces transmitted to the airframe and this may require retuning of vibration reduction systems and requalification of mechanical instability tests. Reduction of rotor speed will cause the transmission speeds to decrease and may then require modifications of gear driven electrical generators, pumps and other accessories.

During an autorotative maneuver the kinetic energy of the rotor is expended during the flare just prior to touchdown by increasing collective pitch, thus reducing rotor speed by generating thrust to slow the descent rate. Any modification to the helicopter which reduces rotor kinetic energy will result in increased rate of descent. As a result, the strength and dynamic characteristics of many parts, as well as the flying qualities of the total aircraft, would have to be checked, and possibly modified due to a change of an 'in-production' helicopter.

V. HELICOPTER COST/BENEFIT TRADE INVESTIGATIONS

In order to present the cost/benefit trade investigation in a concise manner, each aircraft is described in separate subsections of the report in terms of its noise sources, configuration changes, the impact which these configuration changes have on performance, weight and cost, as well as a discussion of any other factors which could limit normal operation of the helicopter.

BO-105

Vehicle Description

The BO-105 is a five-place, twin turbine, single rotor helicopter with a maximum gross weight of 5070 pounds. A three-view is shown in Figure 14. The BO-105 has a four-bladed rigid rotor with glass fiber reinforced plastic blades and has been flying since 1967. A description of the configuration is presented in Table 1.

Noise Sources

Identification of noise sources of the BO-105 was determined from measured flyover noise data for a centerline microphone with the aircraft at an altitude of 500 feet and an airspeed of 108 knots ($0.9V_H$). The initial analysis was a determination of the PNL time history from tone corrected spectra. The narrowband spectra and 1/3 octave spectra also were produced at two-second intervals for the flyover and main rotor, tail rotor, turbine and other sources identified as shown in Figures 15 and 16.

As shown in Figure 15, the dominant noise source on the BO-105 is the tail rotor which also determines PNL_{max} for flyover noise. For a period of approximately four seconds prior to the overhead position, the Noy-max value is set by tail rotor noise on each of the 1/3 octave band spectra. Also identifiable are main rotor harmonics of blade passage although the Noy value for these frequencies are not main contributors to the PNL. The input pinion bevel gear mesh frequency can be identified in Figure 15, varying from approximately 1900 Hz on approach to just over 1400 Hz for the departing helicopter.

Configuration Changes - Tradeoff Variables

Two configuration changes have been defined for the BO-105 with respect to the baseline aircraft which displays a flyover noise level of 89.5 EPNdB as measured by ICAO procedures. These changes are identified in Table 2.

The configuration changes to the BO-105 to achieve noise reductions are illustrated in Figure 17. Modification (Mod) 1 reduces tail rotor tip speed from 722 ft/sec. to 702 ft/sec. by the use of an advanced airfoil and by increasing tail rotor blade twist. The advanced airfoil also allows the reduction in tip speed without loss in performance of this rotor. The existing airfoil is a NACA 0012, and this was changed to an airfoil producing a higher rotor L/D with a revised twist schedule. The reduction in tip speed would be obtained with a modified pinion gear in the tail rotor transmission located at the rotor hub. The hub itself would require strengthening due to the lower rotational speed. Data obtained by Messerschmitt Boelkow Blohm, on a tail rotor test stand indicates a noise reduction of the order of 3 EPNdB is achievable.

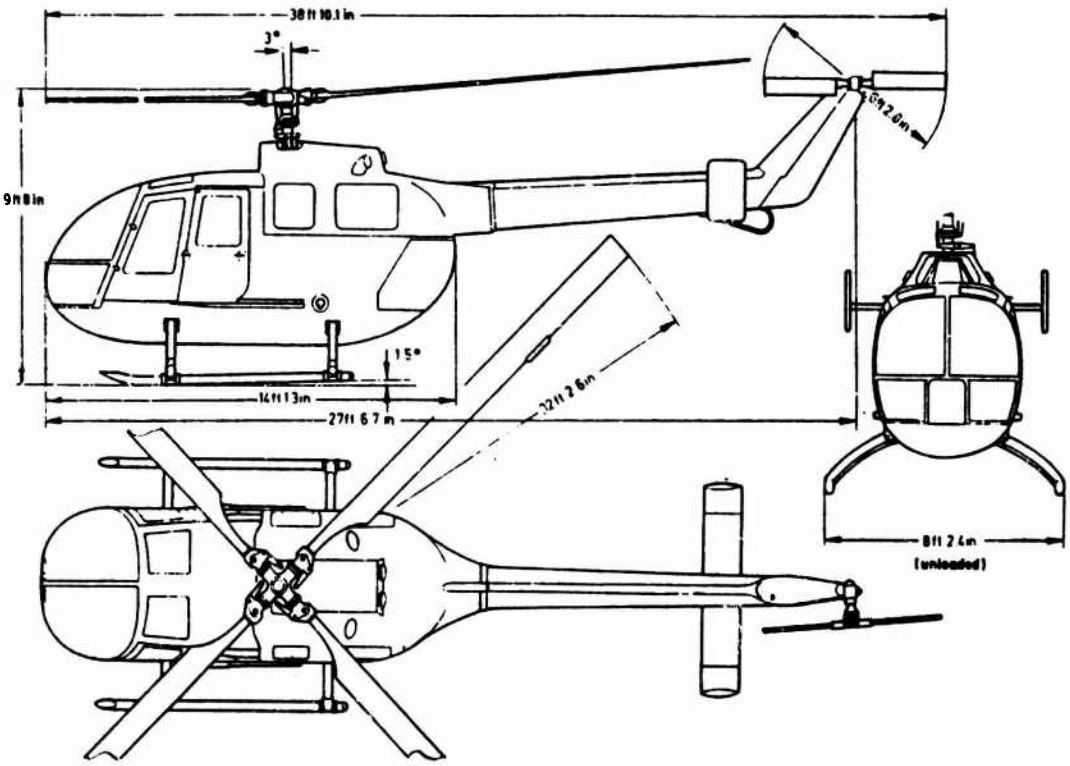


Figure 14. BO-105 General Arrangement

Table 1. BO-105 Characteristics

Maximum Takeoff Gross Weight	5070 Pounds
Operating Weight Empty	2949 Pounds
Engines	Allison 250-C20
Main Rotor	
Type	Hingeless
Number of Blades	4
Radius	16.09 Feet
Chord	10.6 Inches
Tail Rotor	
Type	Teetering
Number of Blades	2
Radius	3.1 Feet
Chord	6.8 Inches
Normal Cruise Speed	125 Knots
Main Rotor RPM (100%)	425
Tail Rotor RPM	2224

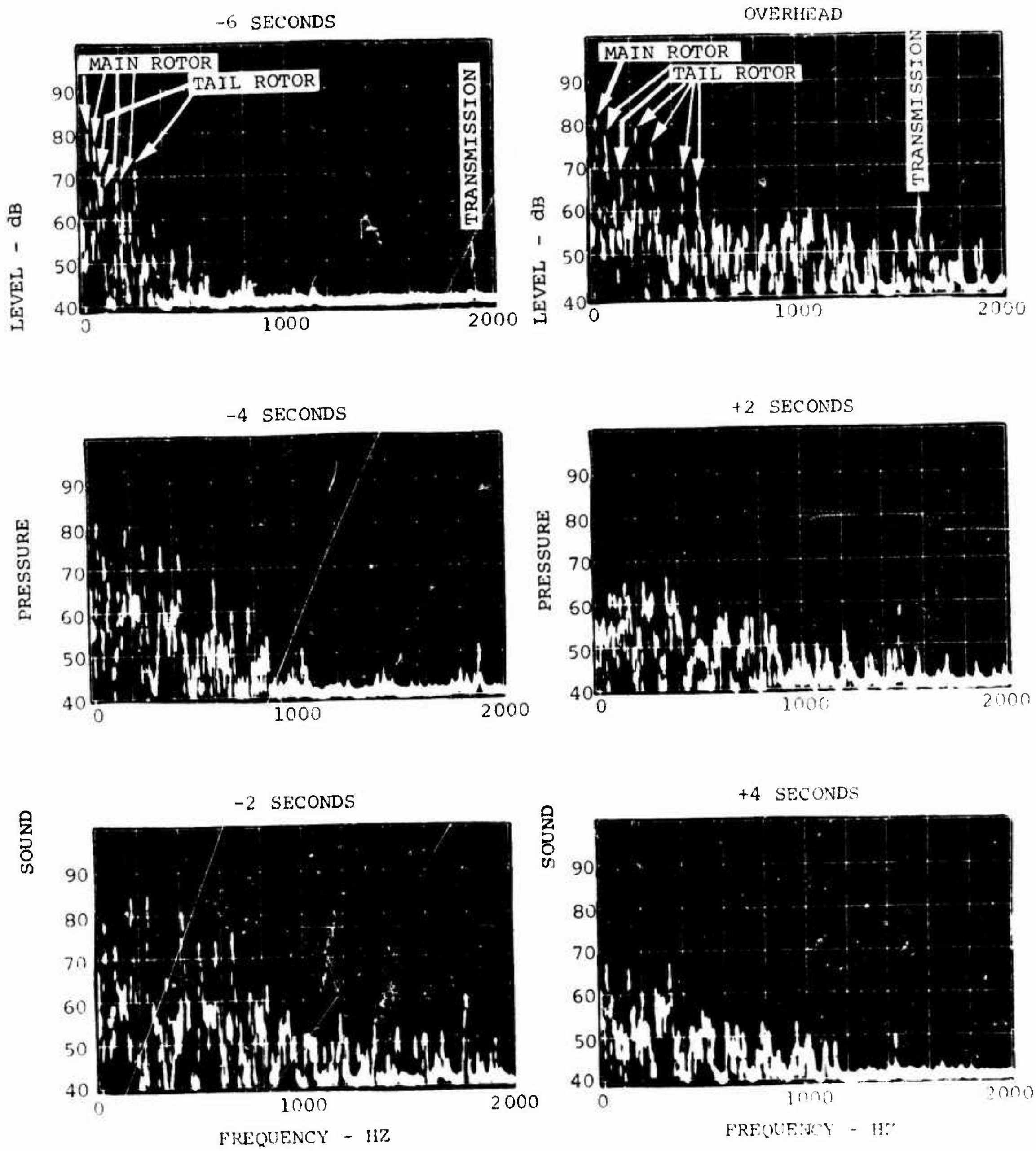
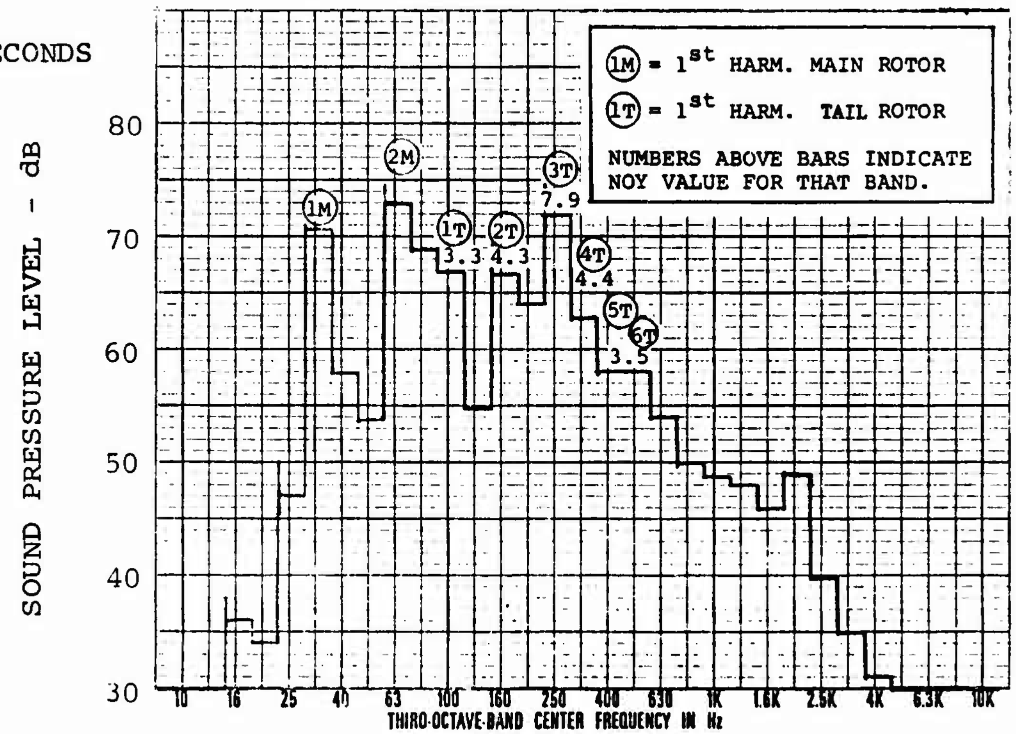


Figure 15. Narrowband Spectra of BO-105
Flyover Noise

-6 SECONDS



-4 SECONDS

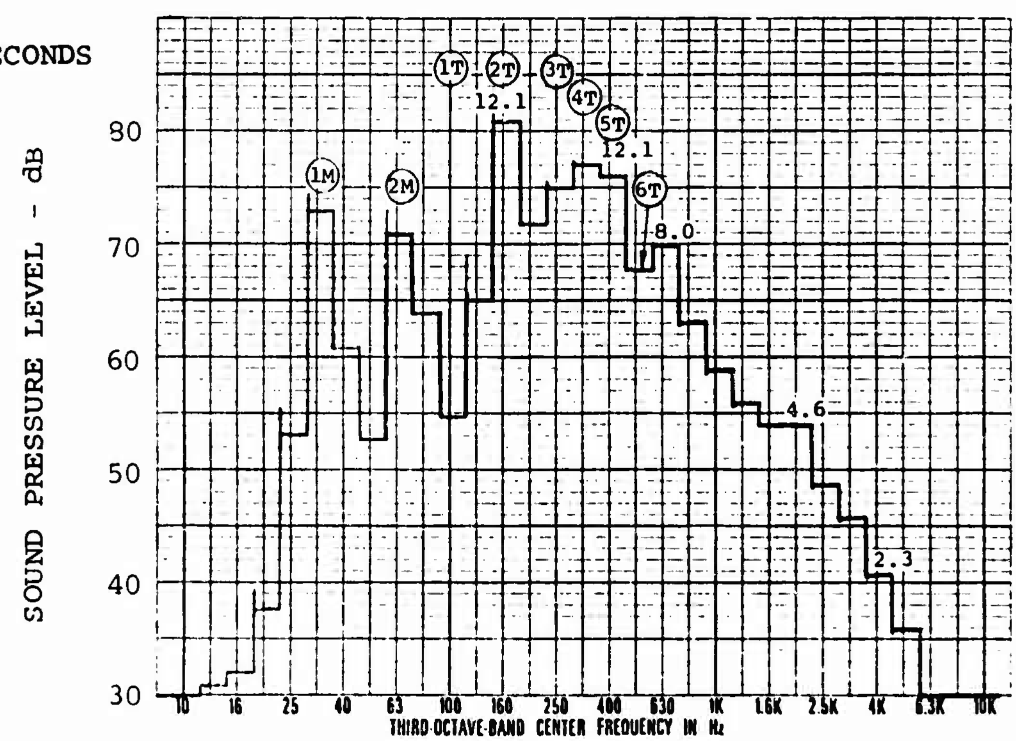
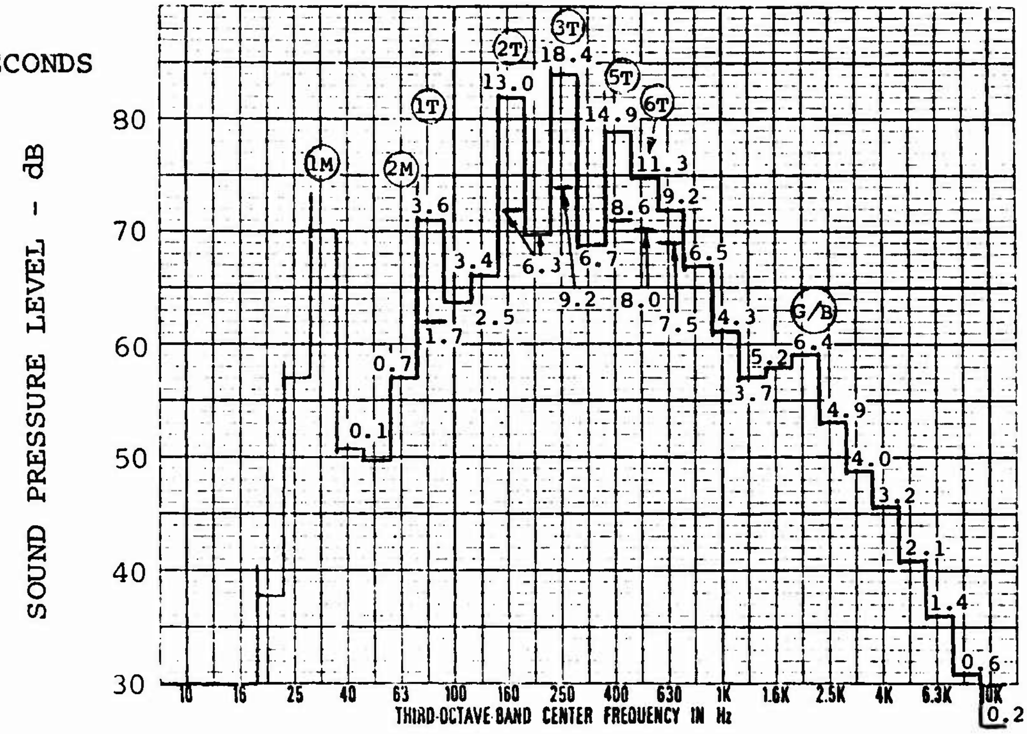


FIGURE 16. ONE THIRD OCTAVE SPECTRA OF Bo-105 FLYOVER 6 SECONDS AND 4 SECONDS PRIOR TO OVERHEAD

(Sheet 1)

-2 SECONDS



OVERHEAD

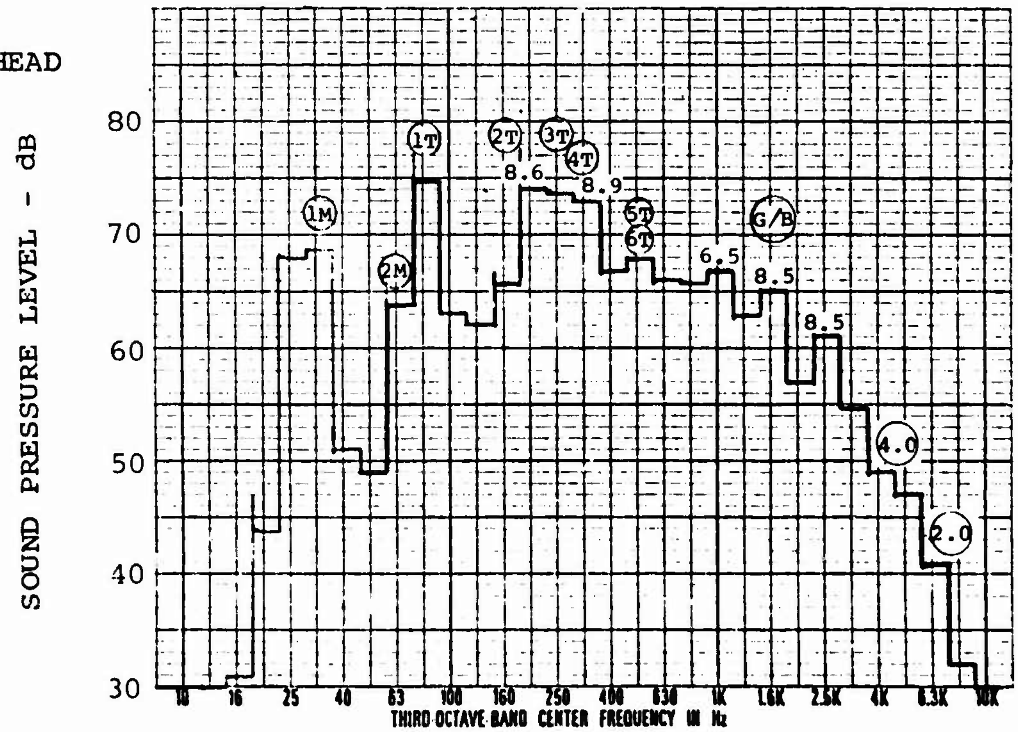
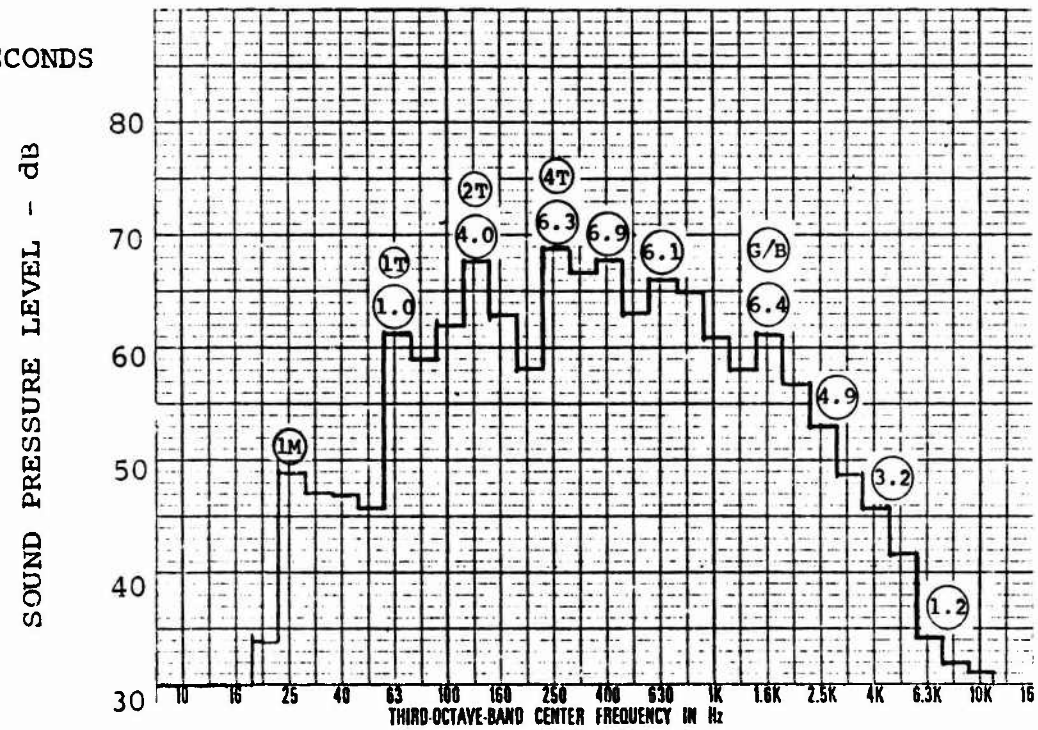


FIGURE 16. ONE THIRD OCTAVE SPECTRA OF Bo-105 FLYOVER 2 SECONDS PRIOR TO AND OVERHEAD

(Sheet 2)

+2 SECONDS



+4 SECONDS

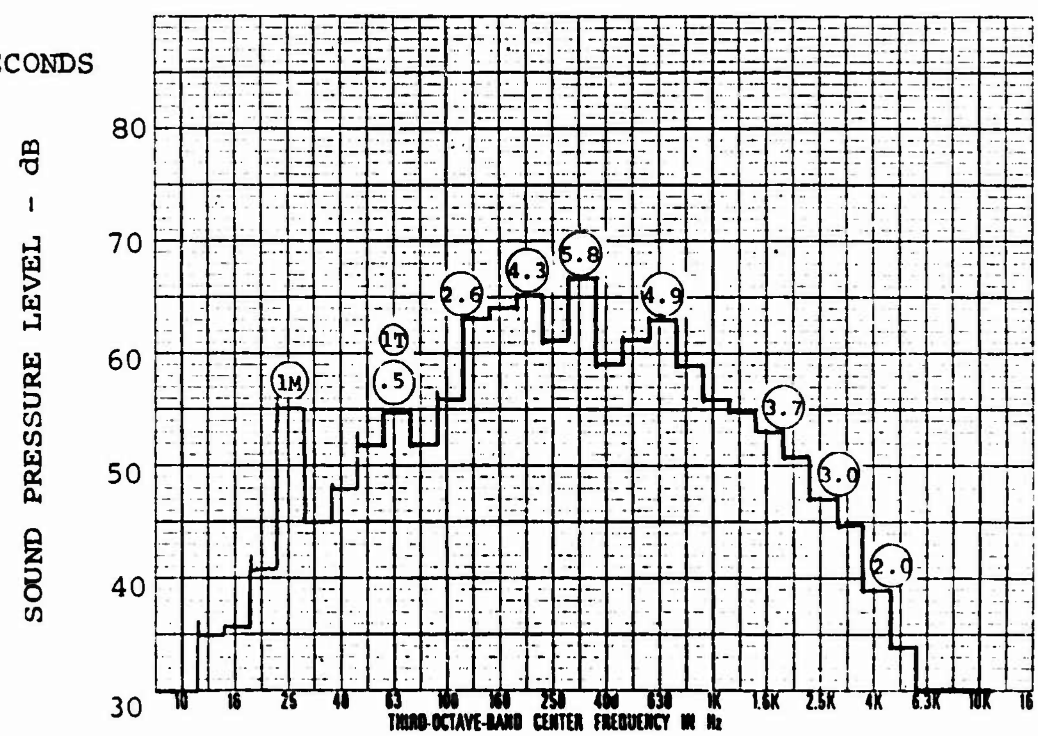


FIGURE 16. ONE THIRD OCTAVE SPECTRA OF Bo-105 FLYOVER 2 SECONDS AND 4 SECONDS AFTER OVERHEAD

(Sheet 3)

Table 2. BO-105 Configuration Changes

	Baseline	Modification 1	Modification 2
MAIN ROTOR			
V_t (ft/sec)	716	716	700
RPM	425	425	415
No. of Blades	4	4	4
Airfoil	23012	23012	23012
Chord (ft)	0.883	0.883	0.971
TAIL ROTOR			
V_t (ft/sec)	722	702	702
RPM	2224	2162	2162
No. of Blades	2	2	2
Airfoil	0012	Advanced airfoil, higher L/D, increased twist.	Same as Mod. 1 plus 10% increase in solidity.
Chord (ft)	0.58	0.58	0.61
Flyover EPNL	89.5	86.5	83.5
Dynamic System	Basic	New T/R speed, T/R gearbox.	M/R transmission acoustical treatment.
Airframe	Basic	Basic	Tail Rotor offset laterally by 1.77 ft.
Powerplant	Allison 250-C20	Allison 250-C20	Allison 250-C20
Weight Change (lb)		1.5	56.5

Mod 2 has been altered more substantially in order to achieve flyover noise levels 6 EPNdB below the baseline helicopter. In addition to the changes of Mod 1, main rotor tip speed has been reduced from 716 ft/sec. to 700 ft/sec. to obtain a required reduction in broadband noise. Also, the tail rotor as defined for Mod 1 has been modified further by laterally offsetting the tail rotor disc an additional 1.77 feet to the left. This offset provides additional clearance between main and tail rotors and has been incorporated into the design to reduce both the amplitude and impulsive characteristic of tail rotor noise. The criteria for this offset was developed in References 9 and 10. The impact of this design change on aircraft flying qualities and handling is discussed in the following section. The final change required directly for aircraft noise reduction is an acoustical treatment of the main rotor transmission to reduce the noise of this component in the direction of forward flight. This would require an enclosure of the transmission case and would be fabricated as a molded elastomeric product with a surface density of 1-2 lb/ft^2. Several manufacturers of acoustical products currently market such a material and since the input pinion gear tooth mesh is in the higher frequency range, reduction of levels by 5-7 dB is well within design limits.

Impact of Design Changes on Performance, Weight and Cost

Performance

A sensitivity study was performed to examine the effect which the reduction in main rotor tip speed has on the aerodynamic performance of the BO-105. The impact of main rotor tip speed reduction on aircraft gross weight is shown in Figure 18, along with related FAR Part 27 requirements and flying quality standards. Although not a requirement of FAR Part 27, the existing maneuverability capability of the BO-105 was maintained as a criterion for this study.

As shown in Figure 18, any reduction in tip speed with the existing rotor would result in a decrease in maximum gross weight, which would come directly out of payload or range (fuel). In order to maintain the basic mission profile, therefore, an increase in rotor solidity was required. In reality, a modification to the rotor system would not likely be made for less than a 20-25 percent increase. A 25 percent solidity increase could be accomplished through either addition of a fifth blade to the rotor or by a 25 percent increase in chord of each of the four existing blades. The second approach was adopted to maintain dynamic system frequencies for which the airframe had been proven. This dictated development of a new blade, but this was estimated to be less costly and contain less risk than the addition of a fifth blade. A fifth blade would require a redesign of the upper control system, a new rotor hub and an extension of the tail boom to provide for an increase in rotor radius resulting from a requirement to spread the root end fittings. Blade redesign utilizing the existing four-bladed rotor would be preferable from the design standpoint as well, since it also would allow the most recent airfoil technology to be incorporated into a new blade, with its associated performance improvements. The 25 percent increase in solidity by way of a fifth blade with the unimproved airfoil would not be as efficient an approach.

An increase in rotor solidity by 25 percent also introduces sufficient growth into the rotor such that future reductions in flyover noise of derivative aircraft required by certification would be achievable without unreasonable increases to the rotor system costs.

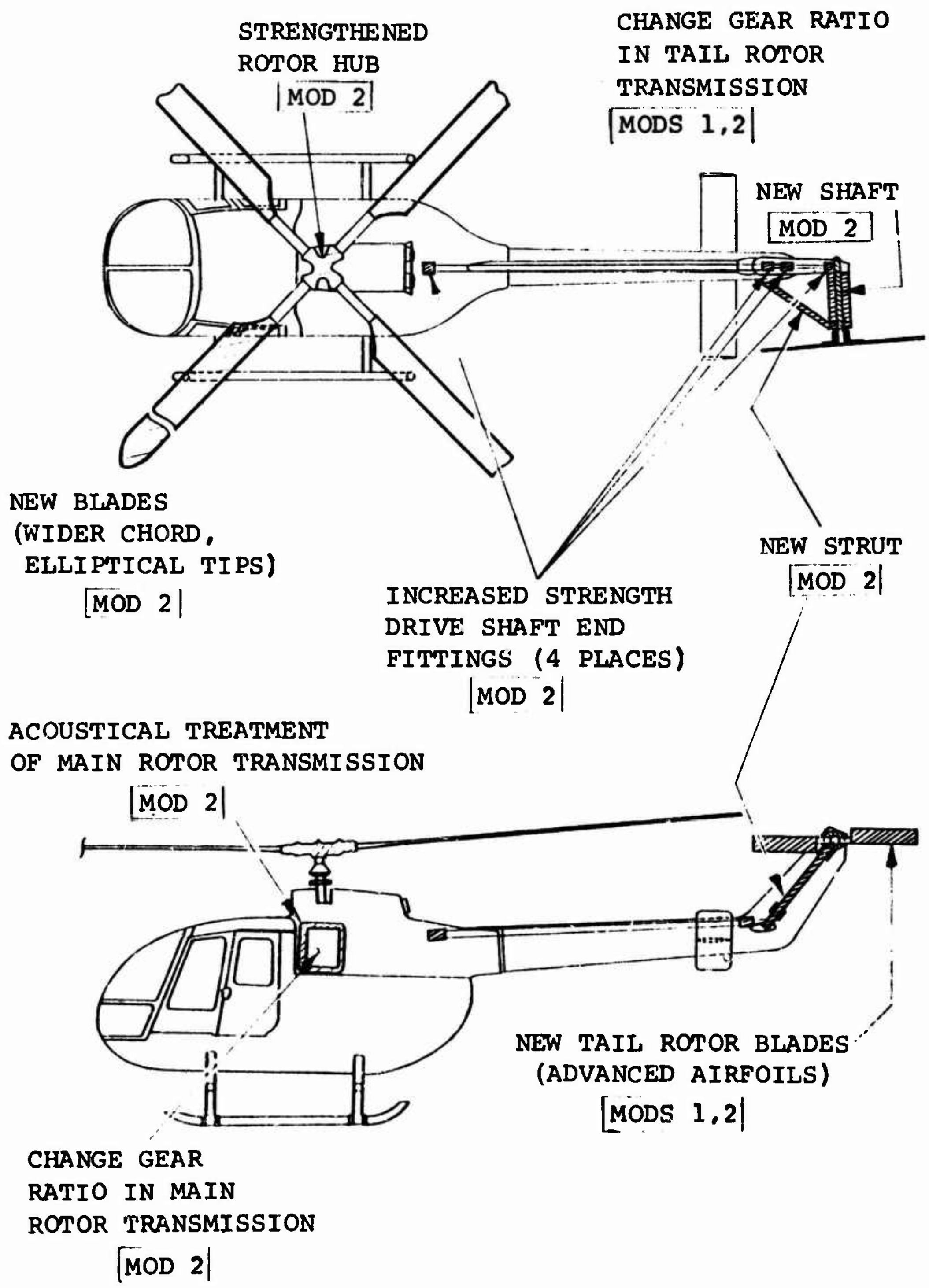


FIGURE 17. CONFIGURATION CHANGES TO BO-105

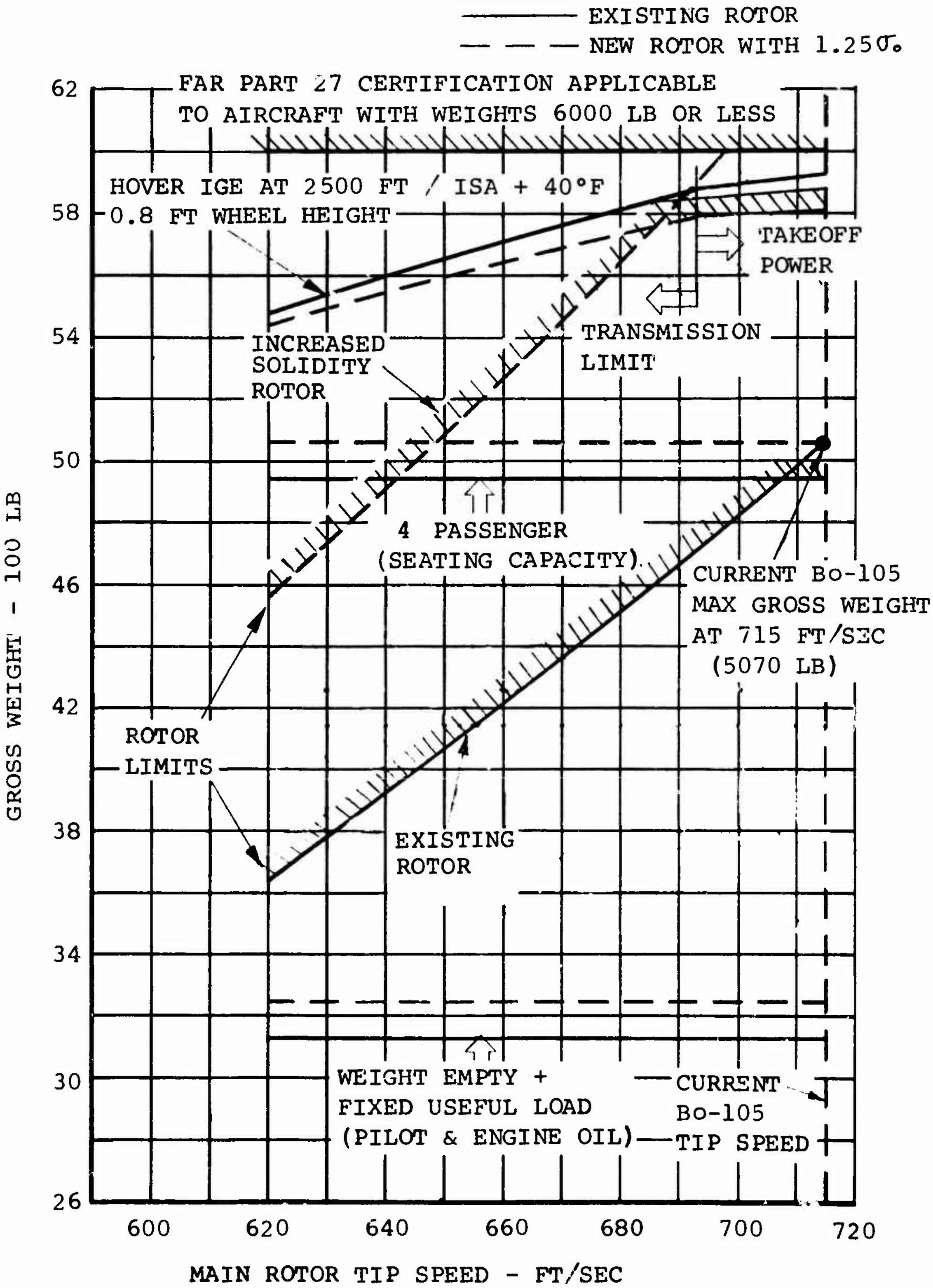


Figure 18. Bo-105 Performance Limitations

The additional chord added to the main rotor blade would increase aircraft weight empty from approximately 3130 pounds to 3240 pounds, in increase of 3.5 percent.

The effect of tip speed reduction and increase in rotor solidity on aircraft speed and range are shown in Figure 19. Below V_t = 707 ft/sec, there is a discontinuity in the trend due to the increased rotor solidity. Best range speed improved slightly due to increased rotor thrust below 707 ft/sec, and range decreases from 232 miles to 217 miles (6.5%) as a result of an increase in rotor power required.

Weight

The effect of design changes on BO-105 weight is presented in Table 3. The changes defined by Modification 1 result in a minimal increase in weight (1.5 pounds) since the advanced airfoil tail rotor blades are designed to the same criteria as the existing 0012 section blades. The increase in drive system torque is accommodated by the existing dynamic system, although a new gear set has been installed in the tail rotor transmission resulting in a weight increase of 0.5 pound.

Costs

The BO-105 has been treated as an 'in-production' helicopter for the purpose of determining the cost impact of noise reduction changes. As such all original development costs were assumed to have been charged to prior production, and additional nonrecurring costs for noise reduction were spread over the remaining production quantities. Figures 20 and 21 show the effect of noise reduction on the factory flyaway cost and direct operating cost, respectively, for a wide range of production quantities.

Model 179

Vehicle Description

The Boeing Vertol Model 179, as used for the present study, is the 19 passenger twin turbine helicopter shown in Figure 22. The aircraft has a four-bladed rigid main rotor with glass fiber reinforced plastic blades. Maximum takeoff gross weight is 17,400 pounds and cruise speed is 137 knots (99% best range). Table 4 presents some pertinent characteristics of the Model 179.

Noise Sources

Noise source identification for the Model 179 was determined from flybys of the parent aircraft, the YUH-61A. Data were available from a flyby at an altitude of 137m above the microphone and a 61m sideline distance at 140 knots ($0.9V_H$). These levels were adjusted to estimate average flyover levels based on centerline and 150m sideline microphones for a 150m altitude flyby. A Perceived Noise Level time history was determined from tone-corrected spectra and the Effective Perceived Noise Level calculated. The narrowband spectra and 1/3 octave band spectra were produced at two-second intervals for the flyover and sources identified as shown in Figures 23 and 24. On approach, the main rotor dominates in the spectra while the tail rotor dominates for approximately four seconds prior to PNL_{max}. These sources are identified on the illustrations.

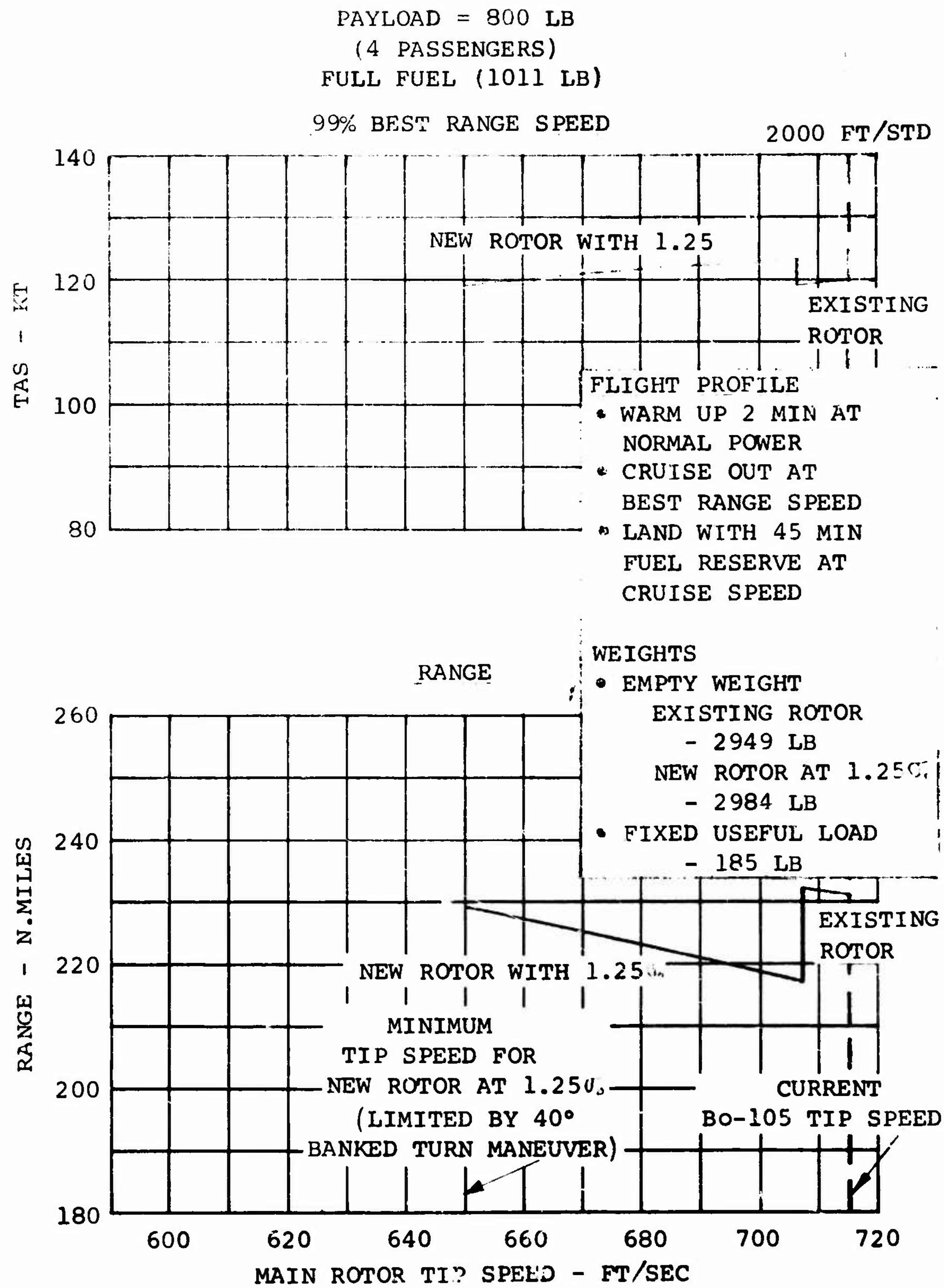


Figure 19. Effect of Tip Speed on Bo-105 Range Capability

Table 3. Impact of Design Changes On BO-105 Weight

A/C Mod	Component	Baseline Weight (Lb)	Additional Weight (Lb)	Description of Change
1	Tail Rotor Transmission	15	0.5	Change Gear Ratio in Tail Rotor Transmission
	Tail Rotor Blade	22	1.0	New Tail Rotor Blades with Advanced Airfoil and Stronger Hub
	Total Weight		1.5	
2	Body	521	6.0	Add Strut to Tail Rotor Fin. New Horizontal Stabilizer (Shaft Pairing)
	Main Rotor Transmission	307	4.0	Change Gear Ratio in Main Rotor Transmission
	Tail Rotor Transmission	15	0.5	Change Gear Ratio in Tail Rotor Transmission
	Tail Rotor Drive Shaft	23	2.0	Increase Strength of Shaft End Fittings
	Tail Rotor	0	5.0	Add Shaft from Tail Rotor Transmission to Tail Rotor Blades
	Main Rotor Blade	264	11.0	Increase Chord of Main Rotor Blade
	Tail Rotor Blade	22	1.0	New Tail Rotor Blades with Advanced Airfoil and Stronger Hub
	Main Rotor Hub	198	12.0	Increase Strength of Main Rotor Hub
	Rotor Transmission	157	15.0	Add Acoustical Treatment to Main Rotor Transmission
	Total Weight		56.5	

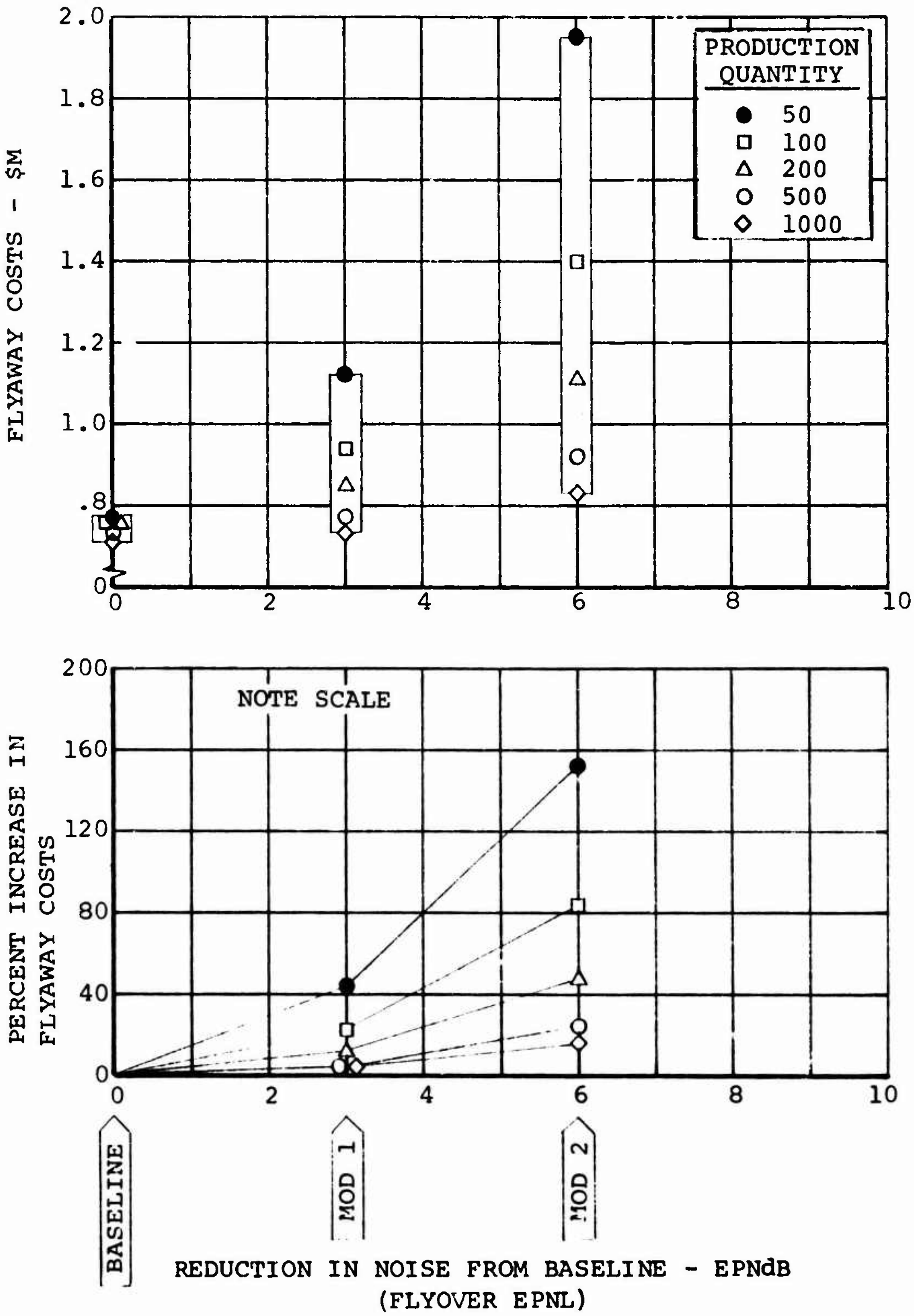
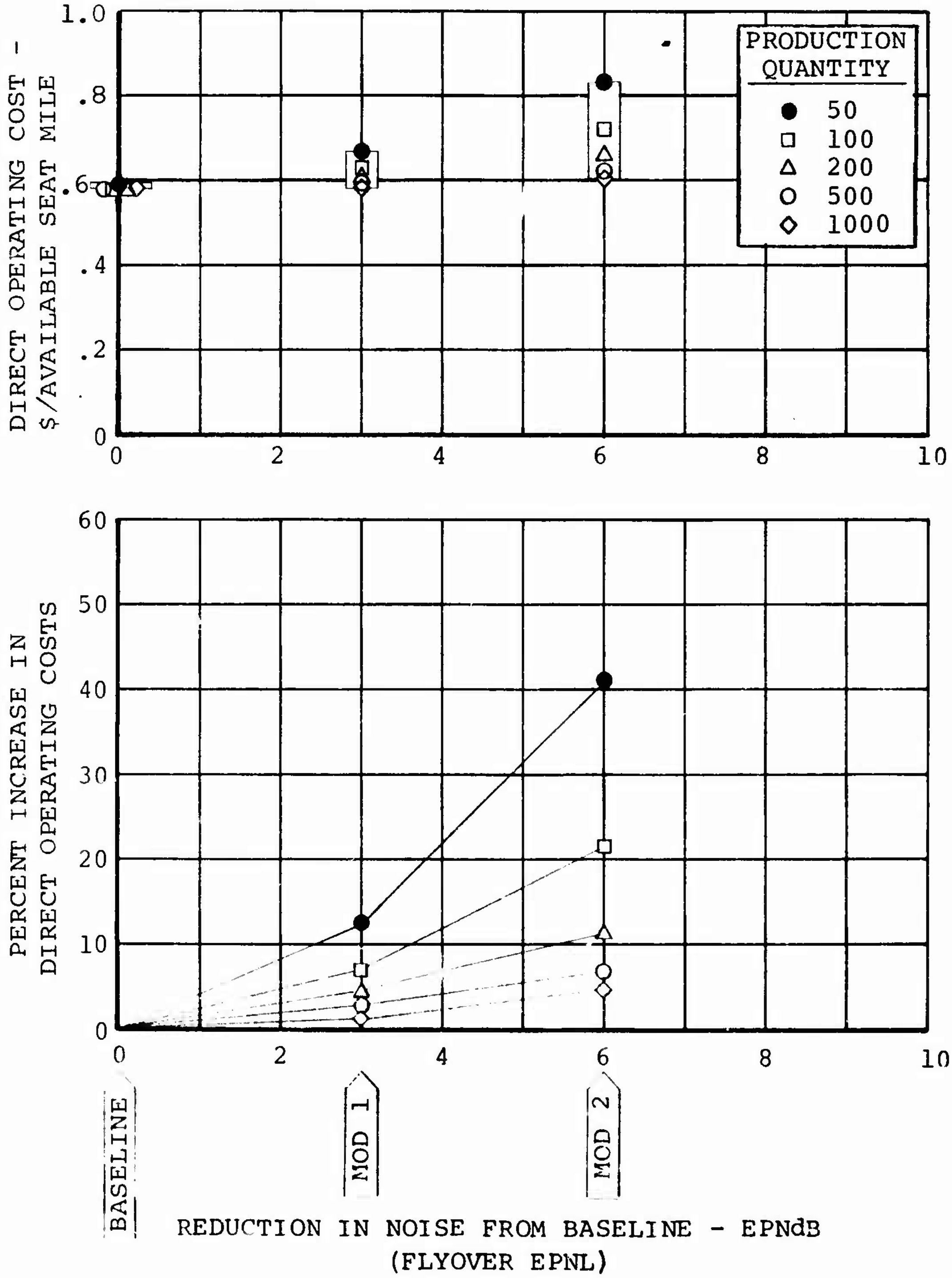


Figure 20. Effect of Configuration Changes on Flyaway Cost, BO-105



Fiugre 21. Effect of Configuration Changes on Direct Operating Cost, BO-105

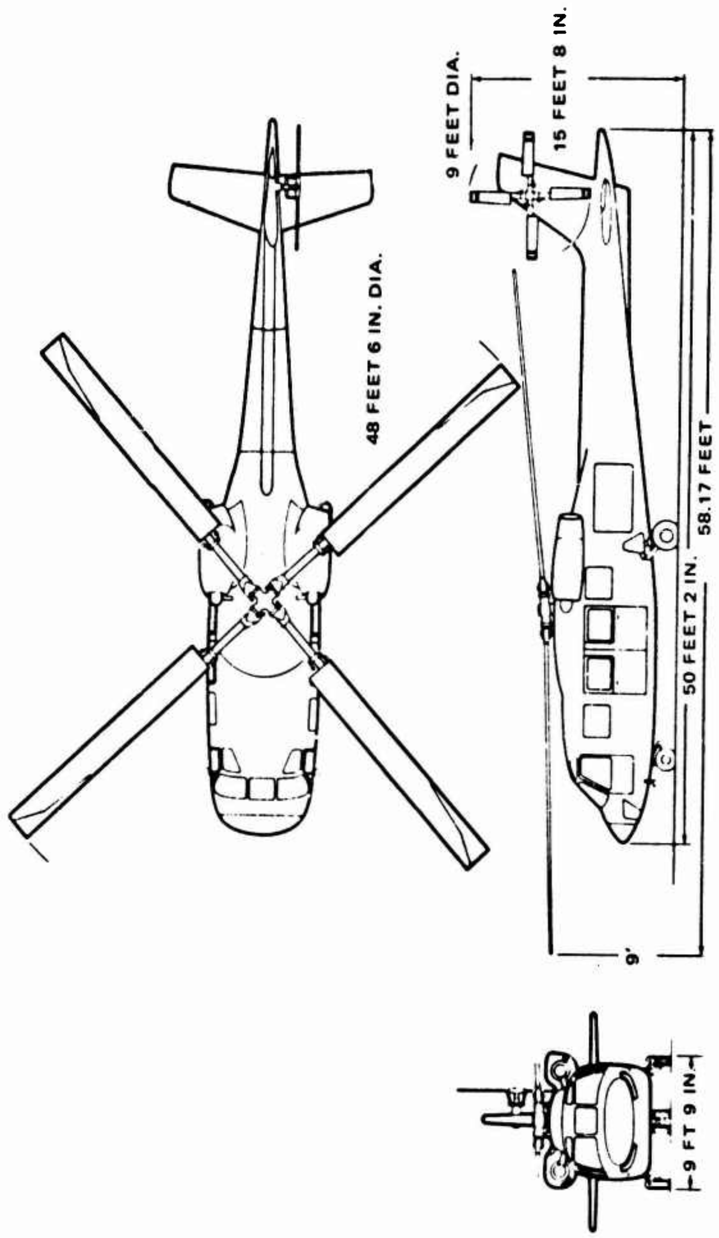


FIGURE 22. MODEL 179 GENERAL ARRANGEMENT

Table 4. Model 179 Characteristics

Maximum Takeoff Gross Weight (Normal Payload)	17,400 Pounds
Operating Weight Empty	9,574 Pounds
Engines	(2) GE CT 7-1
Main Rotor	
Type	Hingeless
Number of Blades	4
Radius	24.5 Feet
Chord	23.0 Inches
Tail Rotor	
Type	Flex Strap
Number of Blades	4
Radius	5.08 Feet
Chord	0.73 Inches
Normal Cruise Speed	137 Knots
Main Rotor RPM	286 (97%)
Tail Rotor RPM	1296
Accommodations	19 Passengers

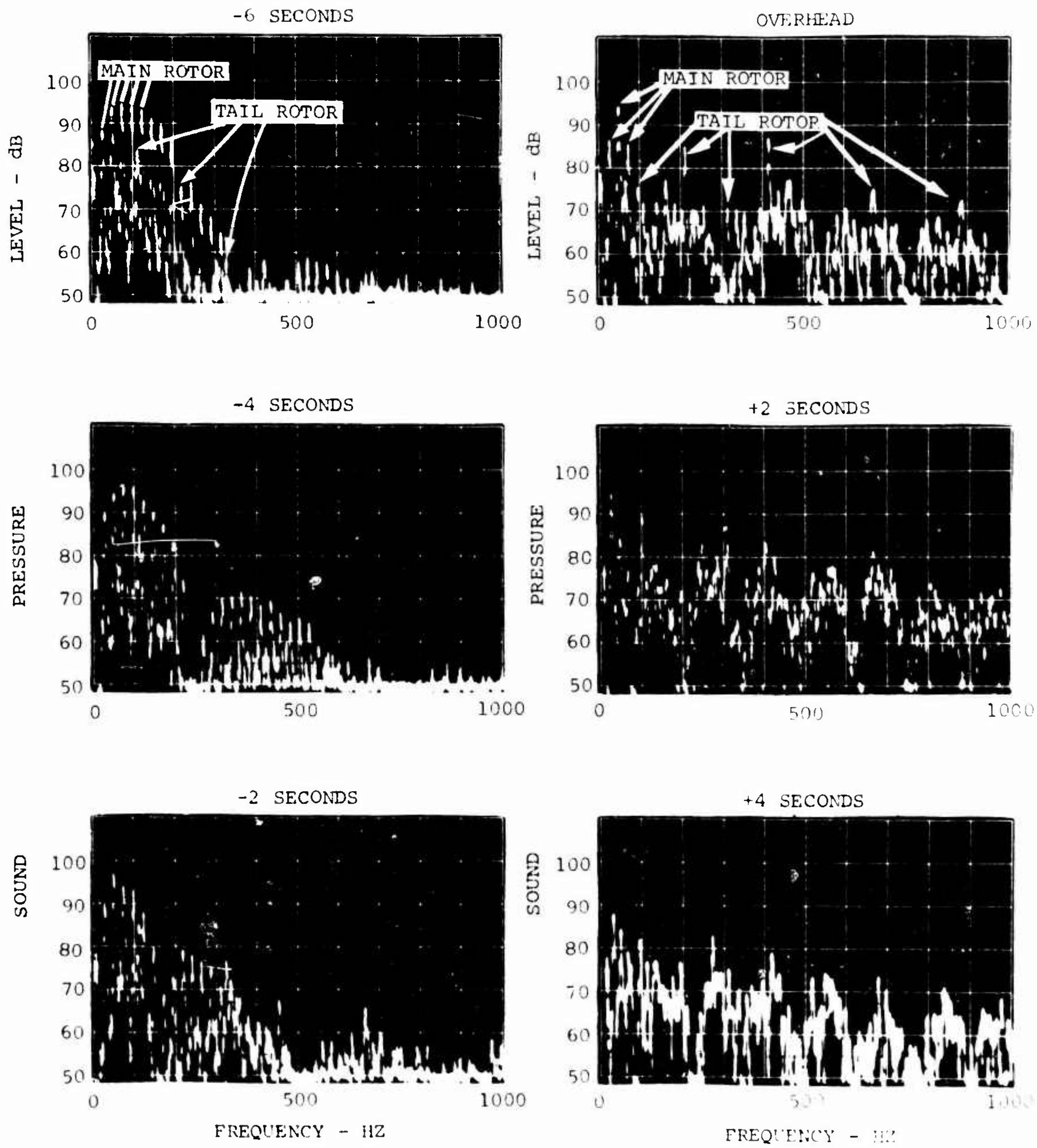


Figure 23. Narrowband Spectra of Model 179 Flyover Noise

-6 SECONDS

-4 SECONDS

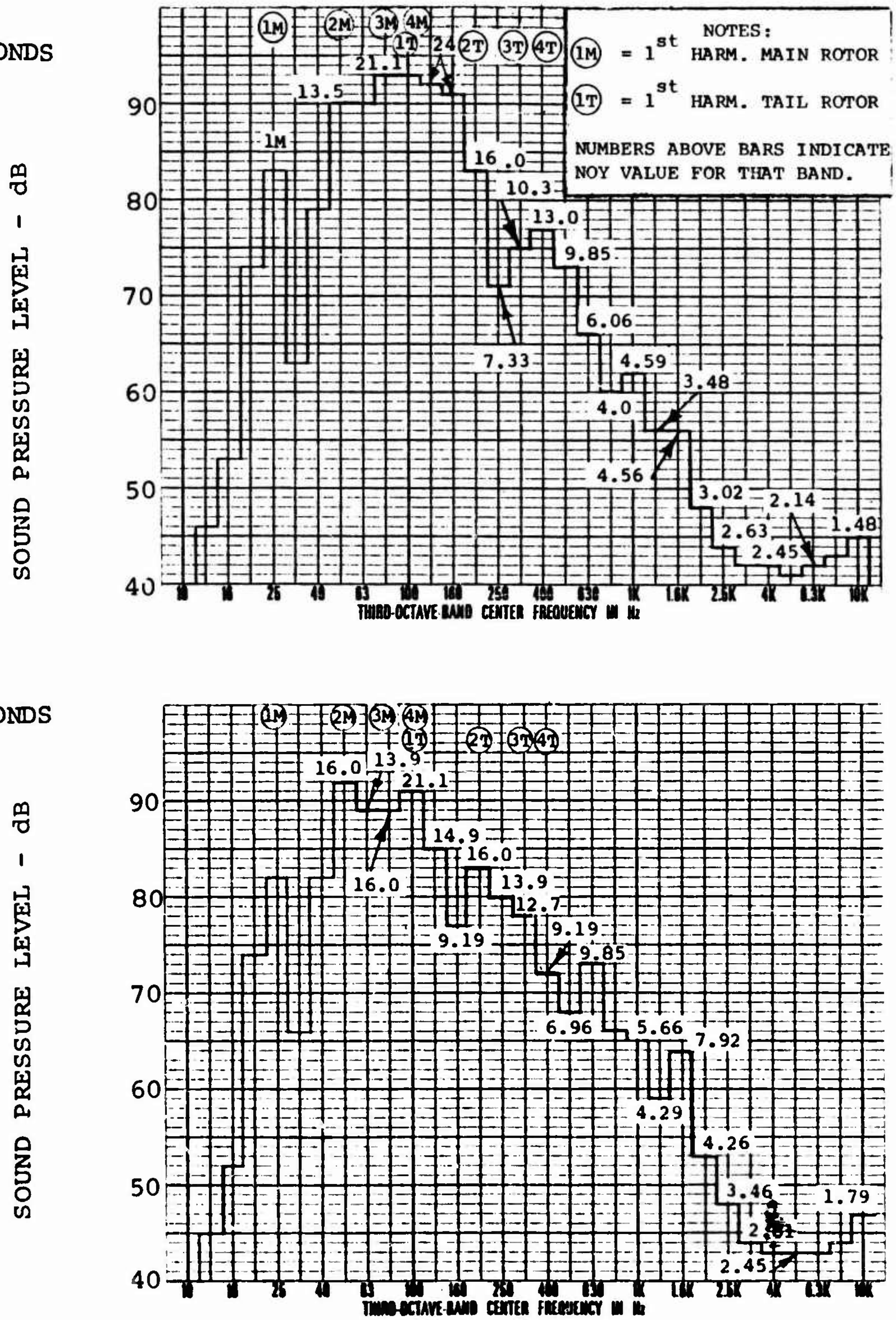
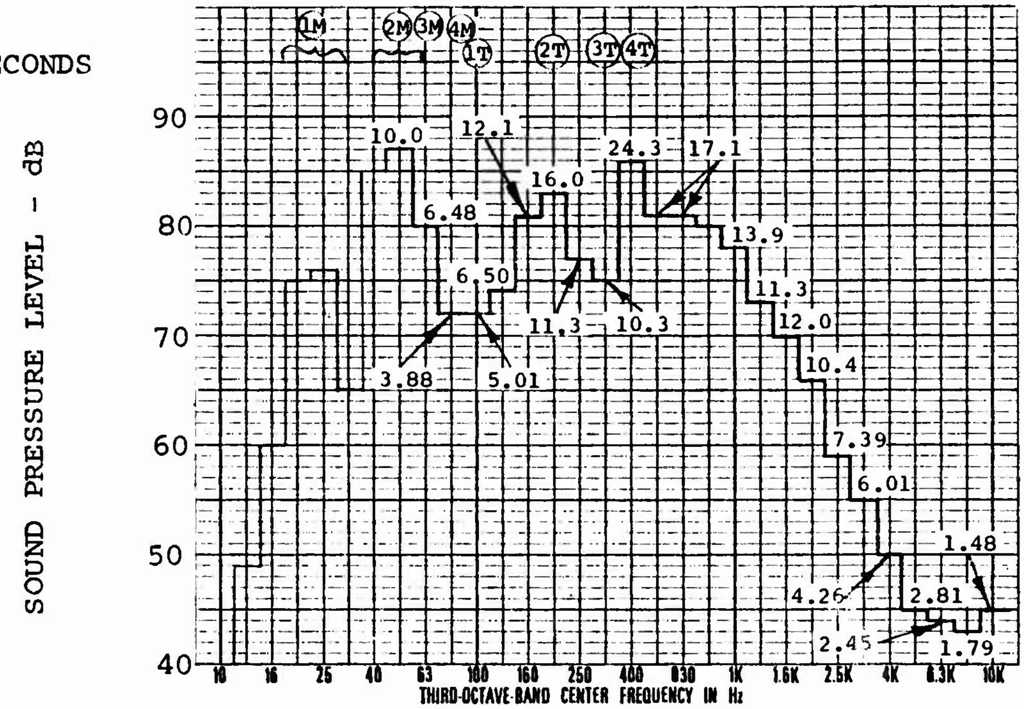


FIGURE 24. 1/3 OCTAVE BAND SPECTRA OF MODEL 179
FLYOVER NOISE (SHEET 1)

-2 SECONDS



OVERHEAD

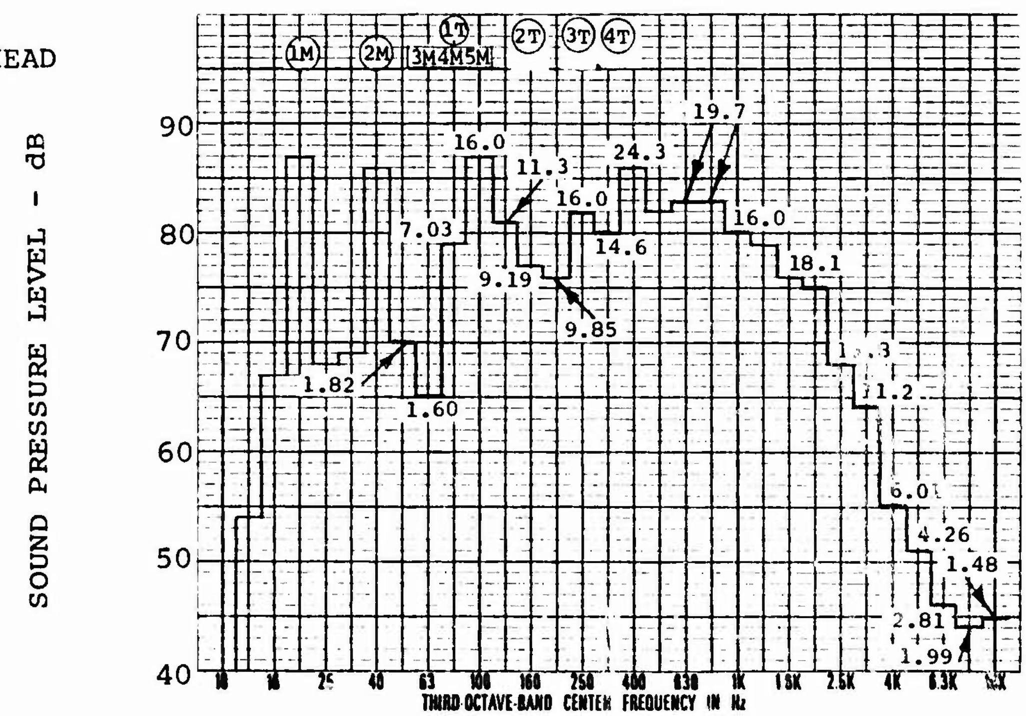
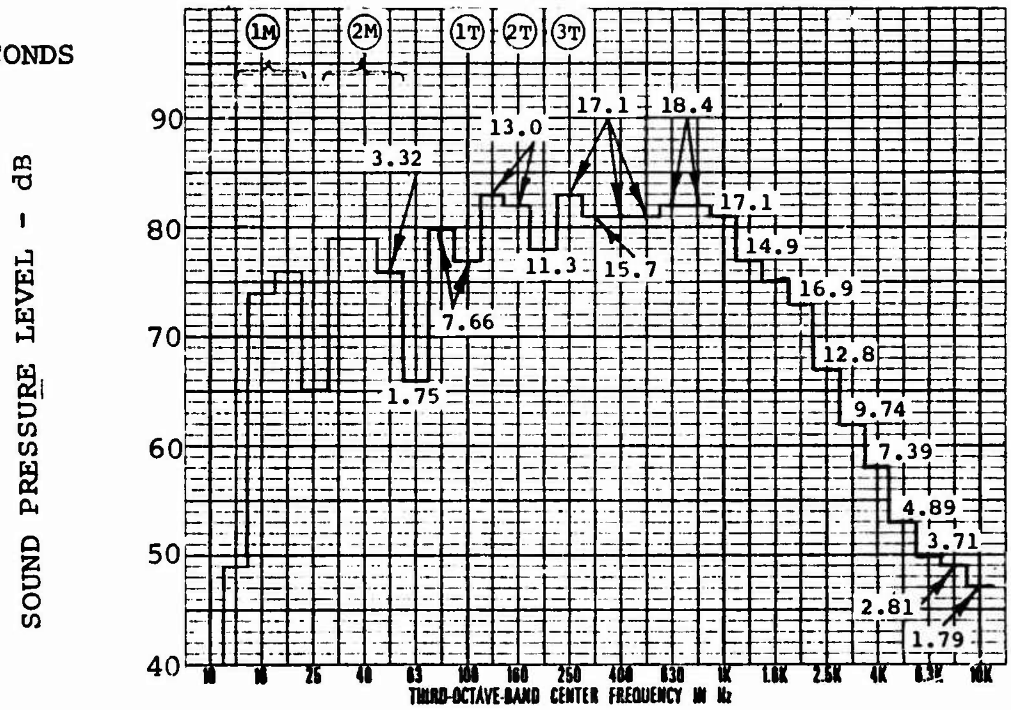


FIGURE 24. 1/3 OCTAVE BAND SPECTRA OF MODEL 179
FLYOVER NOISE (SHEET 2)

+2 SECONDS



+4 SECONDS

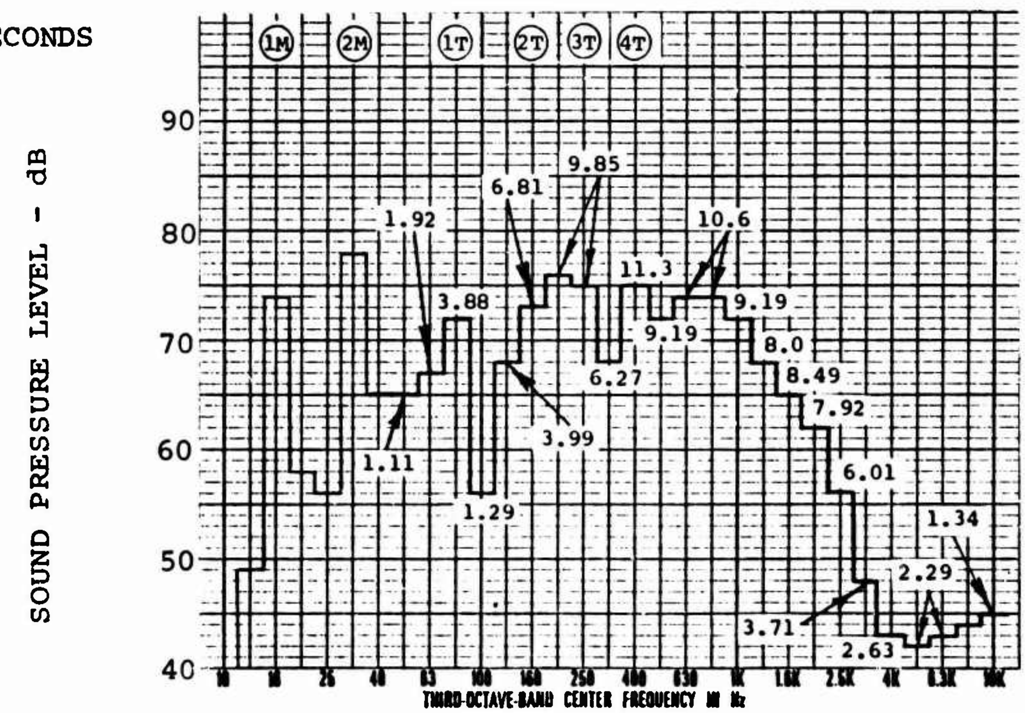


FIGURE 24. 1/3 OCTAVE BAND SPECTRA OF MODEL 179 FLYOVER NOISE (SHEET 3)

Frequencies which displayed a major contribution to the Perceived Noise Level were noted and the amplitudes converted to 1/3 octave Noy weightings. Levels of PNL and EPNL were then reduced in increments by an iterative process whereby the sound pressure in the appropriate frequency band was reduced as required, based on estimates determined by the analytical procedures of References 3-6, in a process similar to that described for the BO-105.

Configuration Changes - Tradeoff Variables

The cost/benefit study was conducted for the baseline Model 179 and three modifications to the baseline helicopter. Changes to the helicopter from the baseline are identified in Table 5 and illustrated in Figure 25.

Modification 1 to the baseline configuration of the Model 179 achieves a 3 EPNdB reduction in flyover noise by reducing main rotor tip speed from 734 to 713 ft/sec and tail rotor tip speed from 690 to 668 ft/sec. In addition, the tail rotor blade has been modified by increasing the twist. Increasing twist has been reported in Reference 9 as an effective method for reducing tail rotor noise by moving the position of spanwise lift to a more inboard location, thus reducing velocity at the station of maximum thrust. The redesigned tail rotor blade also includes a revised tip shape, such as an elliptical or swept configuration that would be incorporated into the design along with the other blade modifications. The lower tip speed of the tail rotor requires a new gear ratio in the tail rotor gearbox and a strengthened hub.

The reduction in main rotor RPM requires a review of all drive train components with regard to accommodating higher torque levels in the drive system. While not all components require strengthening, some modification to the dynamic system, such as strengthened main rotor hub, would be likely.

Reductions in tip speed of the main rotor below that defined for Modification 1 require an increase in rotor solidity in order to maintain baseline helicopter mission profile and performance. In addition to the changes noted above for Modification 1, a wider chord blade is added to the main rotor. Modification 2 illustrates the 'step' changes to the rotor which are required as rotor speed is reduced. Figure 26 shows that below a tip speed of 718 ft/sec, maneuverability criteria require that additional rotor area be provided. For purposes of the study, a 25 percent increase in rotor solidity was incorporated into the Modification 2 change. This could take the form of either a fifth blade on the rotor or a 25 percent increase in blade area of the existing blades. An additional blade on the main rotor hub would require a small fuselage extension. This arises due to a rearrangement of the hub and fittings resulting from insufficient area in which to lay out a fifth blade on the existing hub. Thus, the addition of a fifth blade also results in an increase in rotor radius and an extension to the Model 179 fuselage. For this reason, primarily, the approach with the least impact on the aircraft was a wider chord blade. The amount of increase in rotor solidity acutally selected for a particular design would be determined by the desired noise level. Redesign costs would be similar regardless of the specific increase in chord. Material and recurring costs vary somewhat with blade weight, but these costs are not considered significant in comparison with the complexity of extending the fuselage length. A design modification which incorporates a 25 percent increase in rotor solidity allows some growth to the rotor and is a realistic increase from a manufacturer's viewpoint. A revised tip shape would also be included in the new blade design to reduce tip generated noise. An elliptical shape tip has demonstrated a reduction of 2 dB on the Model 179 rotor.

Table 5. Model 179 Configuration Changes

	Baseline	Modification 1	Modification 2	Modification 3
MAIN ROTOR				
V_t (ft/sec)	734	718	715	694
RPM	286	280	278	270
No. of Blades	4	4	4	4
Airfoil	VR-7,8,9	VR-7,8,9	VR-7,8,9	VR-7,8,9
Chord	23.0 In.	23.0 In.	24.9 In.	24.9 In.
TAIL ROTOR				
V_t (ft/sec)	690	668	665	654
RPM	1296	1256	1250	1229
No. of Blades	4	4	4	4
Airfoil	VR-7,8	VR-7,8 Increased twist, modified tip.	VR-7,8 Increased twist, modified tip.	VR-7,8 Increased twist, modified tip.
Chord	0.73 ft	0.73 ft	0.80 ft	0.80 ft
Flyover EPNL	98	95	94.5	91
Dynamic System	Basic	New T/R Gearbox	New T/R Gearbox	New T/R Gearbox
Airframe	Basic	Basic	Basic	Offset Tail Rotor
Powerplant	GE CT 7-1	GE CT 7-1	GE CT 7-1	GE CT 7-1
Weight Change (lb)	-	+52 lb	+111 lb	+191 lb

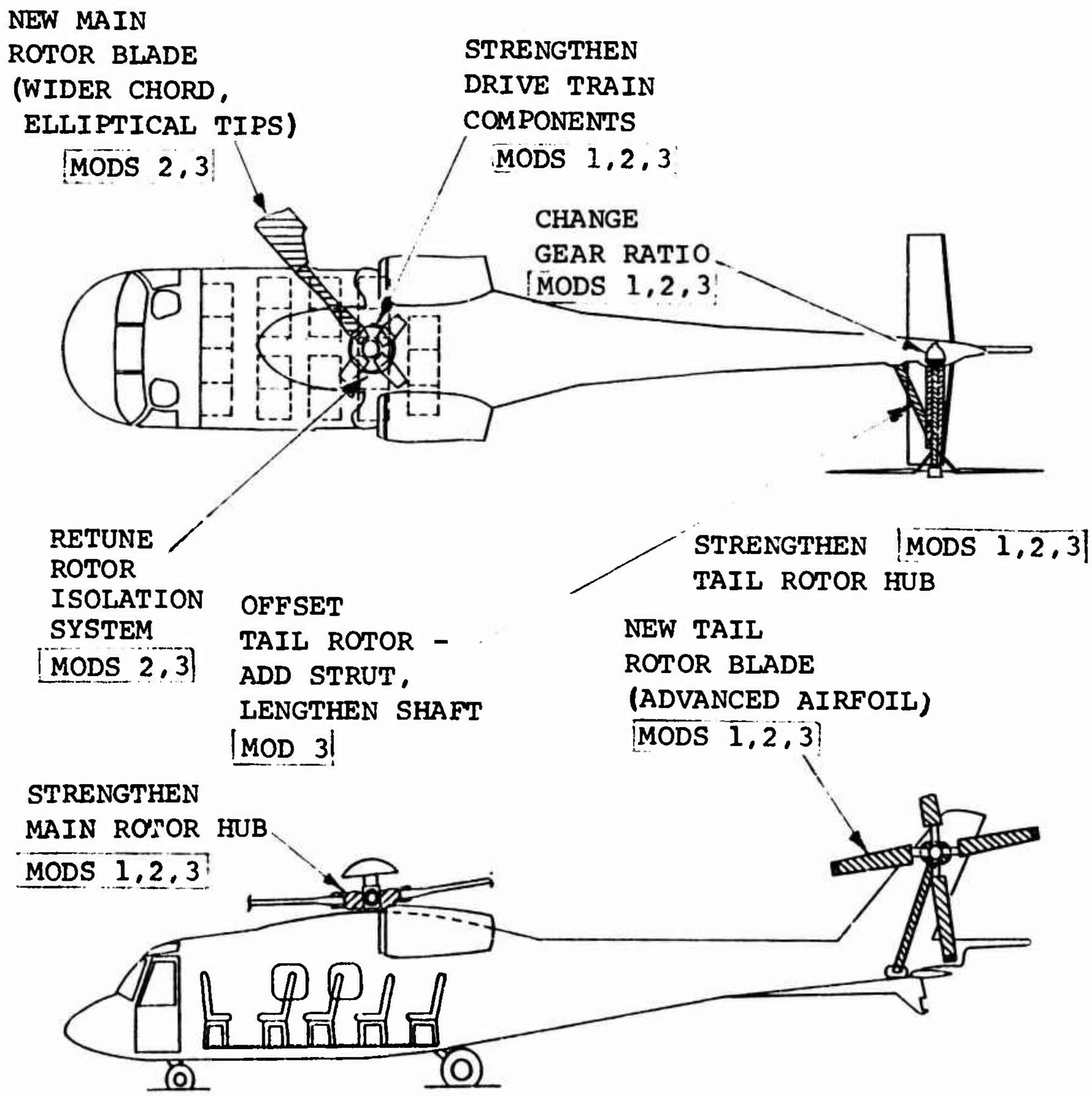


Figure 25. Configuration Changes to Model 179

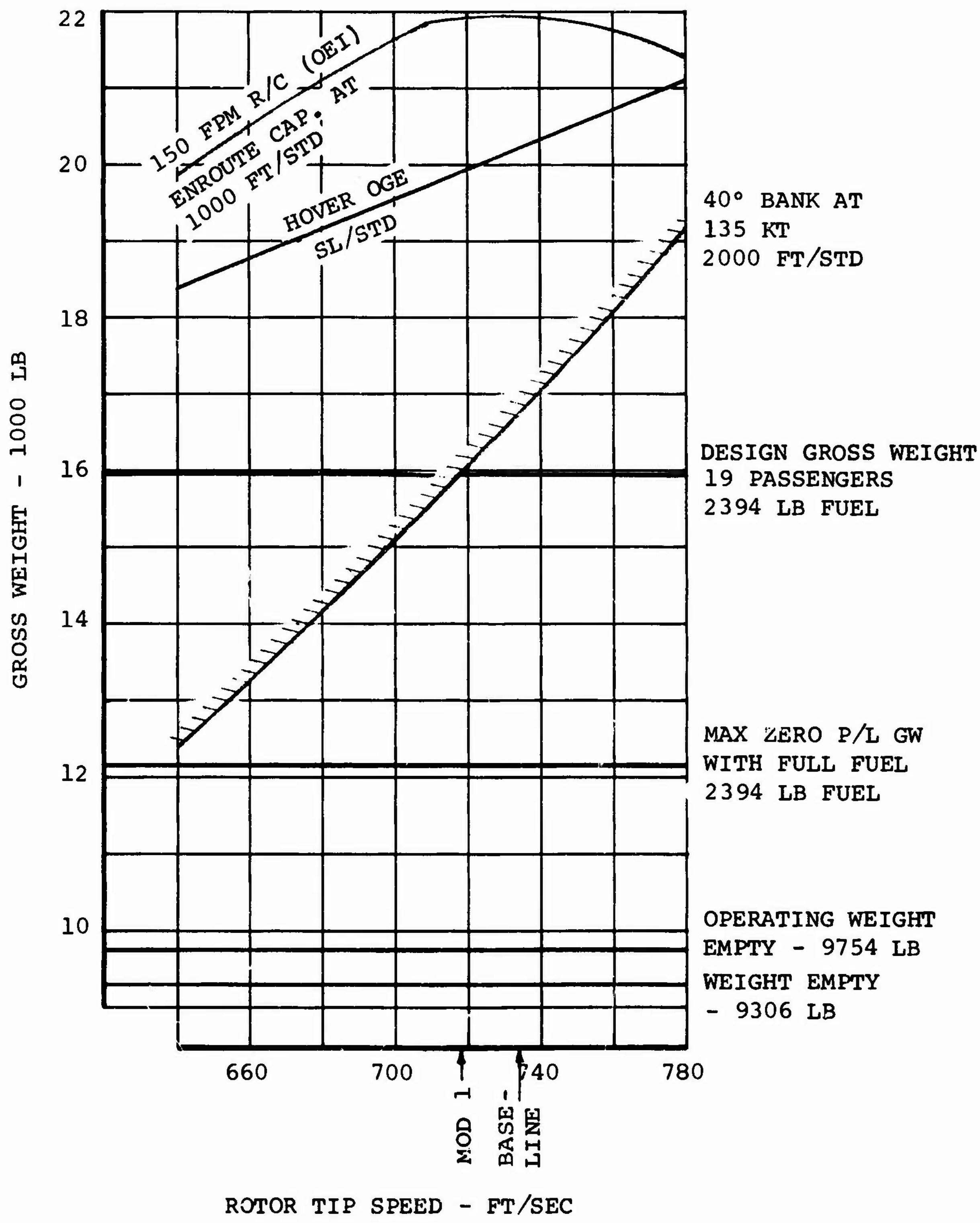


FIGURE 26. MODEL 179 PERFORMANCE LIMITATIONS
SHEET 1 (EXISTING ROTOR)

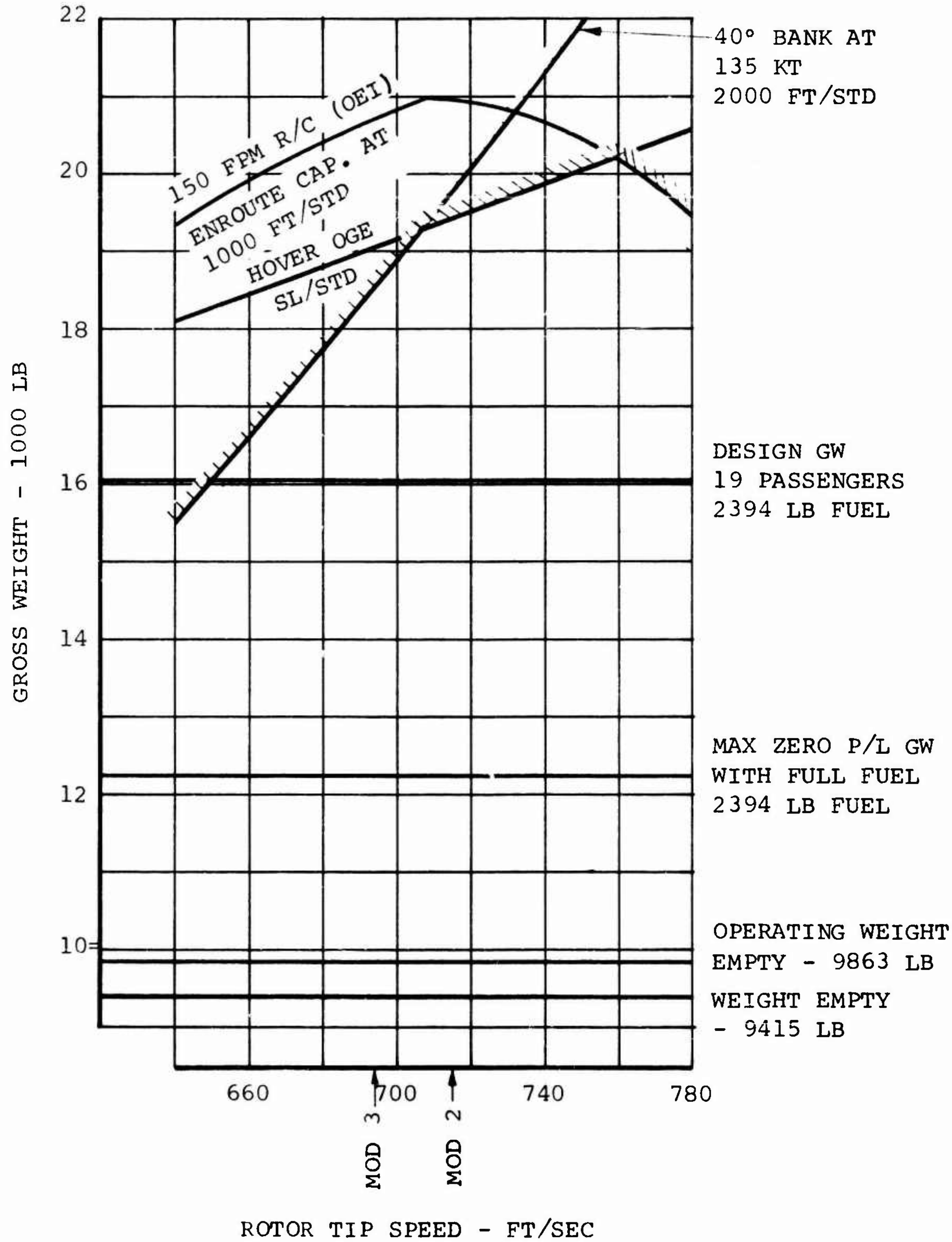


FIGURE 26. MODEL 179 PERFORMANCE LIMITATIONS
SHEET 2 (INCREASED SOLIDITY ROTOR)

The rotor isolation system was reworked for Modification 2 in order to tune the isolator to the new rotor frequency. This involves a small increase in weight in that the rotor frequency was reduced from 286 to 278 RPM and trend curves indicate higher isolator weights at lower frequencies.

The tail rotor blade was redesigned for Modification 2 to increase solidity and provide sufficient thrust for control response at the lower tail rotor speeds. The twist schedule is similar to that described for Modification 1. Tip speed has been reduced to 665 ft/sec for the tail rotor.

Modification 3 of the Model 179 reduces flyover noise to 91.5 EPNdB by further reductions in main and tail rotor tip speeds and a lateral offset of the tail rotor to substantially reduce interference between the wake of the main rotor and the blades of the tail rotor. Main rotor tip speed is reduced to 694 ft/sec and tail rotor tip speed to 654 ft/sec. The tail rotor offset of 3.77 feet represents a clearance between main and tail rotor tip path planes of $\delta = 0.12$ as described in Section IV-3. The modification to the tail rotor includes an extension of the shaft between the tail rotor gear box and hub, a tubular enclosure for this shaft, a bearing at the hub and a strut between the tail boom and the outboard section of the extended hub. The effect of this design change on other aircraft systems and flying qualities is discussed in the following section. In order to maintain performance and flying qualities, the solidity of both the main and tail rotors has been increased as discussed for the Modification 2 configuration.

Other modifications to the helicopter required as a result of the lower tip speeds include strengthening of certain drive train components such as rotor transmission gears, rotor shaft, tail rotor deive shaft and main and tail rotor hubs. In addition, the Rotor Vibration Isolation System is retuned to a lower rotor frequency.

Impact of Design Changes to Performance, Weight and Cost

Performance

The effect of tip speed reduction on cruise speed and range is shown in Figure 27. The percent reduction in cruise speed relative to the baseline aircraft ranges from -0.9 percent for Modification 1 to -1.6 percent for Modification 3.

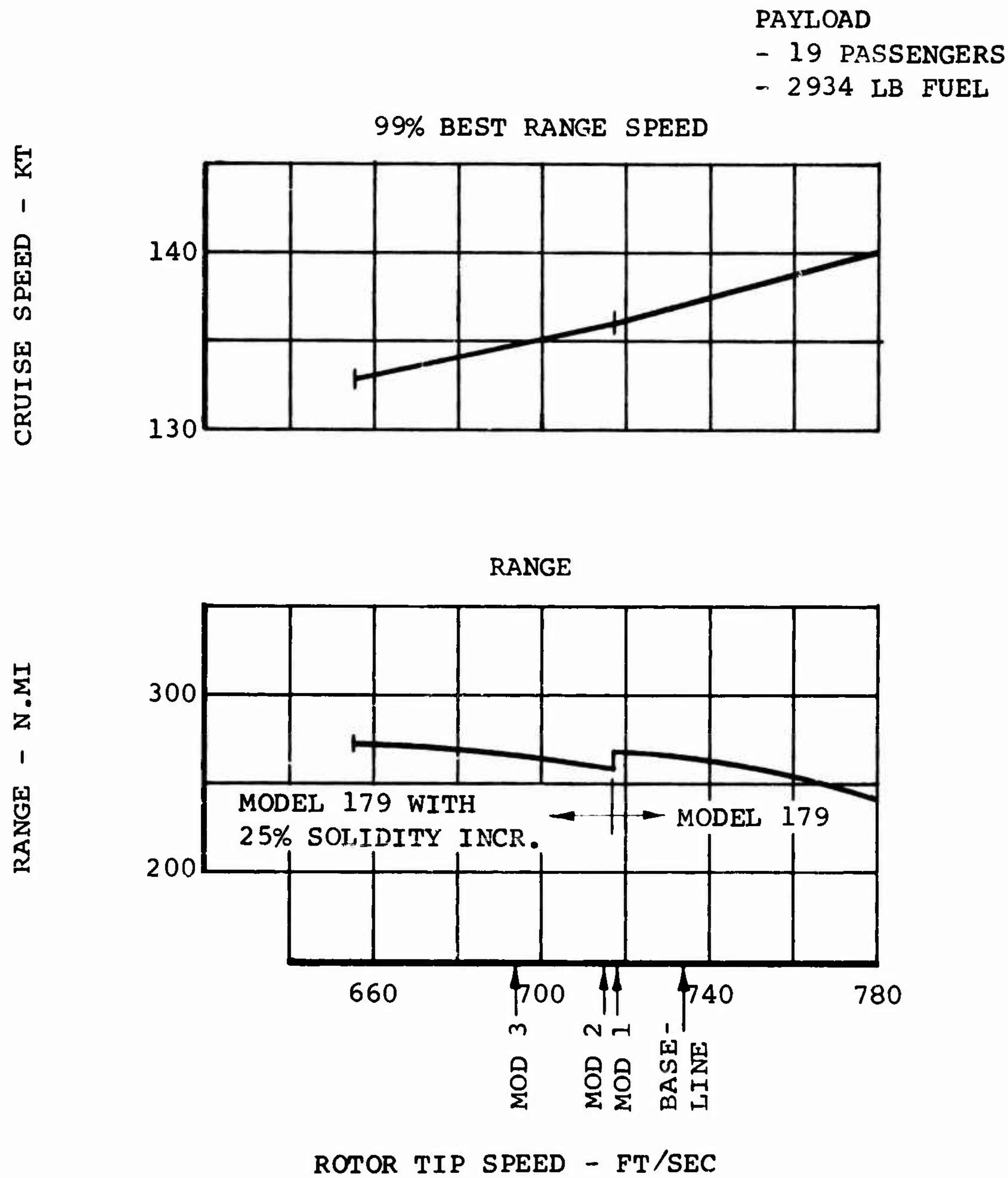
Weight

The baseline weight empty for the Model 179 is 9754 pounds. This increases by 52 pounds for Modification 1, 111 pounds for Modification 2 and 191 pounds for Modification 3. The effect of helicopter modification on system weights is presented in Table 6.

The increase in weight empty for each configuration is 0.5 percent for Modification 1, 1.1 percent for Modification 2 and 2.0 percent for Modification 3.

Costs

For the cost study, the Model 179 was considered both as a derivative aircraft which was several years into the production cycle, and a 'new' aircraft that was early in the design stage without benefit of a current production base.



FLIGHT PROFILE:

1. WARM UP 2 MIN. AT MAX CONTINUOUS POWER.
2. TAKE OFF AND CRUISE OUTBOUND AT 99% BEST RANGE SPEED.
3. LAND WITH 45 MIN. FUEL RESERVE AT 99% BEST RANGE SPEED.

FIGURE 27. EFFECT OF ROTOR SPEED REDUCTION ON MODEL 179 CRUISE SPEED AND RANGE

Table 6. Impact of Design Changes on Model 179 Weights

A/C Mod	Component	Baseline Weight (Lb)	Additional Weight (Lb)	Description of Change
1	Main Rotor Transmission	1300	19	Strengthen Drive Train Components
	Tail Rotor Transmission	67	3	Change Gear Ratio-Strengthen Hub
	Tail Rotor Blade	38	0	New Twist Schedule
	Main Rotor Hub	590	30	Strengthen Hub
	Total Weight		52	
2	Main Rotor Transmission	1300	24	Strengthen Drive Train Components
	Tail Rotor Transmission	67	3	Change Gear Ratio-Strengthen Hub
	Rotor Isolation System	228	4	Retune Isolation System
	Tail Rotor Blade	38	2	New Twist Schedule
	Main Rotor Blade	1256	45	Wider Chord Blade
	Main Rotor Hub	590	33	Strengthen Hub
	Total Weight		111	
3	Tail Rotor	1546	15	Lateral Offset of Tail Rotor
	Main Rotor Transmission	1300	51	Strengthen Drive Train Components
	Tail Rotor Transmission	67	28	Change Gear Ratio-Strengthen Hub
	Rotor Isolation	228	4	Retune Isolation System
	Tail Rotor Blade	38	2	New Twist Schedule
	Main Rotor Blade	1256	45	Wider Chord Blade
	Main Rotor Hub	590	46	Strengthen Hub
	Total Weight		191	

As an 'in-production' helicopter, initial development costs would be spread over a large production base which might be 1000 or more units, and only the costs resulting from the noise reduction modifications would be absorbed into the price of the aircraft. Thus Model 179 total nonrecurring costs for an 'in-production' helicopter were spread over a quantity in excess of 1100 units. This number arises from estimated production quantities of UTTAS and LAMPS military helicopters and represents the largest reasonable production base that might be assumed for a large military contract.

Considered as a 'new' helicopter, all nonrecurring costs were spread over a fixed percentage of the expected production quantities. Production quantities would be expected to be in the vicinity of 100-500 units, although a range extending from 50-1000 aircraft was evaluated. Figures 28 and 29 present the factory flyaway and seat mile costs (DOC) for the Model 179 'new' model baseline helicopter and three modified configurations. The apparent discontinuity in costs at about 3 dB noise reduction is due to the introduction of a new rotor blade.

Although the absolute flyaway and operating costs are largely a function of production quantity, it is of interest to note the relatively small change in cost with each modification compared with the 'in-production' version. This is because the design had been frozen prior to the test and development cycle regardless of the modification, and tooling was configured to that design. The direct operating costs for a 100 unit production quantity, for example, varied only two cents from the baseline aircraft displaying a 98.5 EPNdB flyover signature, to the modification which results in a 91.5 EPNdB level (38 cents/mile to 40 cents/mile).

This represents an increase of 5 percent for a 7 EPNdB flyover noise reduction compared with a range of 8 percent to 40 percent for an 'in-production' model.

CH-47C

Vehicle Description

The baseline CH-47 helicopter for this study is the CH-47C 'Chinook' (Figure 30), a tandem rotor medium-lift helicopter powered by twin Lycoming T55-L-11 gas turbine engines rated at 3759 horsepower each. It has a cruising speed of 130 knots and a maximum gross weight of 50,000 pounds. It was introduced into service with the U.S. Army in 1968. Further development to the rotor blades, transmissions and equipment are being incorporated into the CH-47D which is currently undergoing flight testing. A summary of CH-47C configuration characteristics is presented in Table 7.

The baseline study aircraft has been configured as a 44 passenger civil helicopter with commercial engines (Lycoming AL 5512's). It has a takeoff gross weight of 40,654 pounds and an average cruise speed of 139 knots at 245 rotor RPM. This baseline vehicle maintains the flyover noise signature of the CH-47C, but is typical of a 'derivative' commercial helicopter that stems from a military history. The interior arrangement is identical with the Boeing Model 234 Commercial Chinook, currently under development, and the weight of this installation was available from that program. However, it is unlike the Model 234 in several ways, primarily in that the Model 234 includes advanced

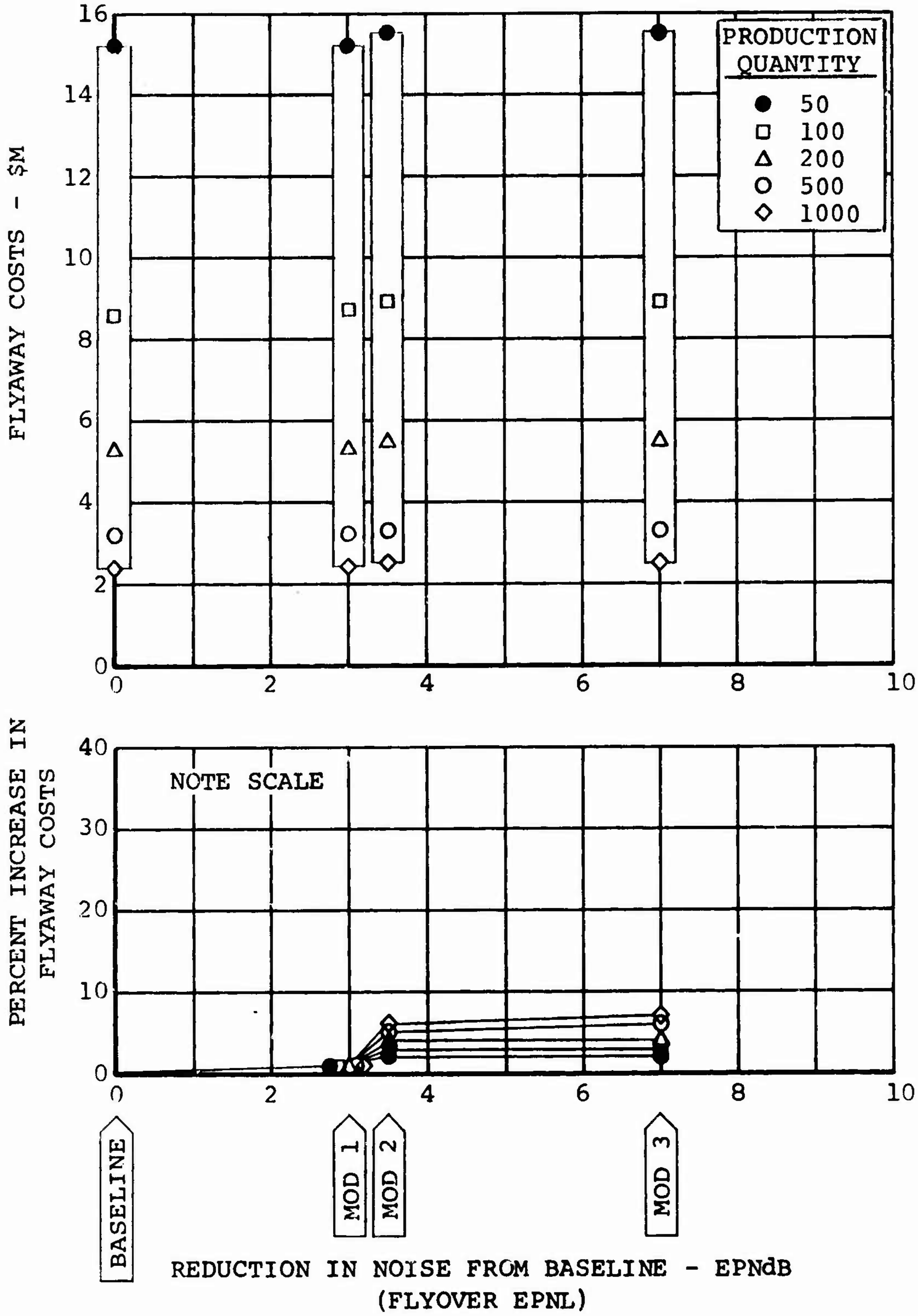


Figure 28. Effect of Configuration Changes on Flyaway Cost, Model 179 "New" Helicopter (Sheet 1)

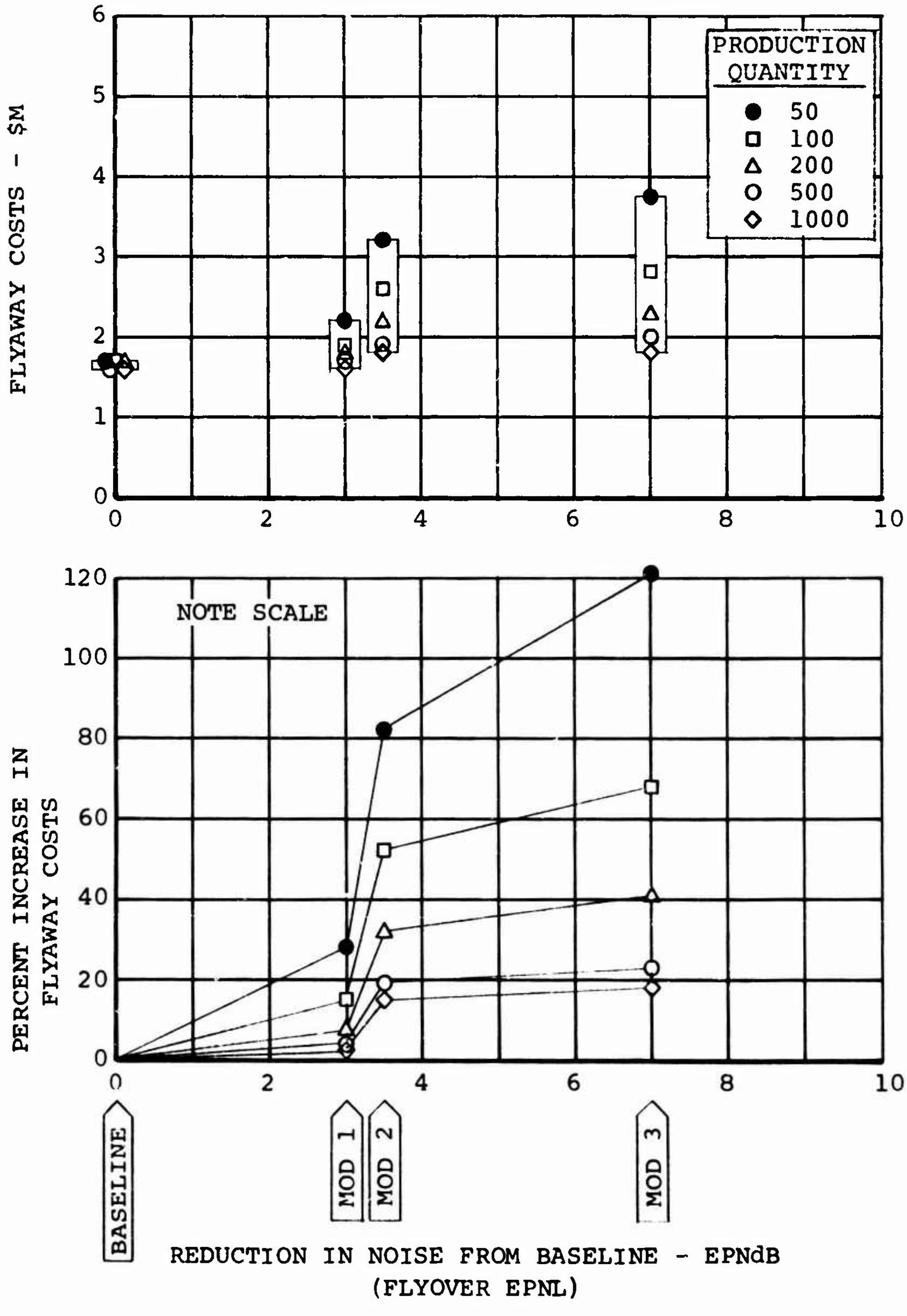


Figure 28. Effect of Configuration Changes on Flyaway Cost, Model 179 'In-Production' (Sheet 2)

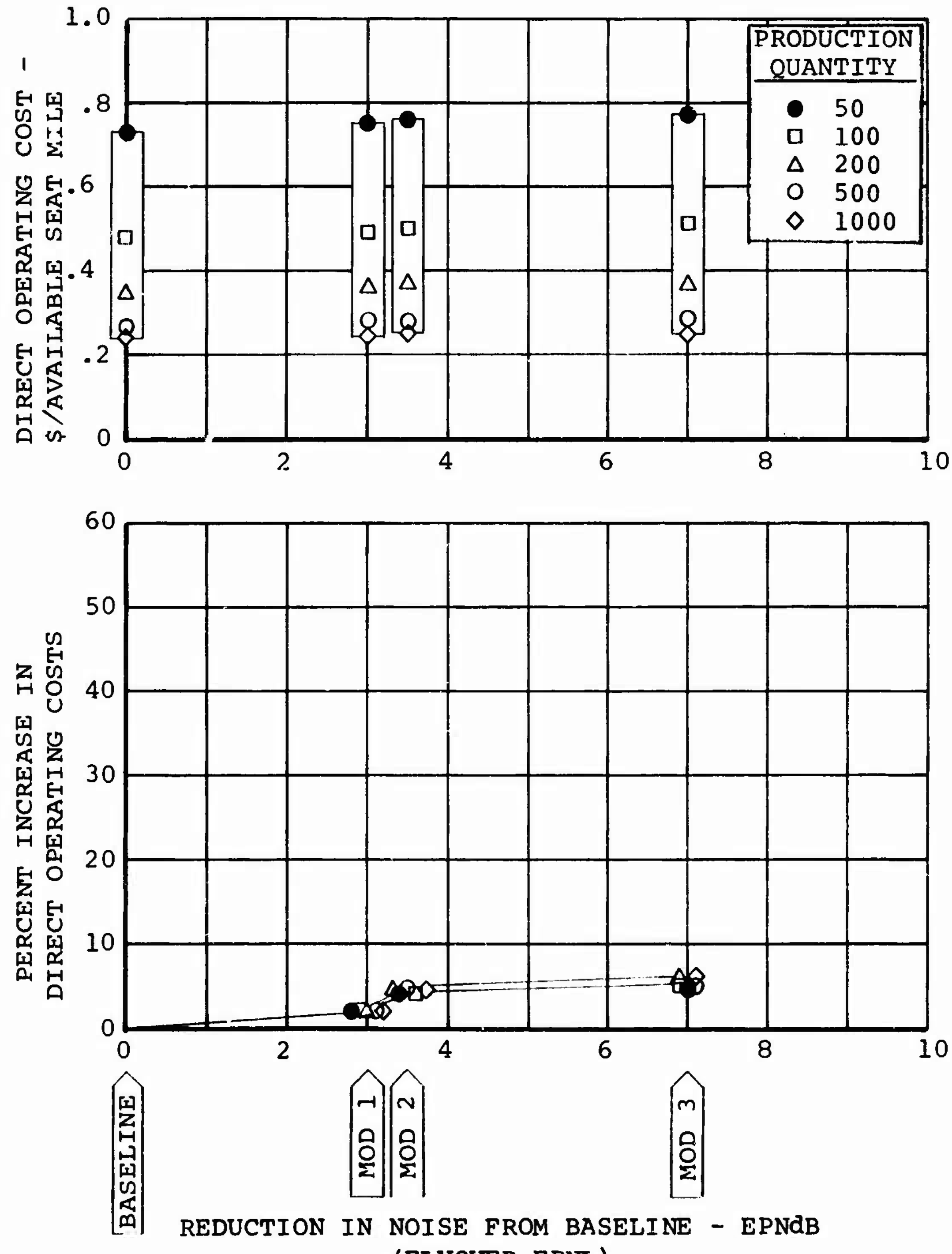


Figure 29. Effect of Configuration Changes on Direct Operating Cost, Model 179 'New' Helicopter (Sheet 1)

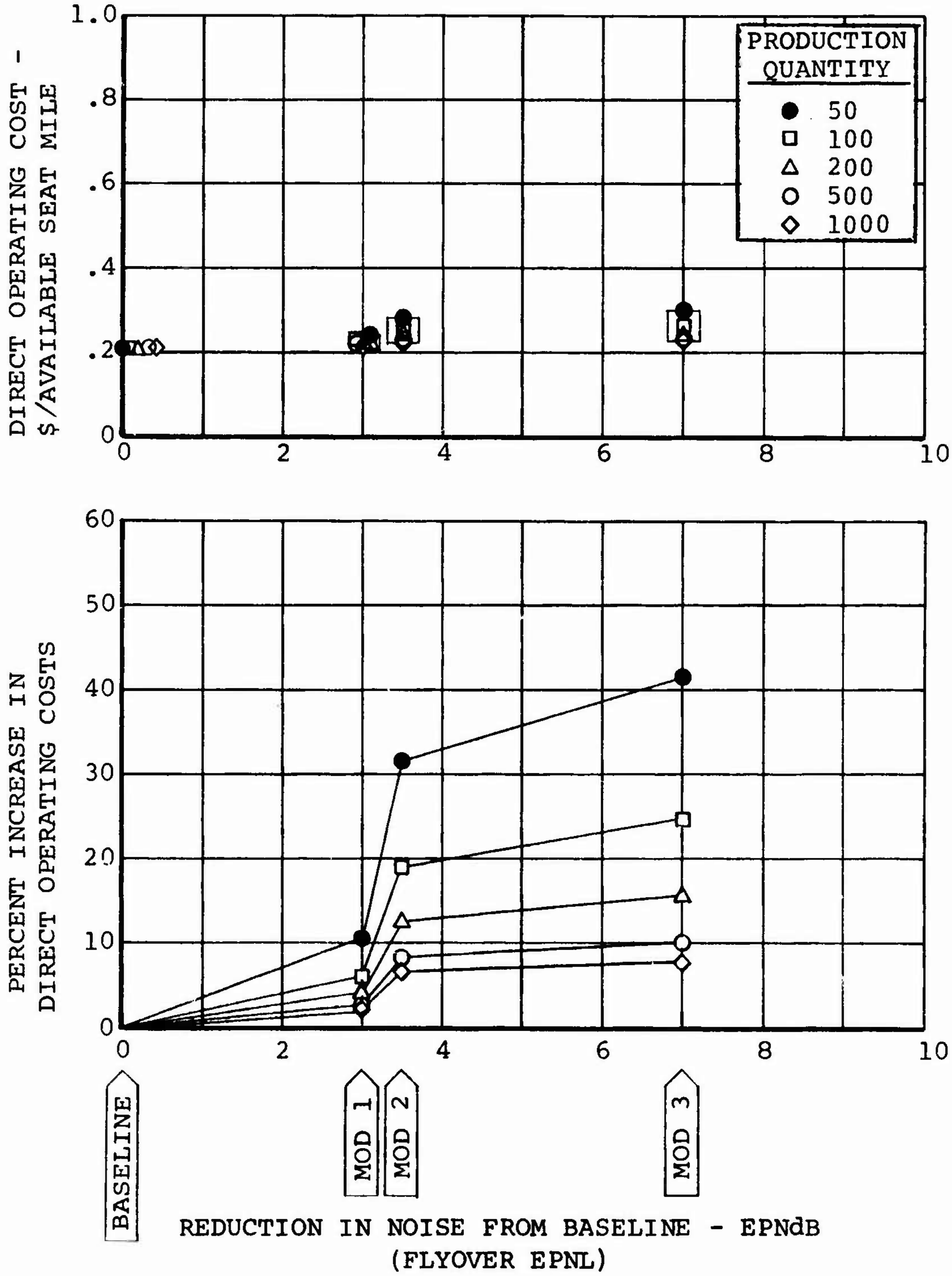


Figure 29. Effect of Configuration Changes on Direct Operating Cost, Model 179 'In-Production' Helicopter (Sheet 2)

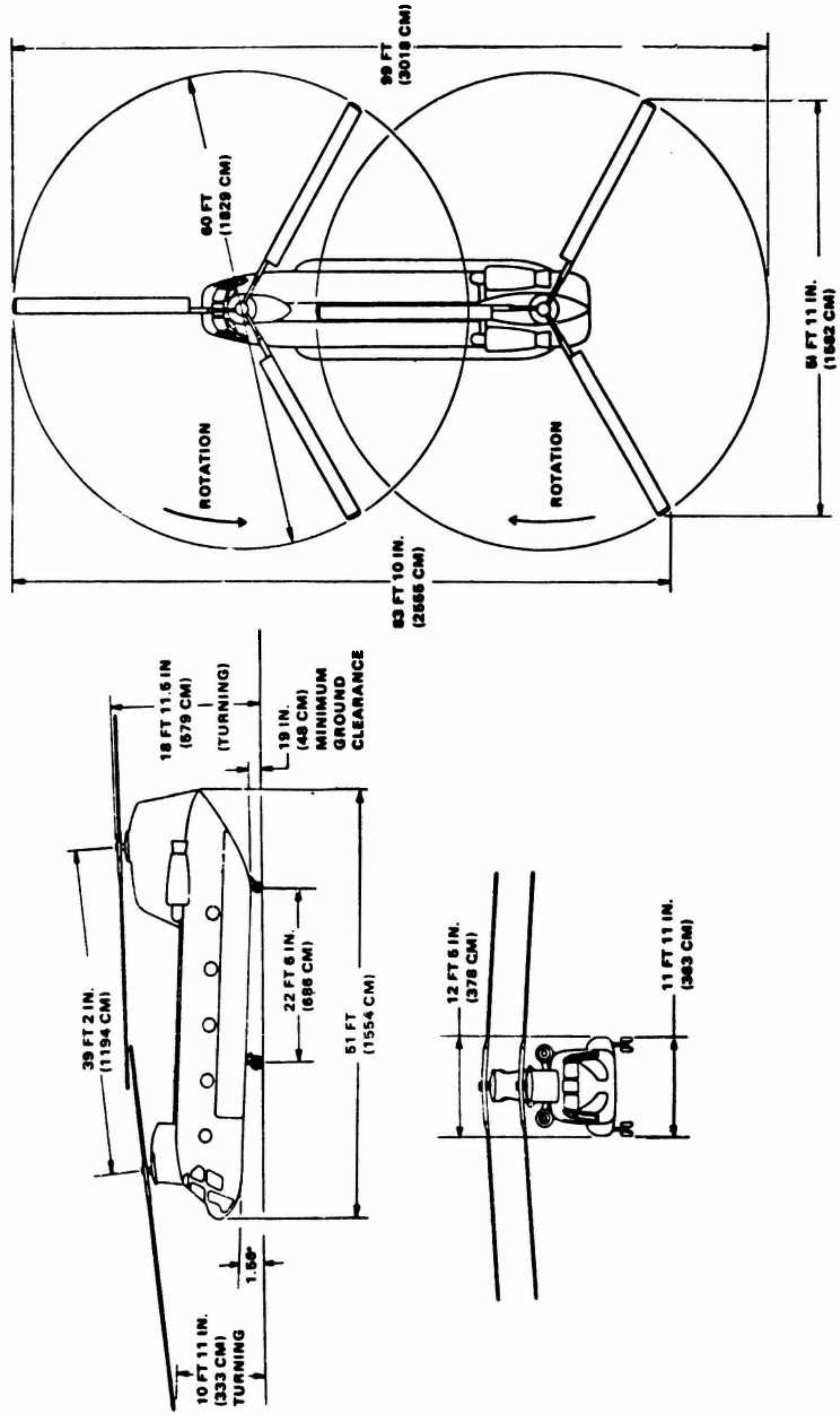


FIGURE 30. CH-47C GENERAL ARRANGEMENT

Table 7. CH-47 Characteristics

	Military CH-47C	Commercial CH-47, Study Baseline
Maximum Takeoff Gross Weight (Pound)	50,000	40,654
Operating Weight Empty (Pound)	22,320	24,862
Normal Payload (lb)	12,000	8,800
Range (Km)	200	229
Engines	(2) Lycoming T-55-11C	(2) AL 5512
Maximum Continuous Power Rating (HP)	3,000	2,975
Main Rotor		
Type	Articulated	Articulated
Number of Blades	3	3
Radius	30 Feet	30 Feet
Chord	25.25 Inches	25.25 Inches
Normal Cruise Speed (Knots)	130	139
Main Rotor Speed	245 RPM	245 RPM
Accommodations	33 Troops	44 Passengers

airfoil fiberglass blades and long range fuel tanks. However, the advanced airfoil rotor blades are included in the study as a configuration change to the baseline helicopter.

Noise Sources

Measured flyover noise data were used to identify noise sources of the baseline CH-47. The microphone was located directly under the flight path of the helicopter which was at an altitude of 500 feet and an airspeed of 141 knots (0.9 V_H). Tone corrected Perceived Noise Levels were used to determine EPNL's. Sideline microphone data were used to obtain an average flyover EPNL. The time history of the flyover at the centerline microphone is shown in Figure 31. Narrowband spectra (Figure 32) and one-third octave spectra (Figure 33) were produced to identify noise sources at selected intervals during the flyover. Source frequencies were verified from calculations of each component by the methods of Reference 3-6. Levels of PNL and EPNL were then reduced in increments by an iterative process.

The narrowband spectra of Figure 32 illustrates the dominance of the rotor as a noise source on the CH-47 during flyover. Main rotor frequency is identifiable for at least 60 harmonics of blade passage. Thickness noise on the 23010 airfoil of the CH-47C blade appears to be the major component of this source of rotor noise at 141 knots, although blade/vortex interaction also has been shown to be a contributor to flyover noise levels by earlier studies. Initial configuration changes treat only thickness noise, while major reductions in flyover noise are obtained from configurations which also affect blade/vortex interactions.

Configuration Changes - Tradeoff Variables

Four configuration changes were identified for the CH-47 with respect to the baseline aircraft, which displays an average flyover noise level of 106 EPNdB as measured by ICAO Procedures. The configuration change with the greatest impact on the exterior acoustical signature is a stretched fuselage aircraft which reduces average flyover levels to 90 EPNdB. This 16 EPNdB reduction has been documented by measurements on a modified CH-47 helicopter.

Specific configuration changes to the Chinook to achieve the incremental reduction in noise are presented in Table 8 and Figure 34. Modification 1 achieves a 7 EPNdB noise reduction by reducing rotor speed from 245 to 225 rpm. (This corresponds to a reduction in tip speed from 770 ft/sec to 707 ft/sec.) No other primary changes to the configuration were required to achieve this level. In order to maintain generator output for the electrical system, however, a gear set in the accessory drive of the aft rotor transmission is replaced to maintain generator speeds to the design range. In addition, the self-tuning cockpit absorbers were modified to bring them into the new operating range of rotor speeds. This reduced rotor speed is in a regime in which earlier models of the CH-47 have had operational experience.

A further reduction of 3 EPNdB in flyover noise (Modification 2) required a major modification to the rotor system. The basic 23010 airfoil of the baseline rotor blade was replaced with an advanced aerodynamic performance airfoil (Boeing Vertol VR-7, VR-8). Rotor solidity increases from 0.0670 to 0.0850.

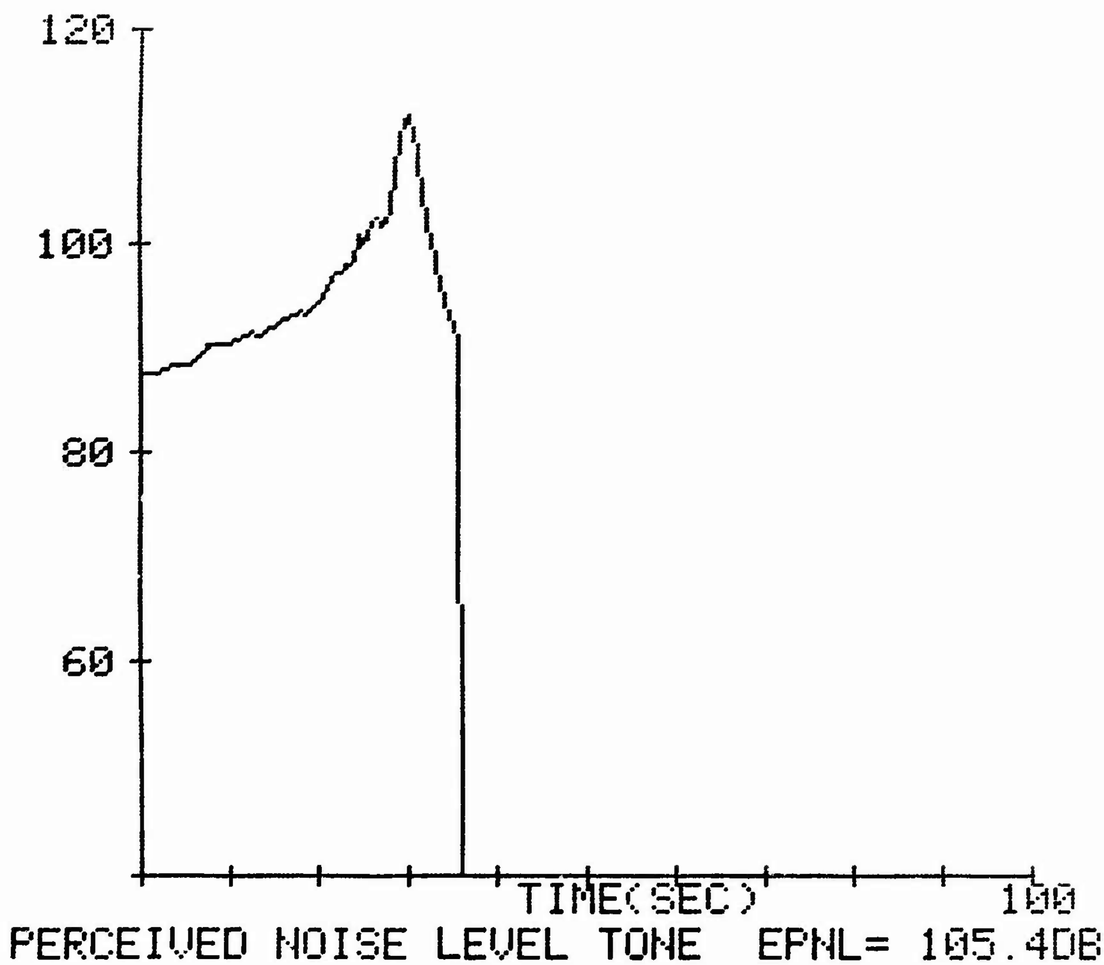


FIGURE 31. CH-47 PNLT TIME HISTORY

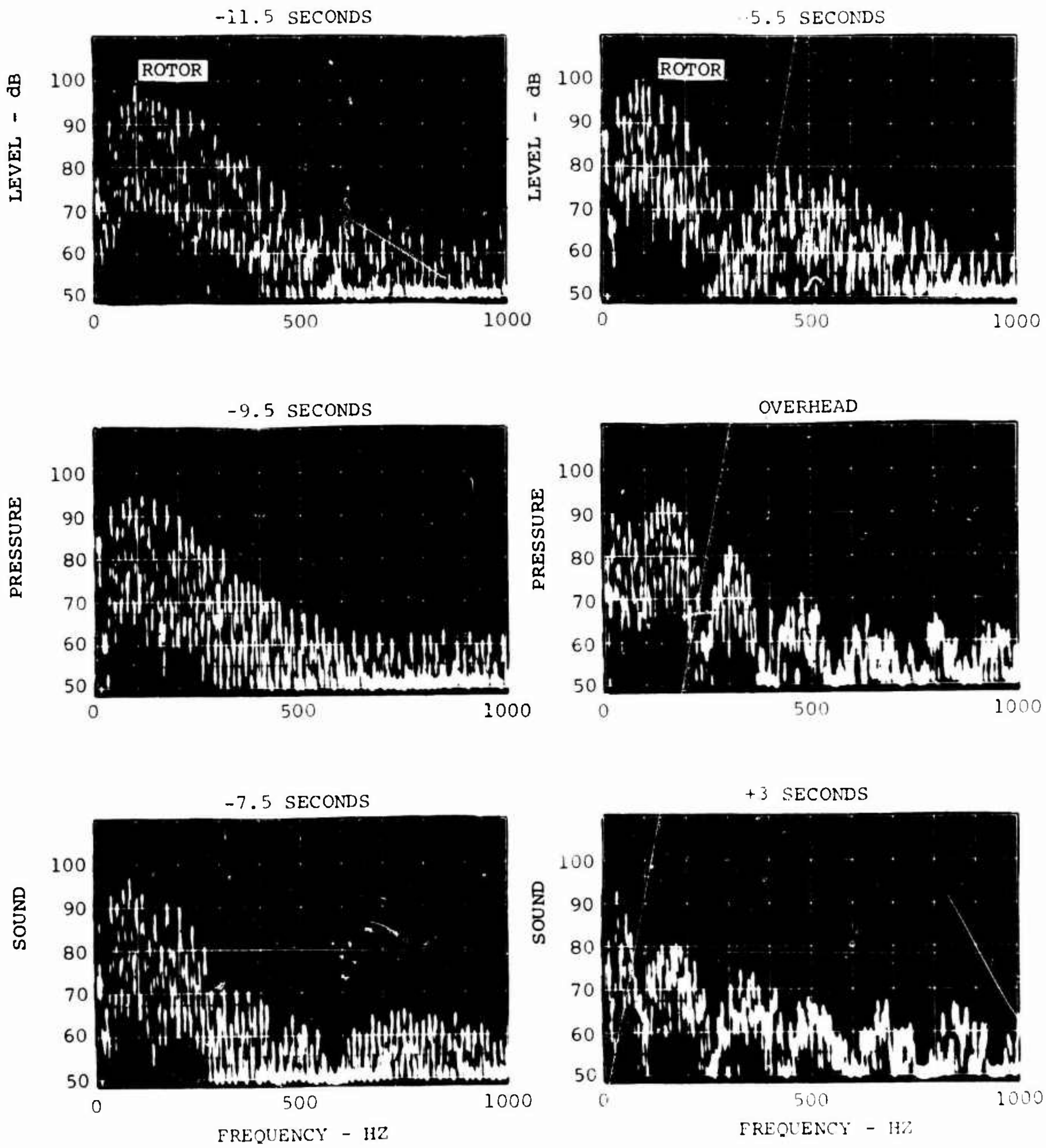
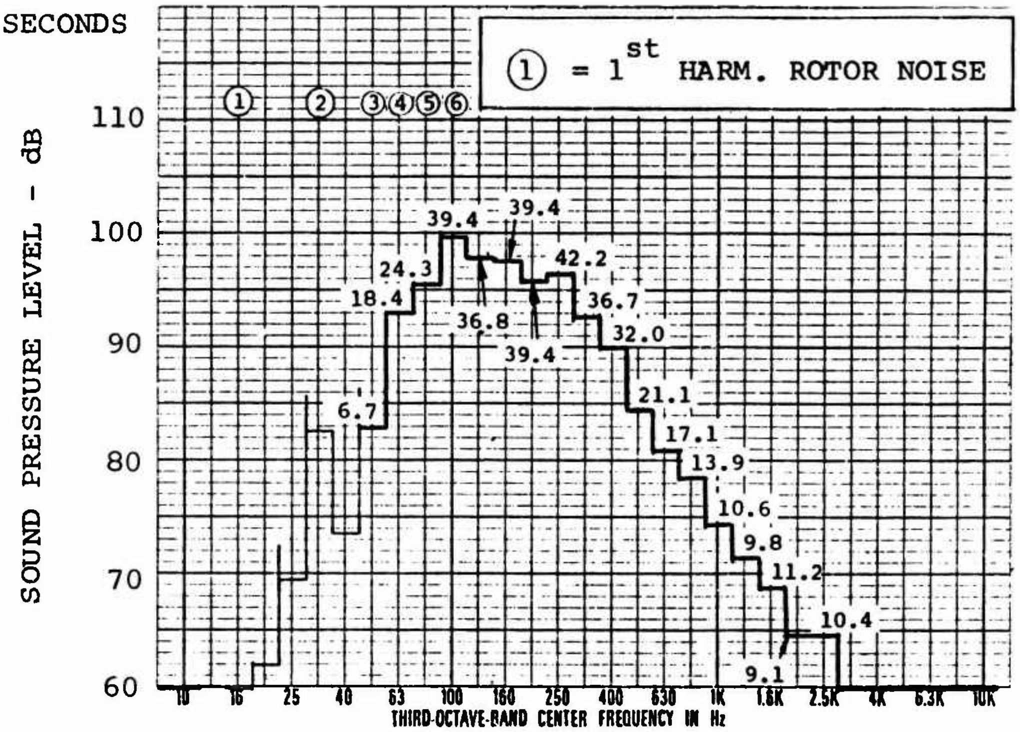


Figure 32. Narrowband Spectra of CH-47C Flyover Noise

-11.5 SECONDS



-9.5 SECONDS

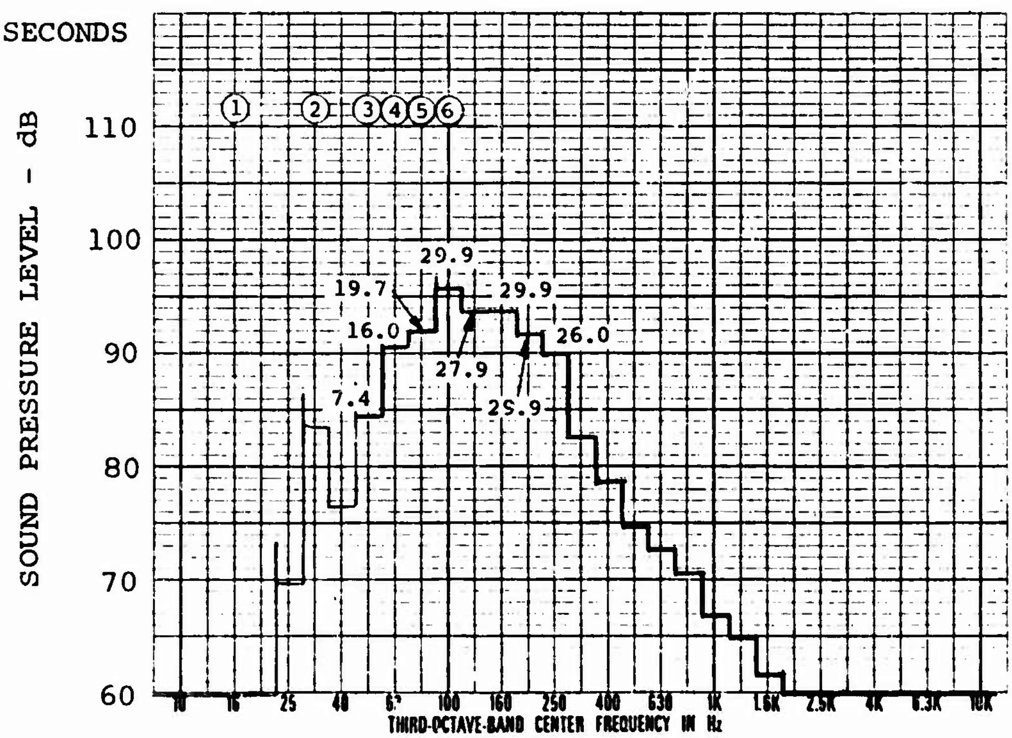
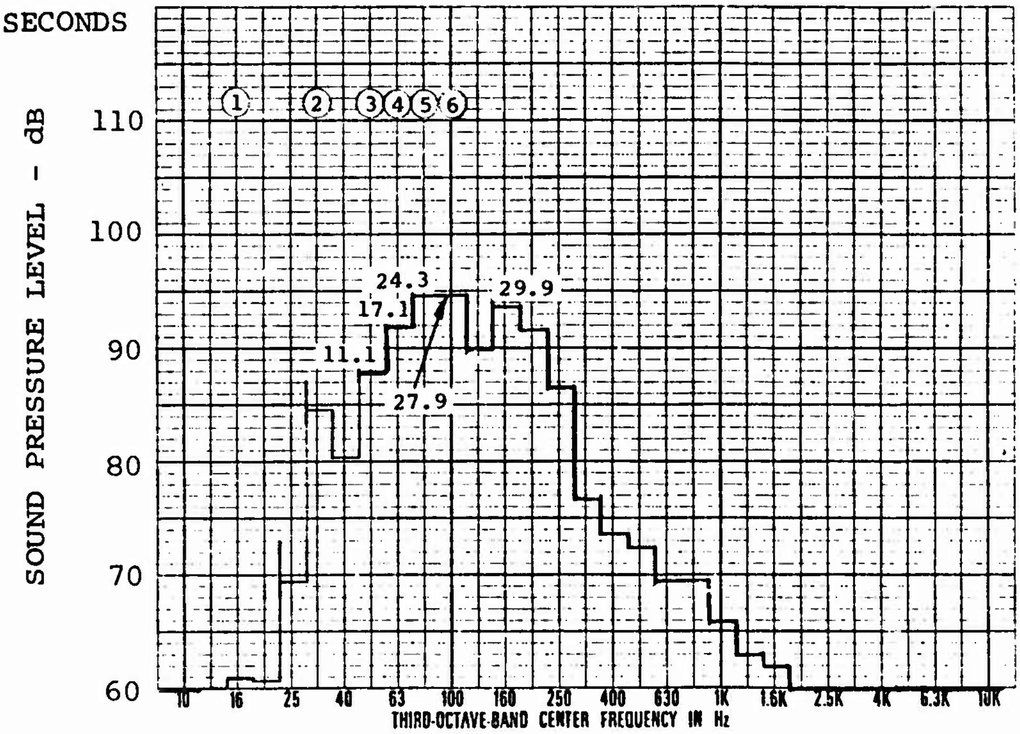


FIGURE 33. 1/3 OCTAVE SPECTRA OF CH-47C FLYOVER - 41,000# G.W., 245 RPM, 141 KT. SHEET 1.

-7.5 SECONDS



-5.5 SECONDS

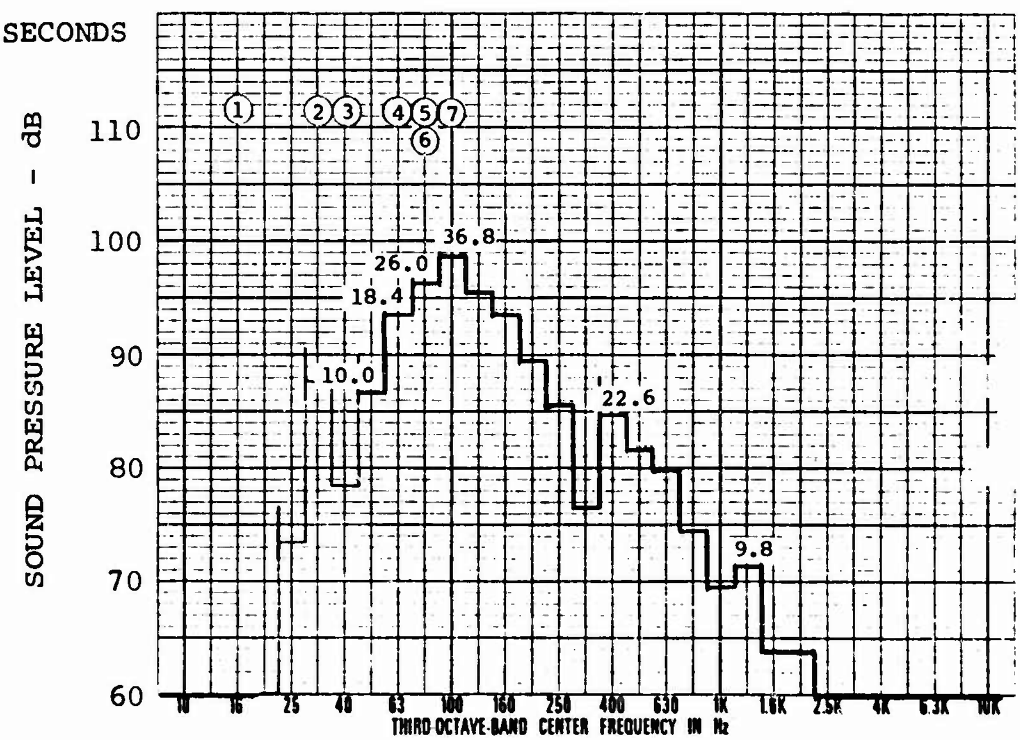
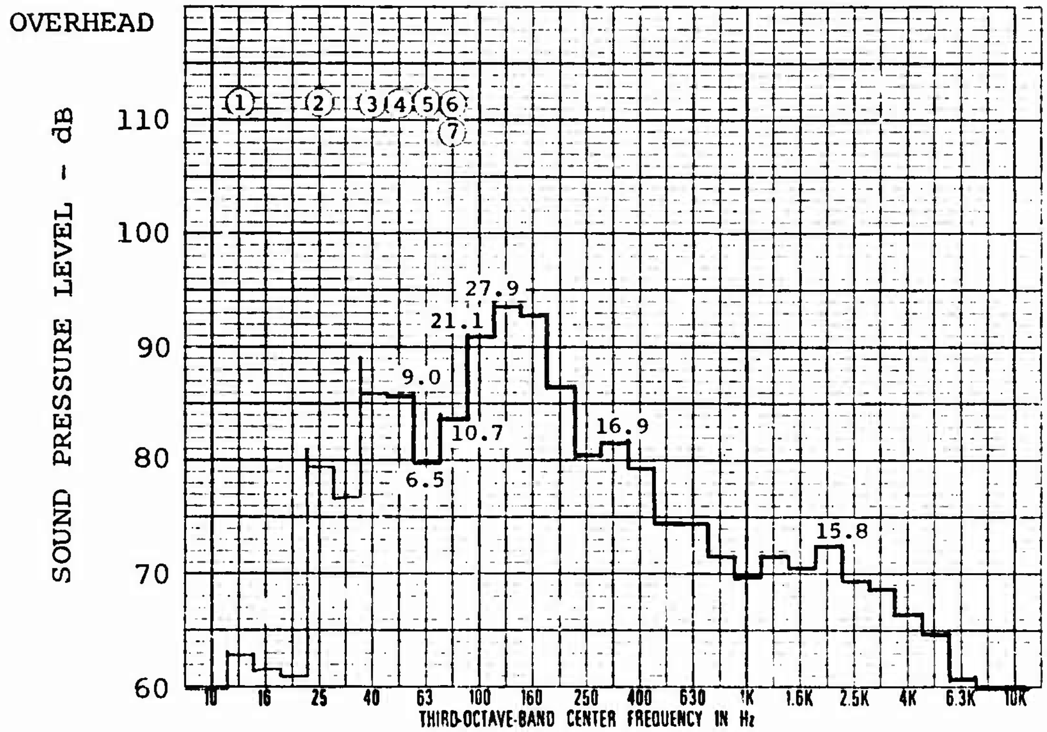


FIGURE 33. 1/3 OCTAVE SPECTRA OF CH-47C FLYOVER - 41,000# G.W., 245 RPM, 141 KT. SHEET 2.



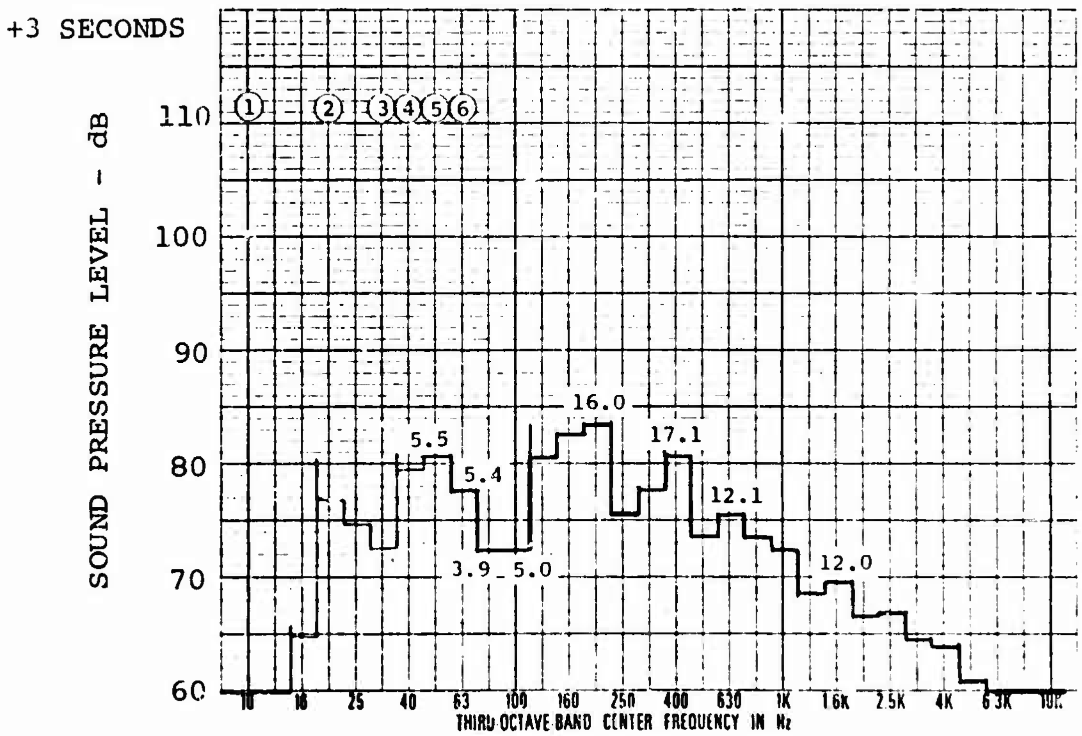


FIGURE 33. 1/3 OCTAVE SPECTRA OF CH-47C FLYOVER - 41,000# G.W., 245 RPM, 141 KT. SHEET 3.

Table 8. CH-47 Configuration Changes

	Baseline	Modification 1	Modification 2	Modification 3	Modification 4
V_t (ft/sec)	770	707	707	675	691
RPM	245	225	225	215	220
No. of Blades	3	3	3	3	4
Airfoil	23010-1.58	23010-1.58	VR-7,8	VR-7,8	VR-7,8
Chord (ft)	2.10	2.10	2.67	2.67	2.67
Radius (ft)	30.0	30.0	30.0	30.0	30.0
Flyover EPNL	106	99	96	93	90
Dynamic System	Basic	Basic	New gear set, accessory drive	New gear set, accessory drive	New gear set, accessory drive, new aft rotor shaft
Airframe	Basic	Basic	Basic	Basic	120 in. fuselage stretch 20 in. aft pylon plug
Powerplant	AL 5512	AL 5512	AL 5512	AL 5512	AL 5512
Weight Change	-	-	+251	+251	+3490

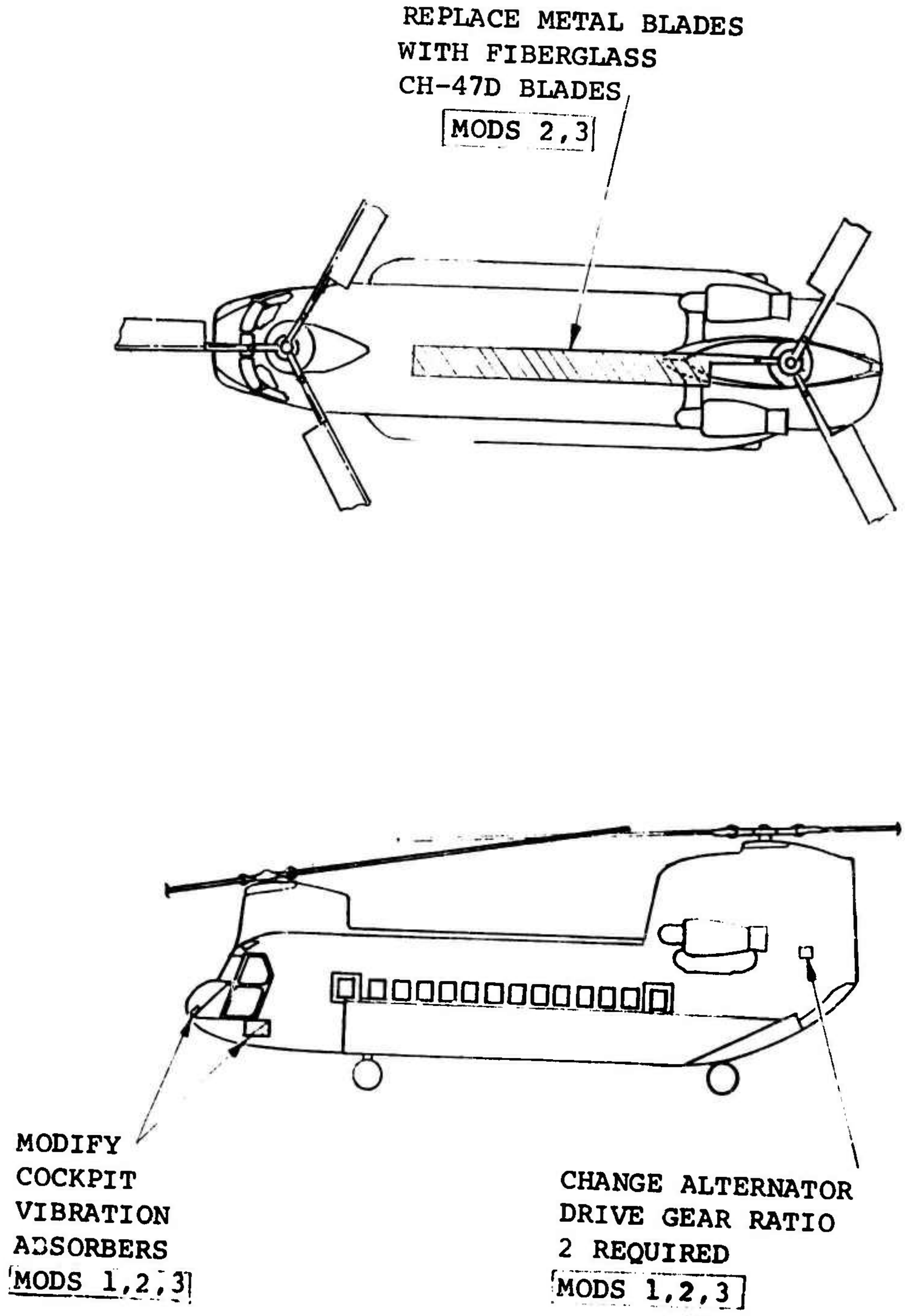


FIGURE 34. CONFIGURATION CHANGES TO CH-47, MODIFICATIONS 1-3

The use of this airfoil was made possible by the introduction of glass fiber reinforced plastic blade technology which permits airfoil contours that are not feasible with rolled steel 'D' spars and bonded blade box techniques. Noise levels of the Chinook in this configuration have been recorded during flyovers of a CH-47D demonstrator aircraft operating at 225 rotor RPM and with the noted VR-7, VR-8 airfoil glass blades. As required for the initial configuration change, both the accessory drive for the electrical system generator and the cockpit vibration absorbers required modification. In addition, there are some minor changes on the forward swiveling actuator lugs (new bearings), forward transmission swiveling actuator lugs (new bushings) and new forward pitch links. Modification 2 results in a 10 dB reduction in flyover EPNL for a 3 microphone average of 96 EPNdB.

Modification 3 to the CH-47 is similar to Modification 2, but has a rotor speed of 215 RPM rather than 225 RPM. This represents a tip speed of 675 ft/sec and is near the lowest rotor speed for which the CH-47 dynamic system has been qualified. Below this speed, dynamic system torques increase above design conditions and redesign of the drive train would be required. Modification 3 results in an estimated 93 EPNdB for a 3 microphone average of flyover noise. This level has been estimated from trends of flyover noise and advancing tip Mach Number developed from measured flyover time histories. Secondary configuration changes are similar to those identified for Modification 2 and also are presented in Figure 34 and Table 8.

Further reduction in flyover noise from the Modification 2 configuration must be achieved through a reduction in blade/vortex interaction noise. The primary source of blade/vortex interaction on a tandem rotor in forward flight arises from the trailed wake of the forward rotor passing through the aft rotor. Not only are the rolled-up tip vortex filaments of the forward rotor intersected by blades on the aft rotor, but rotor inflow is significantly more turbulent. Avoidance of intersections can be obtained by differential cyclic pitch on forward and aft rotors to achieve the desired separation distances. However, at high forward speeds this introduces undesirable rotor shaft bending stresses, reducing component lives and creates undesirable fuselage attitudes with high drag.

A more direct approach is to reduce overlap of the rotors and increase separations by changes to the layout of the airframe. This was the approach taken for Modification 4 to the CH-47 in which the rotor overlap was reduced from 34 percent to 22.5 percent by stretching the fuselage an additional 120 inches and increasing the vertical separation of the rotors by adding a 30 inch plug in the aft pylon (see Figure 35). Concurrently, a fourth blade was added to each rotor. This configuration change was incorporated into a CH-47 in the early 1970's and was identified as the Boeing Vertol Model 347. Flight testing of this aircraft demonstrated average flyover noise levels of 92 EPNdB at an altitude of 61m (200 feet). The Modification 4 configuration is shown with glass fiber reinforced plastic blades with the VR-7, VR-8 airfoil, although the Model 347 flew with blades having a NASA 23010 section. Estimates of noise levels for 150m (492 feet) flyovers when corrected for distance, directivity and duration are 2 EPNdB lower, or 90 EPNdB.

The increased fuselage length and four-bladed rotor system of Modification 4 permits the payload to be increased from 8,800 pounds (44 passengers) to 12,000 pounds (60 passengers) and remain within the capabilities of the

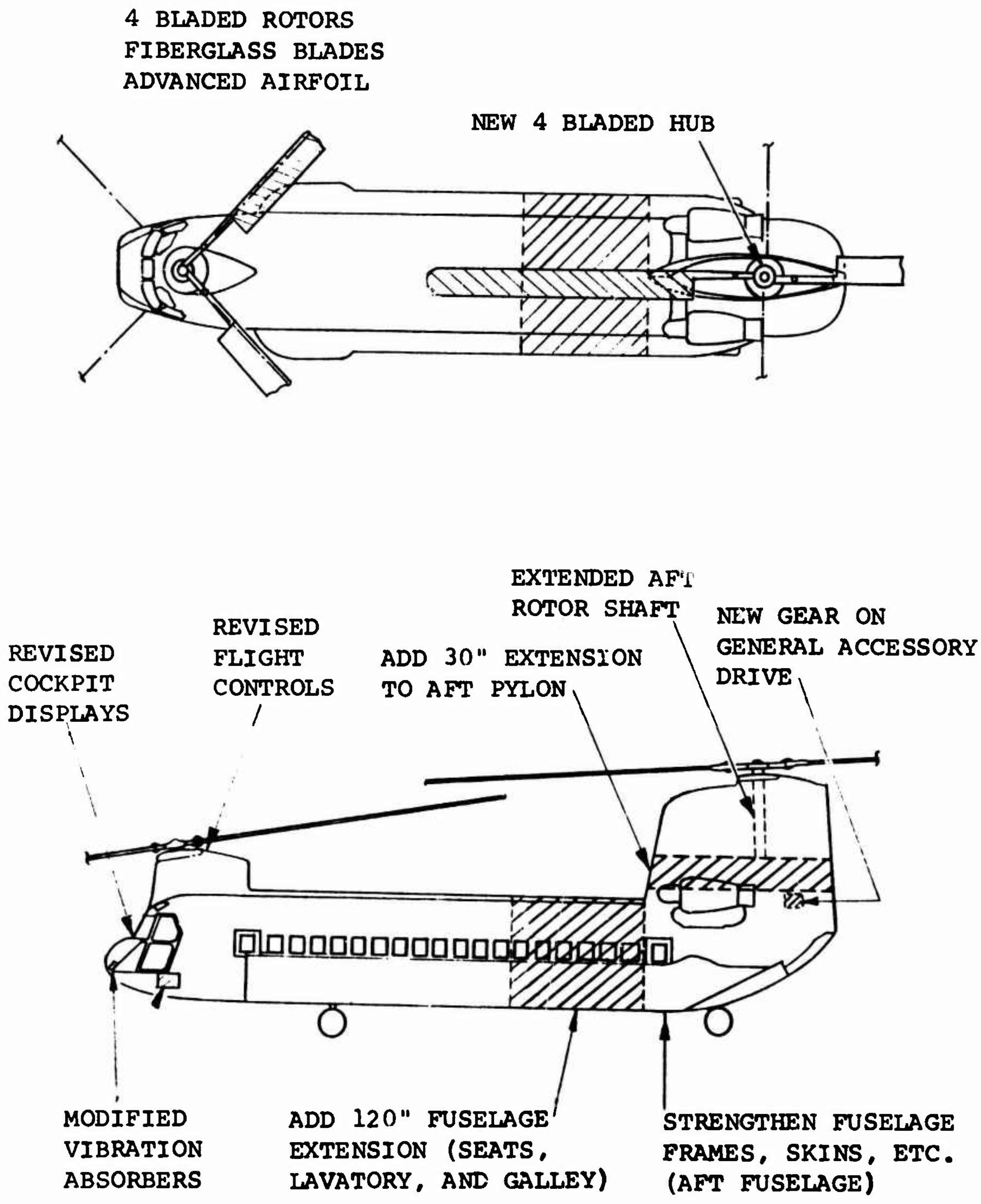


FIGURE 35. CONFIGURATION CHANGES TO CH-47,
MODIFICATION 4.

AL 5512 engines and existing drive train. Therefore, for this study, Modification 4 was considered both as a 44 passenger and a 60 passenger helicopter.

The weight increase for Modification 4 includes the additional airframe and pylon plug. Secondary changes also include an extended aft rotor shaft, additional synchronizing shafts between forward and aft rotor transmissions, new swashplates and flight controls for the four-bladed rotors, additional hydraulic lines, and air comfort system ducting for the fuselage extensions. As noted for previous modifications, the generator drive gear set is replaced with a lower ratio unit and cockpit absorbers are retuned to the lower rotor speed. These changes have been identified as shown in Figure 35 and Table 8. For the 60 seat configuration, additional passenger seats, interior traim, a galley and lavatory are included.

Noise sources on the CH-47, other than the rotors, are sufficiently below the rotor that they do not contribute to the flyover signature. Reductions in EPNL below those identified in Modifications 1-4 would require careful review to insure that engines and the dynamic system do not contribute to flyover noise. It is probable that engine inlet sources and rotor transmissions would require acoustical treatment for further noise reduction. It should be noted that documentation of flyover noise has been made for each of the noted CH-47 modifications except for Mod 3 and that reported levels have been verified in flight test.

Impact of Design Changes to Performance, Weight and Cost

Performance

The sensitivity of rotor speed reduction on aerodynamic performance of the CH-47 is shown in Figure 36. As a derivative of a military helicopter whose performance has been established from hover at 2000 feet altitude at 90°F conditions, and payloads in excess of those required for the civil transport configuration, reductions in rotor speed to 215 RPM are within the capability of the existing airframe and dynamic system.

For the civil transport role, the takeoff gross weight permitted under Category 'A' rules was based on 150 FPM climb with one engine inoperative at 1000 feet above the sea level takeoff site. Aircraft gross weight was composed of maximum passenger capacity and full fuel load.

Reduction in tip speed has a slight impact on cruise speed of the CH-47. Figure 37 shows a reduction of 3 knots for Modification 1 compared with the baseline. Modification 2 average cruise speed is 4 knots higher than the baseline due to installation of the advanced airfoil rotor blades even though tip speed was reduced from 770 ft/sec to 707 ft/sec. The tip speed of Modification 3 (675 ft/sec) reduces average cruise speeds to 136 knots. This drops to 135 knots for the stretched version as a result of higher rotor and fuselage (aft pylon) drag.

The effect of modifying the CH-47 for reduced noise level generally improves the range of the aircraft. Reduced rotor speeds result in lower rotor power requirements and thus lower specific fuel consumption. Compared with the baseline aircraft range of 229 miles, Modification 1 range increased 16 percent, Modification 2 increased 11 percent, Modification 3 increased 13 percent and Modification 4 increased 5 percent.

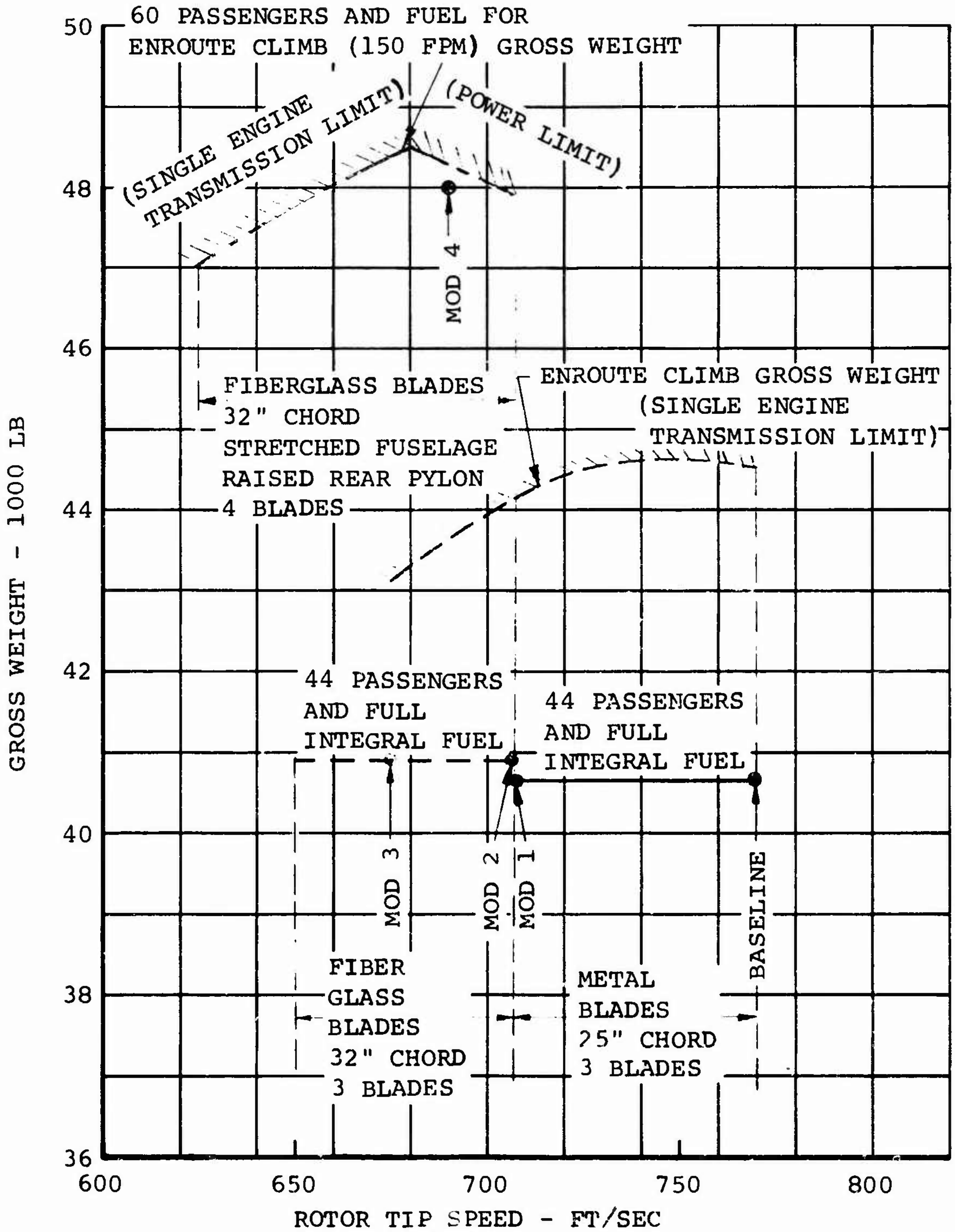


FIGURE 36 CH-47 PERFORMANCE LIMITATIONS

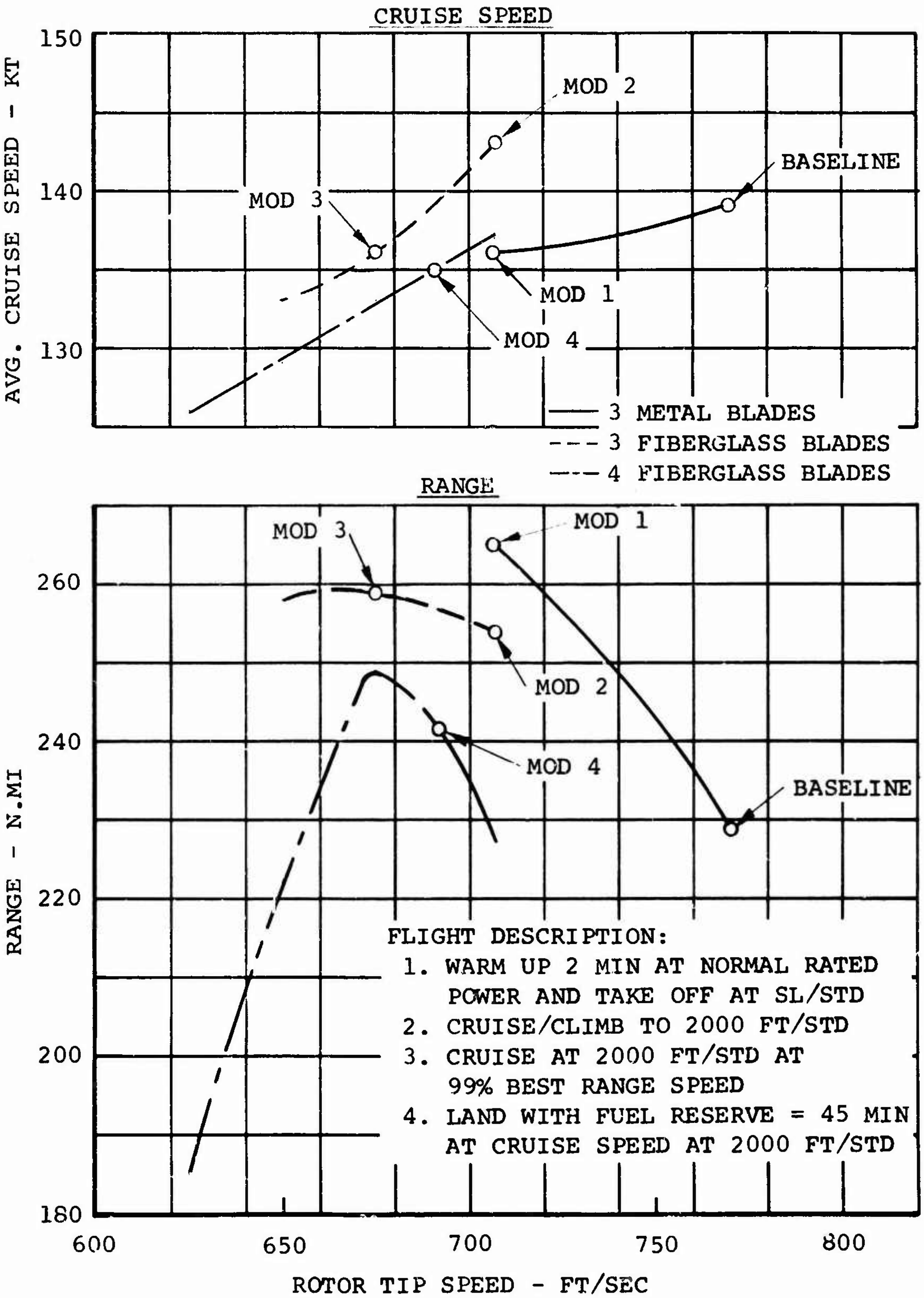


Figure 37 Effect of Rotor Speed Reduction on CH-47 Cruise Speed and Range

Weight

The weight empty for the baseline CH-47 is 23,725 pounds, with no change for Modification 1. There is an increase of 251 pounds for Modification 2 as noted in Table 9 which presents the increase in weight for each system modified. An increase of 3490 pounds accompanies the changes of Modification 4.

Cost

For the cost study, the CH-47 was considered to be a 'current production' helicopter. Costs associated with the design and development of the noted modifications were spread over the remaining production quantities which were assessed over a range of units from 50-1000 as shown in Figures 38 and 39.

Modification 1 consisted of only a reduction in rotor tip speed from 770 to 707 ft/sec resulting in a reduction of flyover noise at 7 EPNdB. Although the higher rotor speed was required to meet military demands, a reduction in rotor RPM from 245 to 225 is within civil gross weight operating limits and the only modifications required were a retuned vibration absorber and a new gear set in the accessory drive for the generator. Figure 39 illustrates that for production quantities of 50 units and greater, a small decrease in operating costs for the Modification 1 configuration occurs relative to the baseline configuration. This results from a reduction in block fuel required from 1862 pounds to 1698 pounds, since specific fuel consumption is reduced at the lower rotor speed.

Modification 2 retains the same rotor speed as Modification 1 but replaces the metal rotor blades with fiberglass units having an advanced airfoil and increased solidity. For the larger quantities, operating costs show a slight improvement even with regard to Modifications 1 and 3. This arises from a small reduction in cruise time resulting from a higher cruise speed (165 mph) for Modification 2 than for Modification 1 (152 mph) or Modification 3 (156.6 mph).

Modification 3 is a similar configuration to Modification 2, but rotor tip speed has been reduced to 675 ft/sec. Operational costs increase due to an increase in block time for the mission, with a production quantity of 50 aircraft resulting in a 5.5 percent increase in seat mile cost relative to the baseline configuration. Modification 3 has a flyover noise signature 13 EPNdB below the baseline CH-47C helicopter.

Modification 4 achieves a 16 EPNdB reduction in flyover noise with a considerable change to the basic airframe is noted in the previous section and illustrated in Figure 35. The configuration change results in a substantial increase in cabin size and payload capability. Figure 39 illustrates the diverse effect on operating cost depending on whether the number of seats are held constant or advantage is taken of the increased size cabin by increasing capacity to 60 seats. This latter configuration is similar to the Model 347 as previously described.

The inclusion of the additional 16 seats for the Modification 4 configuration is an infringement of study guidelines which prescribed that helicopter payload and performance by maintained essentially constant for each configuration

Table 9. Impact of Design Changes on CH-47 Weight (Sheet 1)

A/C Mod	System	Baseline Weight (Lb)	Additional Weight (Lb)	Description of Change
1	Aft Rotor Transmission		0	Change Gear on Generator Accessory Drive
	Vibration Absorbers		0	Retune Cockpit Absorbers
	Total Weight		0	
2,3	Flight Controls		0	New Bushings on Swiveling Actuator
	Aft Rotor Transmission		0	Change Gear on General Accessory Drive
	Vibration Absorbers		0	Retune Cockpit Absorbers
	Rotor Blades		251	New Rotor Blades with Advanced Airfoil
	Flight Controls		0	New Forward Pitch Links
			0	New Bearings in Forward Swiveling Actuator
	Total Weight		251	

Table 9. Impact of Design Changes on CH-47 Weight (Sheet 2)

A/C Mod	System	Baseline Weight (Lb)	Additional Weight (Lb)	Description of Change
4	Body	4972	1194	Add 120" Section to Fuselage. Add 30" Plug to Aft Pylon Reinforced Aft Fuselage, Seats, Lavatory, Galley
	Aft Rotor Transmission		0	Change Gear on General Accessory Drive
	Aft Rotor Shaft	423	400	New Aft Rotor Shaft (30" External)
	Sync Shaft	242	50	New Sync Shaft for External Fuselage
	Rotor Blade	1908	740	New F/G Rotor Blades
	Hub	315	84	New 4-Bladed Hub
	Hinge/Pitch Shaft	1197	316	Add 4th Blade, Hinge and Pitch Shaft System
	Swashplate	381	40	New Swashplate and Pitch Link for 4th Blade
	Electrical System	636	60	Revised Instrument Panels, Console, O/H Panel, Displays, Wiring for 120" Extension
	Hydraulics	225	65	Add Hydraulic Lines for 120" Extension
	Furnishings & Equipment	2818	516	Add Interior Trim, Seats, Galley, Lavatory in 120" Extension
			0	Retune Cockpit Absorber
	Environmental Control System	259	25	Add HVAC Ducting for 120" Extension
	Total Weight		3490	

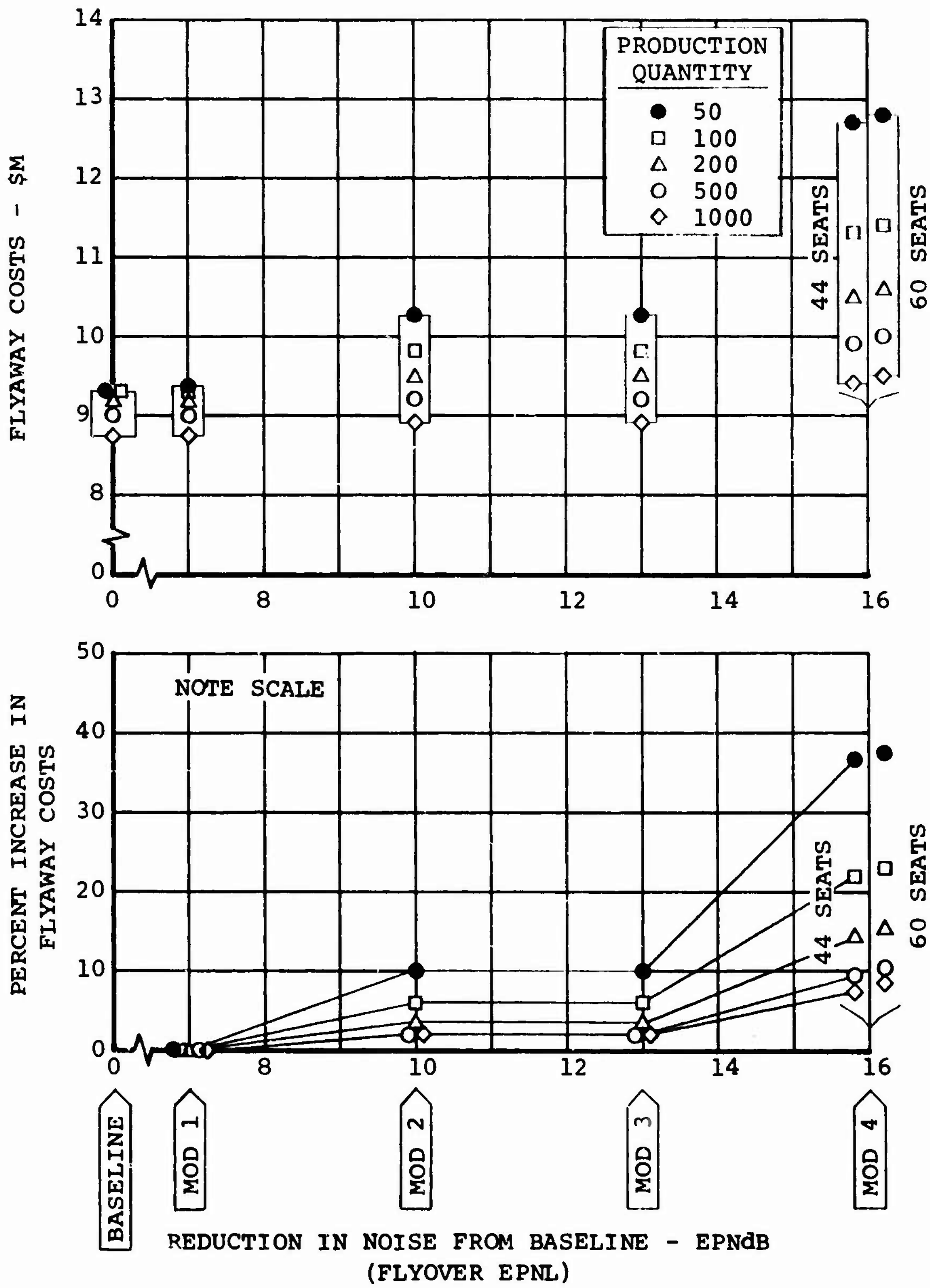


Figure 38. Effect of Configuration Changes on Flyaway Cost, CH-47

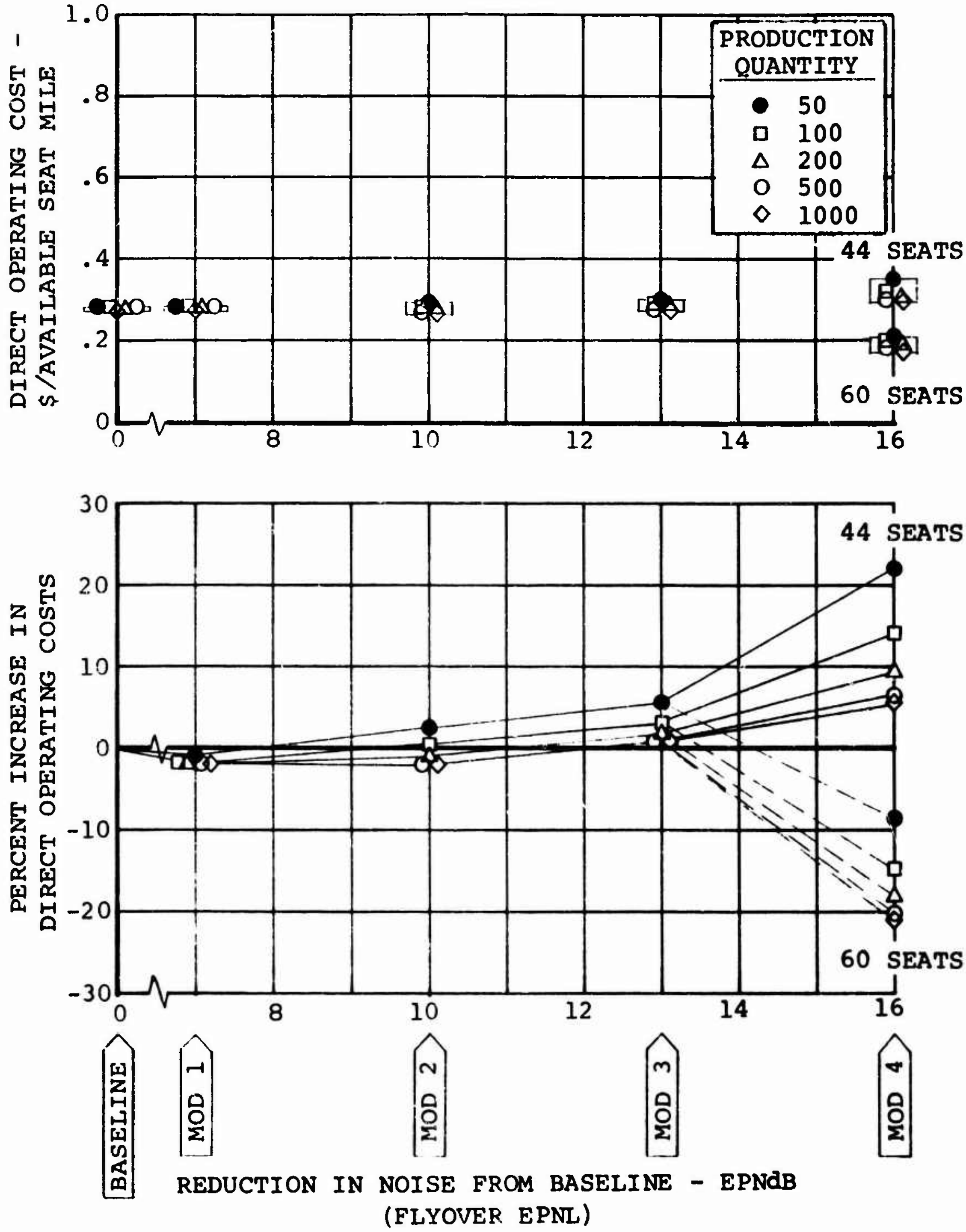


Figure 39. Effect of Configuration Changes on Direct Operating Cost, CH-47

evaluated. Therefore, for study purposes, the Modification 4 case was evaluated utilizing only the 44 seats of the baseline CH-47. Figure 39 shows that the operating costs are substantially higher than the 60 passenger configuration. Although this version represents a lower weight empty with slightly improved fuel consumption, a small increase in cruise speed and a slight decrease in acquisition costs resulted from the smaller interiors package. In reality, these trends can be projected from the costs of the previous modifications to the CH-47. As previously noted, however, it is certain that given the added payload of this stretched CH-47 the additional 16 seat capacity would be utilized to reduce seat mile costs for the civil application.

Heavy Lift Helicopter - Model 301

Vehicle Description

The Boeing Vertol Model 301 is a heavy lift helicopter (HLH) designed to provide vertical airlift capability for large and heavy loads. As originally designed, it was configured as a 'crane' helicopter primarily designed to lift external loads such as standard shipboard container modules, slings, platforms or special pods. For study purposes, a civil transport version of the Model 301 also was investigated. This transport configuration would have a larger fuselage with the capability of transporting 140 passengers in a 7-abreast, dual-aisle airline arrangement. Although similar in many ways, each helicopter has sufficient differences to be treated separately for the crane and transport configurations.

The heavy lift helicopter, described in this section, is the result of an intensive design study performed for the U.S. Army. The HLH rotor system was assembled and whirled on both a rotor tower as well as an integrated powerplant/drive system test facility (DSTR) which included one rotor, transmissions, and engines.

Model 301 Crane

The Boeing Vertol Model 301 crane is a tandem rotor shaft-driven helicopter powered by three T701-AD-700 gas turbine engines of 8079 HP each (see Figure 40 and Table 10). It provides a vertical airlift capability for loads carried externally beneath the fuselage utilizing either a single or two point suspension system. The crew compartment accommodates a pilot, copilot, flight engineer and load controlling crewman. A combination troop/light cargo compartment is aft of the crew compartment. Aft of the troop compartment, the center section contains the cargo handling equipment in the forward and aft positions. Each of the two hoists are located in this section.

The main rotors are four-bladed and operate at 156 RPM (750 fps tip speed). A fly-by-wire flight control system has been incorporated in the aircraft.

Noise Sources

Model 301 flyover noise levels have been estimated using measured test rig data as well as predictive methodology. Noise levels were based on: (1) measurements of an HLH rotor on a whirl tower, (2) data obtained on the dynamic system test rig, (3) comparison with flyover noise levels measured on the Boeing Vertol Model 347 (similar rotor geometry to the Model 301), and (4) analytical predictions.

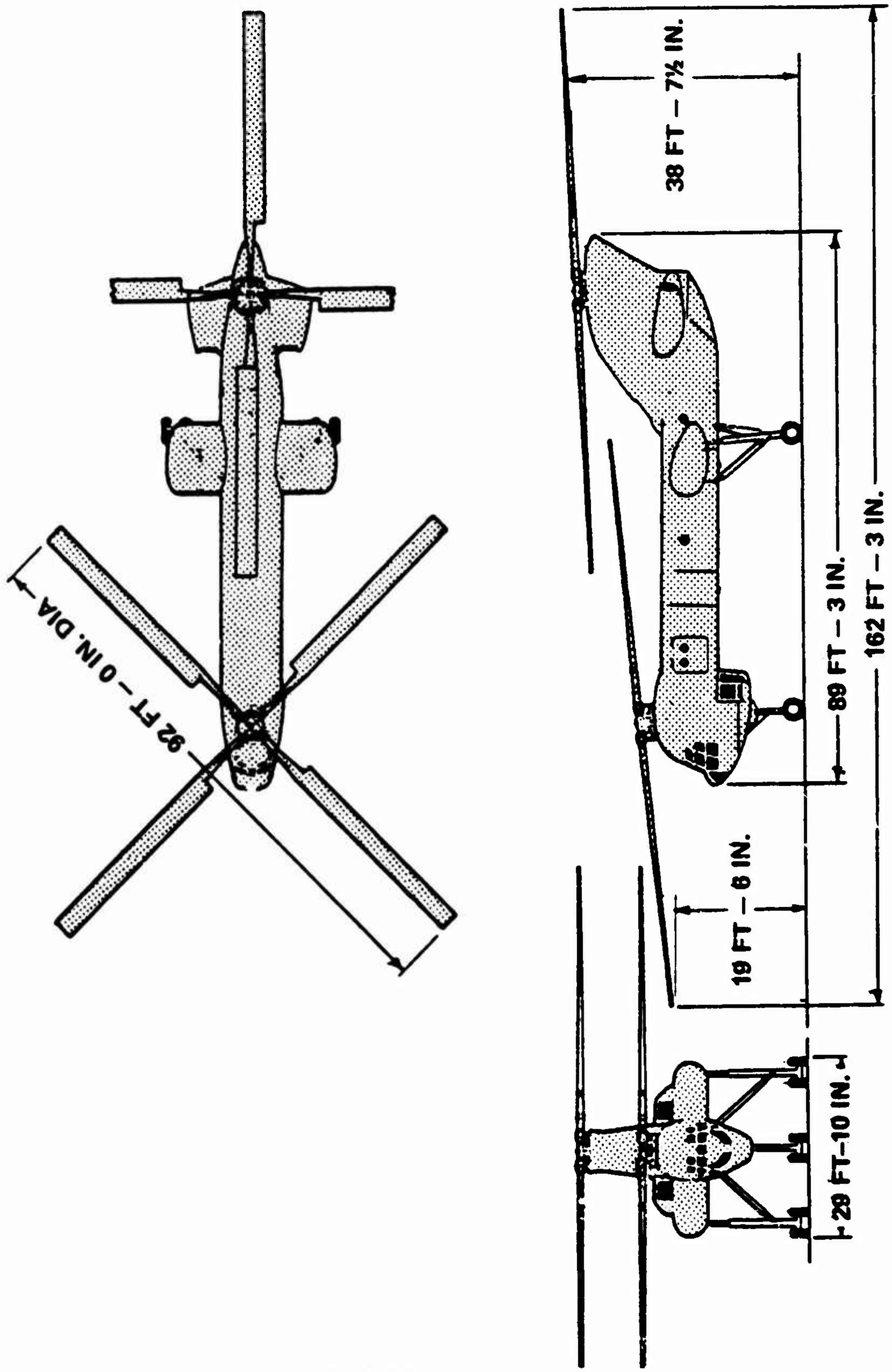


Figure 40   Model 301 Crane   General Arrangement

Table 10. Model 301 Characteristics

Maximum Takeoff Gross Weight	118,000 Pounds
Operating Weight Empty	64,594 Pounds
Normal Payload (SL/Std)	28.3 Tons
Range	(See Text)
Engines	T701-AD-700
Maximum Continuous Power Rating	8079 HP
Main Rotor	
Type	Articulated
Number of Blades	92.0 Feet
Chord	40.0 Feet
Normal Cruise Speed	130 Knots
Main Rotor RPM	156
Accommodations - Crane	5 Crew 12 Troop Seats Cargo Area
- Transport	6 Crew 140 Passengers

The basic approach to estimating Model 301 flyover EPNL was to relate it to Model 347 flyover data since overlap and rotor configuration are similar on the two helicopters. Therefore, PNL flyover time histories were assumed to have similar characteristics, although absolute levels differed.

The absolute values for PNL_{max} for the Model 301 and Model 347 were predicted by the methods of Section III and the Appendix and are shown in Figure 41. This spectrum results in a $PNLT_{max}$ of 108 PNdB.

Configuration Changes - Tradeoff Variables

The configuration changes to the Model 301 crane are listed in Table 11 and illustrated in Figure 42. The modification includes new rotor blades, modified cockpit vibration absorbers, a new gear ratio for the generators and acoustically lined engine inlets.

Impact of Design Changes on Performance, Weight and Cost

Performance

The sensitivity of rotor speed reduction on aerodynamic performance of the Model 301 is shown in Figure 43. As noted for the CH-47, the 301 is a derivative of a military helicopter. At a gross weight of 118,000 pounds reductions in rotor speed to 141 RPM (681 ft/sec) are within the capability of the existing airframe and dynamic system.

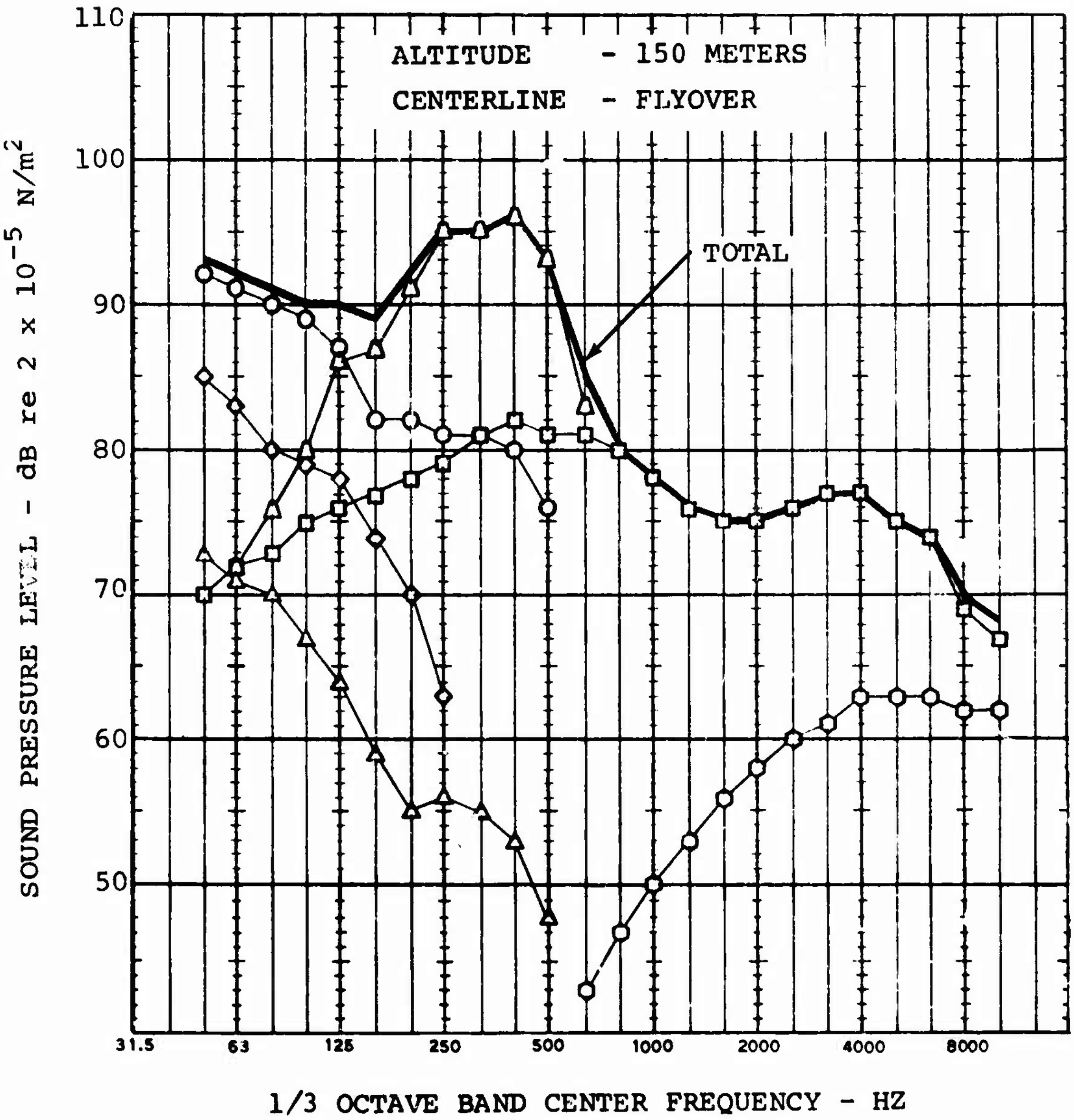
Figure 44 shows the relationship between the number of sorties that can be conducted and the mission radius of each sortie. In addition, at a mission radius of 3 nautical miles, for example, reduction of tip speed from 750 ft/sec to 681 ft/sec reduces rotor power required and SFC, effectively increasing the number of sorties from 37 to 40. A reduction in payload of approximately 9000 pounds is associated with this rotor speed reduction, however, limiting the payload to 41,000 pounds. This represents a reduction in payload of 18 percent. Obviously, payloads in excess of 41,000 pounds would be the only loads affected.

Weight

The modification to the crane version of the Model 301 increases weight empty from 62,120 pounds to 63,534 pounds, an increase of 1414 pounds. This is comprised of an additional 643 pounds due to increased blade chord, 150 pounds for the retuned vibration absorbers and 621 pounds resulting from the lined engine inlet plenum area (see Table 12).

Cost

For the cost study, the mission of the Model 301 crane was based on 2.3 statute mile sortie as defined in the 'performance' section, cruising outbound at 90 knots to a work area. The hover pickup of load and inbound cruise, hover and deposit of load results in a block speed of approximately 49 mph. Since the crane is designed to carry external loads, not passengers, the cost study was based on ton-mile rather than seat-mile costs. Ton-mile costs were developed from air-mile costs divided by the payload which is a function of tip speed. Thus, at a tip speed of 750 fps payload was approximately 25 tons while at 680 ft/sec the payload was reduced to 21 tons (Figure 43).



COMPONENTS

○ ROTATIONAL
□ BROADBAND
△ COMPRESSIBILITY
◇ THICKNESS
△ BLADE/VORTEX INTERACTION
○ COMPRESSOR BROADBAND

Figure 41 Model 301 Predicted Noise Spectra at PNL_{MAX}.

Table 11. Model 301 Configuration Changes

	Baseline	Modification 1
V_t (ft/sec)	751	680
RPM	156	141
No. of Blades	4	4
Airfoil	23 Series VR-7, 8	23 Series VR-7, 8
Chord (ft)	3.33	4.17
Flyover EPNL	102	99
Dynamic System	Basic	Basic
Airframe	Baseline transport, crane	Baseline transport, crane
Powerplant	Allison 501-M62B	Allison 501-M62B Lined inlet plenum, all nacelles.
Weight Change		+1594 transport +1414 crane

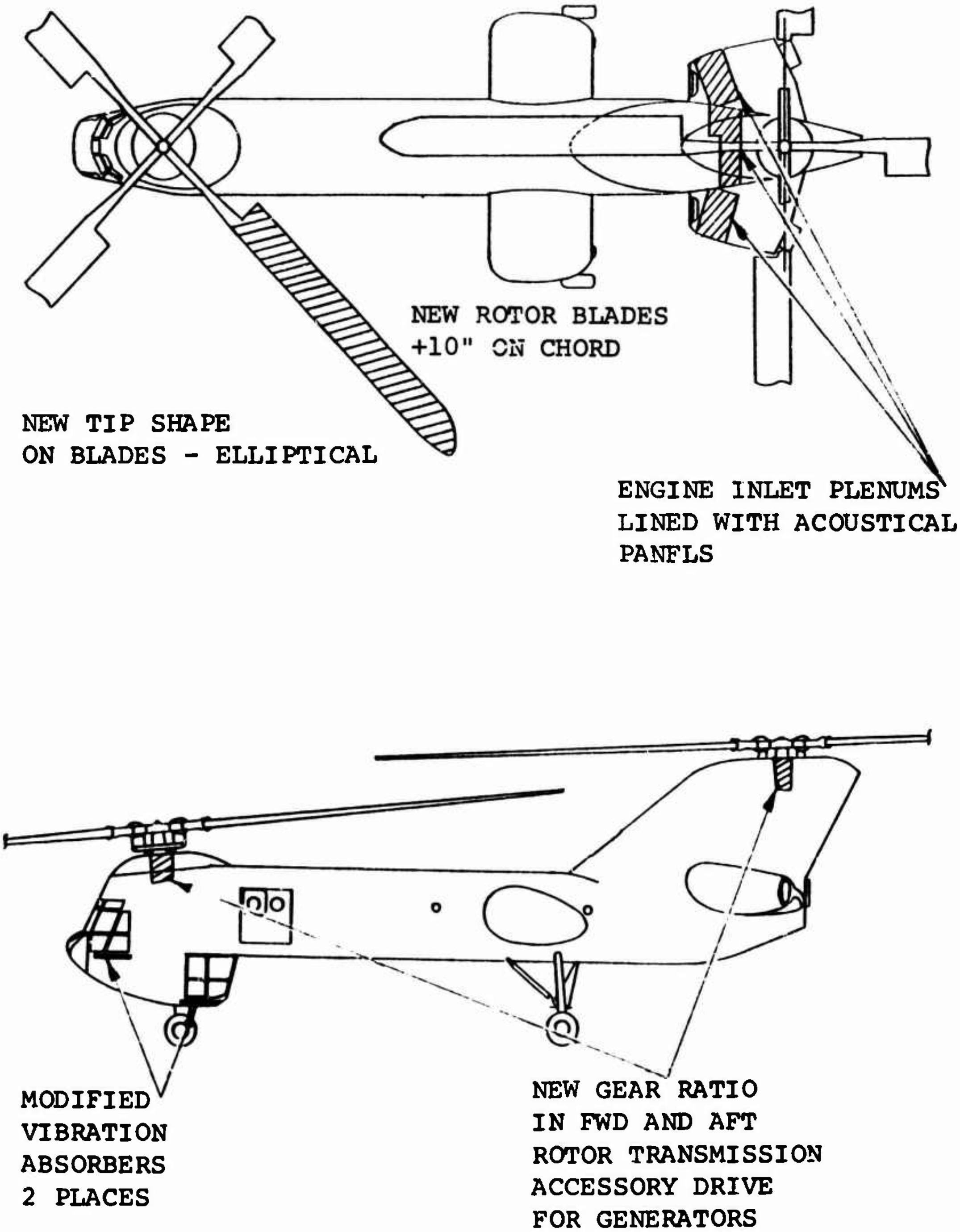


Figure 42 Configuration Changes to Model 301 Crane

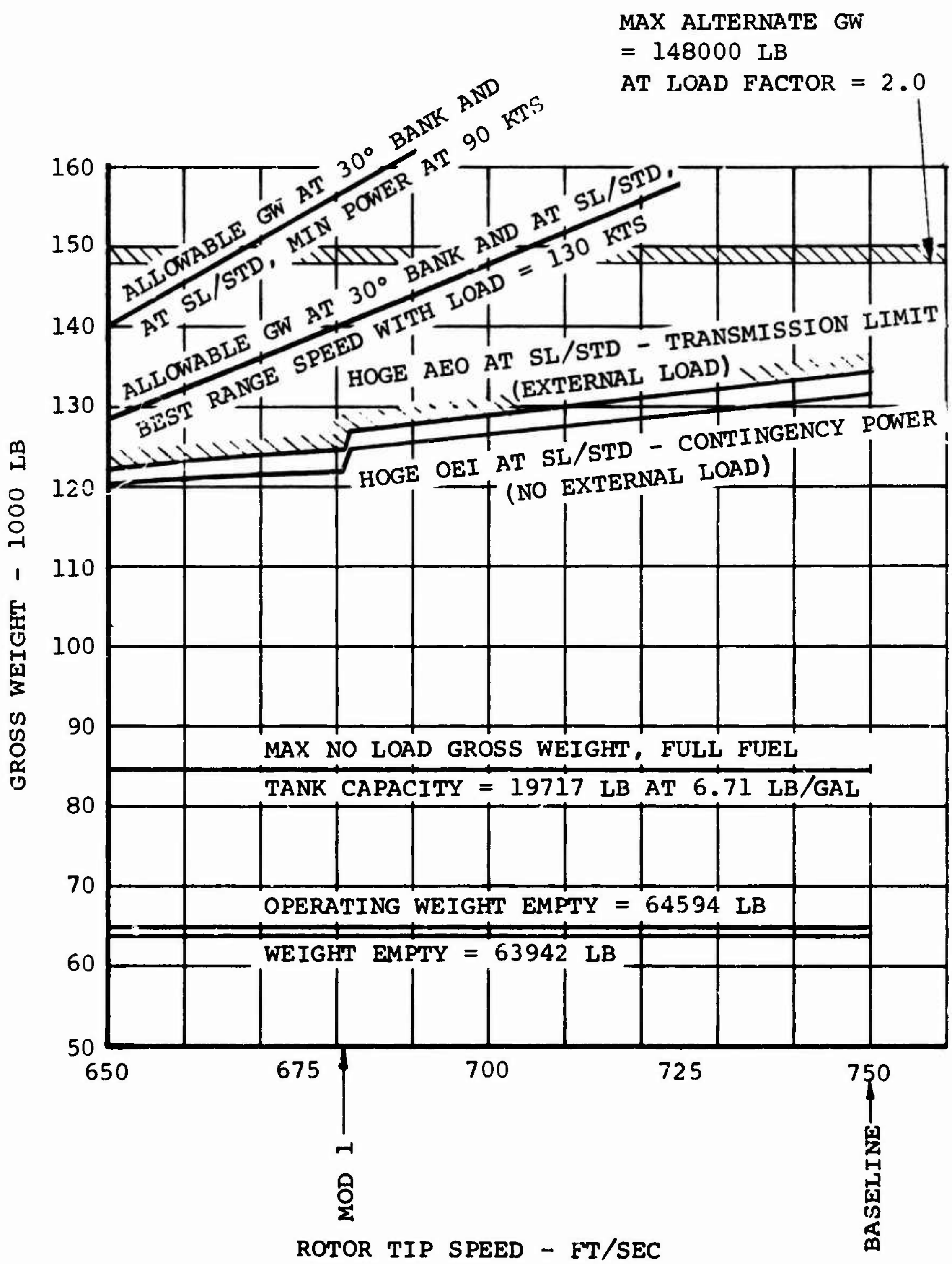


Figure 43 Model 301 Crane Performance Sensitivity to Tip Speed

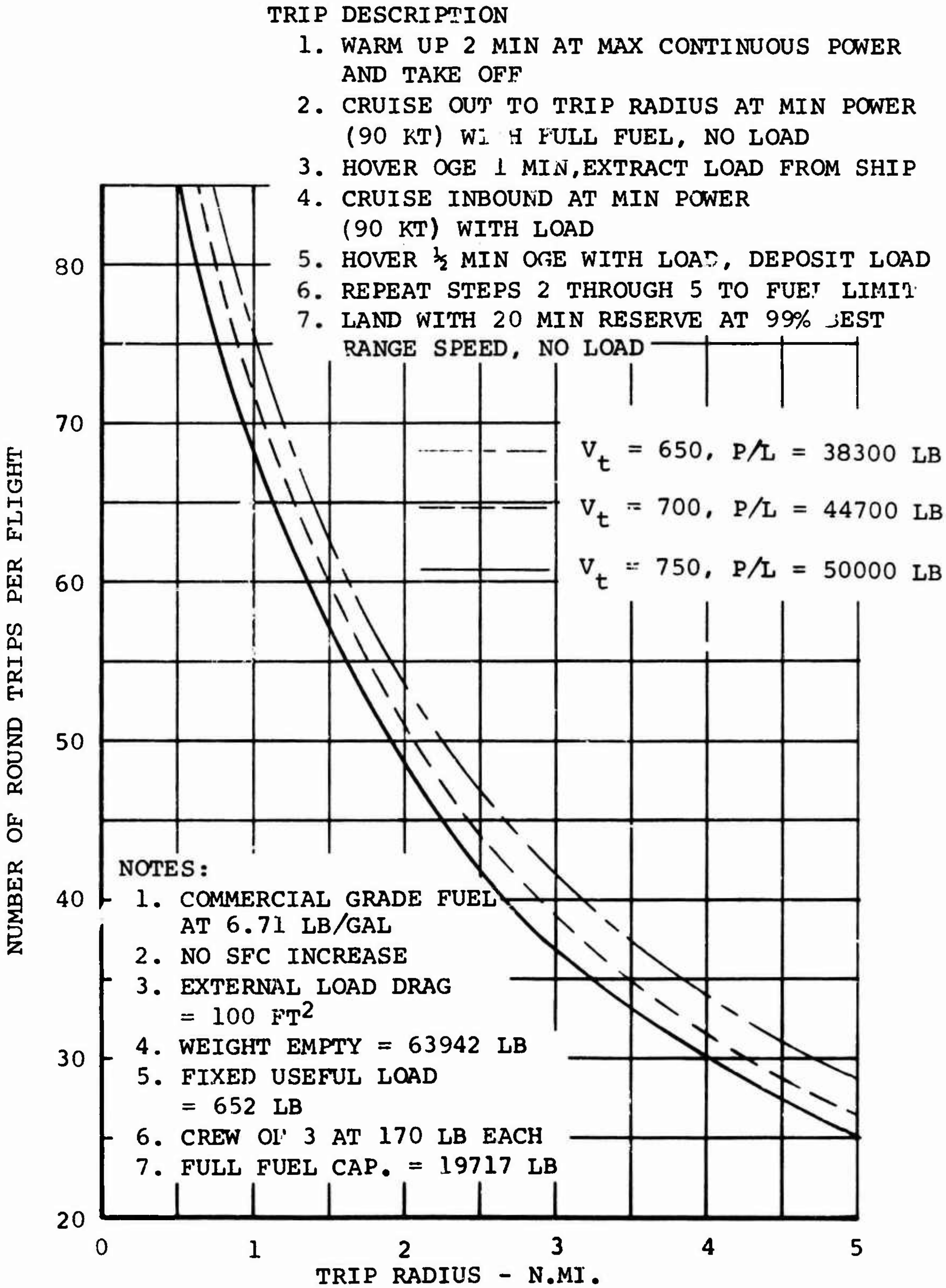


Figure 44 Model 301 Crane Mission Capability

Table 12. Impact of Design Changes on Model 301 Weight

CRANE

A/C Mod	Component	Baseline Weight (Lb)	Additional Weight (Lb)	Description of Change
1	Rotor Transmission		0	Change Gear Ratio on Generator Drive
	Rotor Blade	6264	643	Increase Blade Chord by 10% - Modify Tip Shape
	Furnishings & Equipment		150	Retune Vibration Absorbers Pilot, Copilot, Load Controller
	Powerplant Installation		620	Line Engine Inlet Plenum with Acoustical Panels
	Total Weight		1414	

TRANSPORT

A/C Mod	Component	Baseline Weight (Lb)	Additional Weight (Lb)	Description of Change
1	Body		112	Retune Floor Isolation System
	Rotor Transmission		0	Change Generator Gear Ratio
	Rotor Blade	6264	643	Increase Blade Chord by 10% - Modify Tip Shape
	Fuel System	5223	118	Retune Fuel Cell Isolation System
	Furnishings & Equipment		100	Retune Vibration Absorbers Pilot, Copilot Seats
	Powerplant Installation	840	620	Line Engine Inlet Plenum with Acoustical Panels
	Total Weight		1594	

The Model 301 crane was considered to be a 'new' helicopter for this study with the implied assumption that nonrecurring costs were spread over the entire production base. This production base was assumed over a range of 50 to 1000 units to display the effect which this variable has on operating costs.

Modification 1 to the baseline crane configuration results in a 3 EPNdB reduction in flyover noise as previously noted in this Section. Air-mile costs remain essentially unchanged as a result of the configuration change to the crane. However, the reduction in rotor tip speed from 750 to 685 ft/sec results in a reduction in payload from 25 tons to 21 tons resulting in a ton-mile cost increase by 17-19.7 percent (Figure 46). If crane payload had been 21 tons or less initially, no increase in ton-mile costs would be incurred due to the reduced tip speed of Modification 1.

Reductions in flyover EPNL below that defined by Modification 1 did not appear to be achievable as predicted by the methodology of Section III. A reduction in tip speed to 650 ft/sec with the increased solidity main rotor blade showed only a 1 dB reduction in Perceived Noise Level. Further reductions in rotor noise are not apparent for this 90 foot diameter rotor. Additional research is required in the area of broadband noise reduction of large rotors.

Model 301 Transport

Vehicle Description

The Boeing Vertol Model 301 transport, like the crane, is a tandem rotor, three engine helicopter. It has the same drive system and rotor as the crane, but the airframe is configured to transport 140 passengers (see Figure 47 and Table 10). The load controllers cab and associated flight controls, etc. have been removed.

Noise Sources

Since rotor geometry, powerplants, drive system and gross weight are similar on the crane and transport configuration, noise sources are similar. Configuration changes on the transport rotor system also are the same as those for the crane.

Configuration Changes - Tradeoff Variables

Configuration changes to the Model 301 transport are identified in Table 11. Modification 1 to the baseline 301 reduces flyover noise to 99 EPNdB.

Modification 1 illustrated in Figures 42 and 48, shows rotor speed reduced from 156 RPM to 141 RPM (751-681 ft/sec) and an increase in rotor solidity by the addition of 0.83 feet to the main rotor chord (40 inches to 50 inches). This increase in chord requires a redesigned blade which would include a modified tip configuration which is shown as elliptically shaped in planform.

The engine inlet plenum has been lined with absorptive panels similar to those currently employed on fixed wing transports to reduce engine inlet noise. This reduces powerplant levels and permits reductions in broadband noise to be realized.

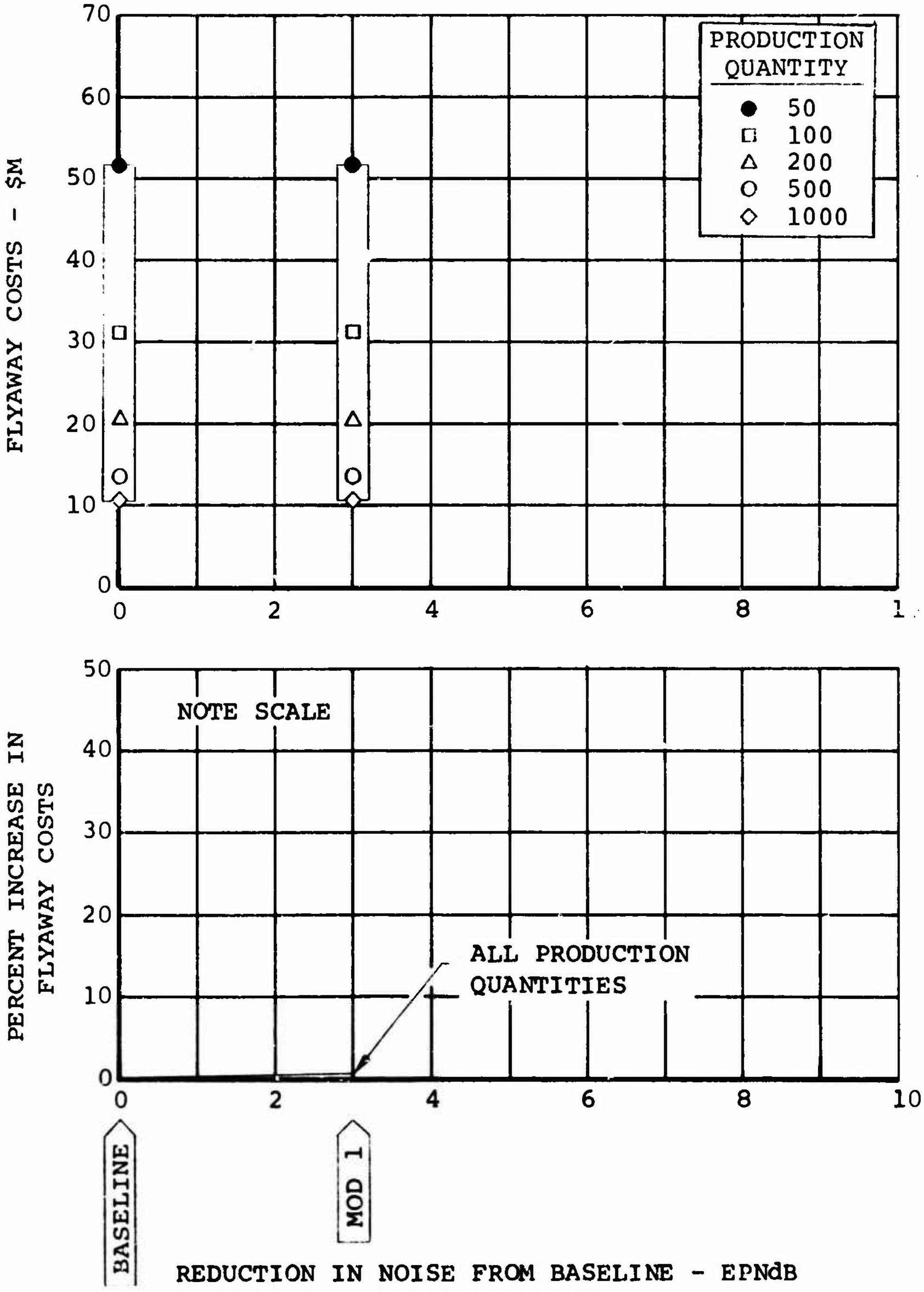


Figure 45. Effects of Configuration Changes on Flyaway Cost, Model 301 Crane

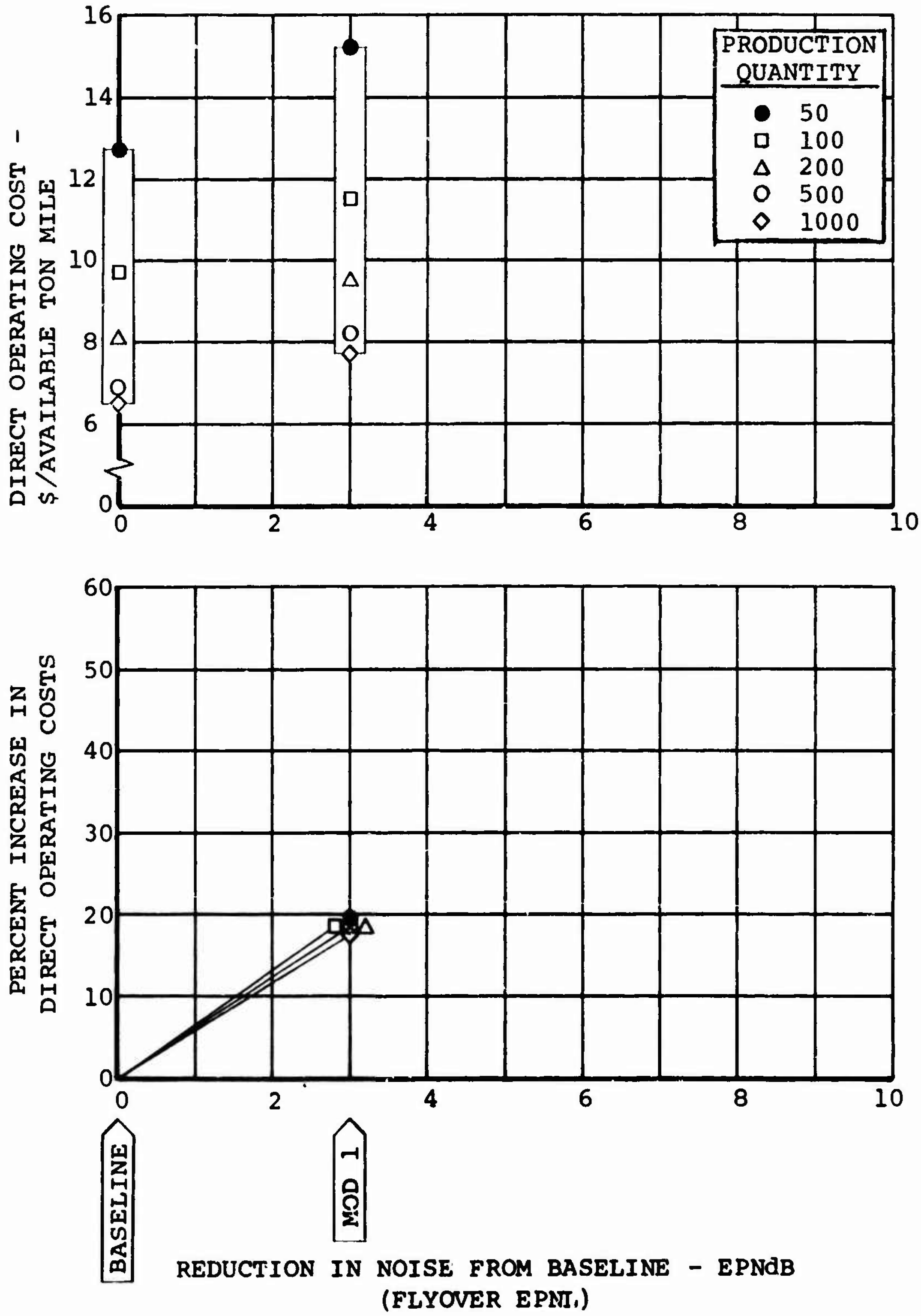


Figure 46. Effect of Configuration Change on Ton-Mile Cost, Model 301 Crane

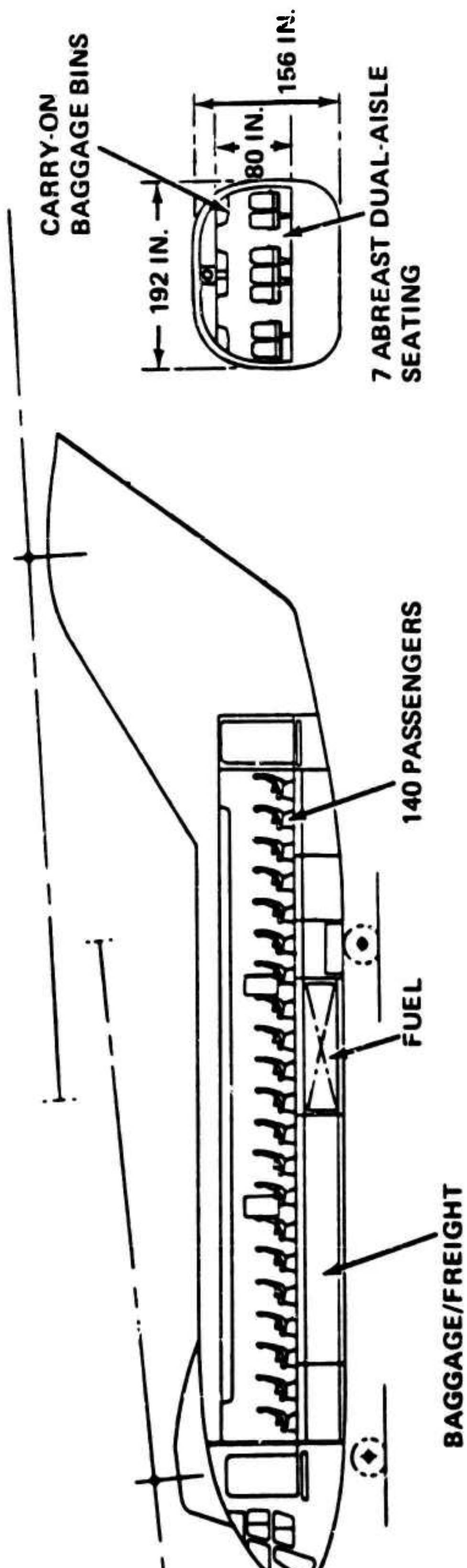
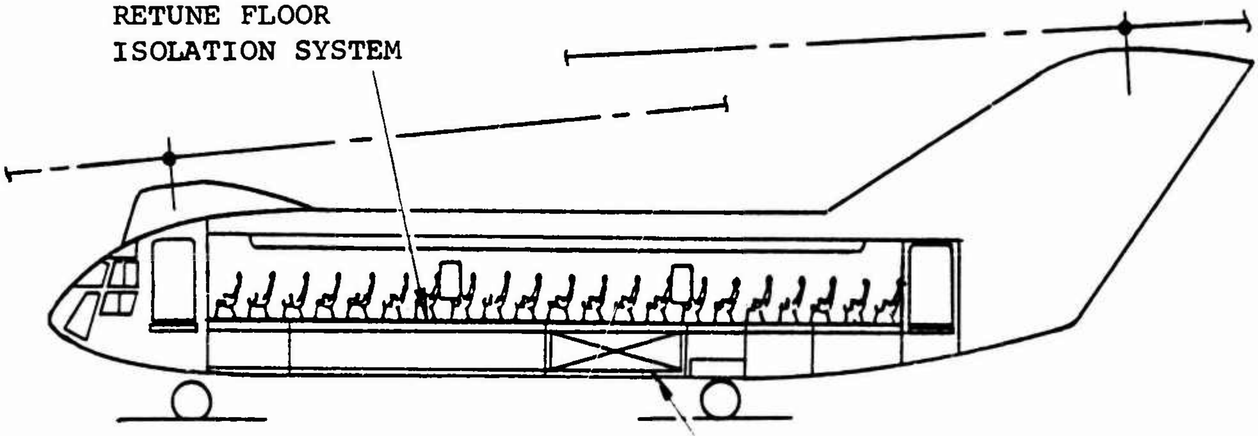


Figure 47 Model 301 Transport General Arrangement



REVISIONS TO TRANSPORT (CONFIGURATION 2) SAME AS FOR CRANE (CONFIGURATION 2) PLUS CHANGES SHOWN ABOVE.

REF: CRANE REVISIONS
- NEW ROTOR BLADES, INCLUDING TIP SHAPE
- MODIFIED COCKPIT ABSORBERS
- NEW GEAR RATIO FOR GENERATORS
- LINED ENGINE INLET PLENUMS

Figure 48 Configuration Changes to Model 301 Transport

Secondary changes required to the configuration include a new gear ratio in the forward and aft rotor transmission's accessory drive section to maintain operating speed for the electrical system generators. In addition, vibration absorbers in the cockpit and load controller's station must be retuned to the lower rotor speed and the rotor hubs require strengthening due to higher torques resulting from lower operating speed and also from higher loads resulting from increased weight of the blade. The higher torque would be partially offset by lower centrifugal loads at the lower rotational speed.

Impact of Design Changes on Performance, Weight and Cost

Performance

The sensitivity of rotor speed reduction on performance of the Model 301 transport is shown in Figure 49. The transport mission is a two minute warmup at maximum continuous power and takeoff at sea level-standard day conditions, climbing to 2000 feet and cruising out at 99 percent optimum range and landing with a 45 minute fuel reserve for a 99 percent optimum range cruise speed. For the transport gross weight of 118,000 pounds which includes 25,000 pounds fuel and 140 passengers and baggage at 200 pounds each, no impact on payload is noted to tip speeds of 665 ft/sec. Below this rotor speed, the number of passengers or fuel load would be reduced by a requirement to maintain a 35 degree band angle maneuver. Modification 1 operates at 681 ft/sec, which is within this limit.

Figure 50 presents the effect which reduced rotor speed has on cruise speed and range. Lower rotor power required results in an increase in cruise speed from 138 kt to 151 kt, a 9 percent increase. Similarly, the range increases from 417 to 545 nautical miles, an increase of 14 percent.

Weight

The weight empty of the transport increases by 1594 pounds for Modification 1, from 64,638 to 66,232 pounds. A brief weight statement for only those items that have been modified is presented in Table 12. Note that for the transport, cockpit vibration absorber weight has been reduced to 100 pounds to reflect the deletion of the load controller's station. In addition, the floor isolation system must be retuned for Modification 1 to the baseline helicopter, adding 112 pounds to weight empty.

This PNLmax level was then converted to an equivalent flyover EPNL based on Model 347 tests. The results of this testing indicated that for the average of the three microphones Effective Perceived Noise Level was 5 dB less than PNL_{max} measured on the centerline for a flyover EPNL of 103 EPNdB.

Cost

The mission of the Model 301 transport for this study as defined in the performance section consisted of a 100 seat mile flight at a cruise speed of 138 knots. This cruise speed represents 99 percent best range speed. Block speed associated with this is 125.8 knots. The transport configuration seats 140 passengers resulting in a design gross weight of 118,000 pounds as noted in the section on Performance. This gross weight is well below the engine torque limits shown in Figure 49 for all rotor speeds, and the reduced tip

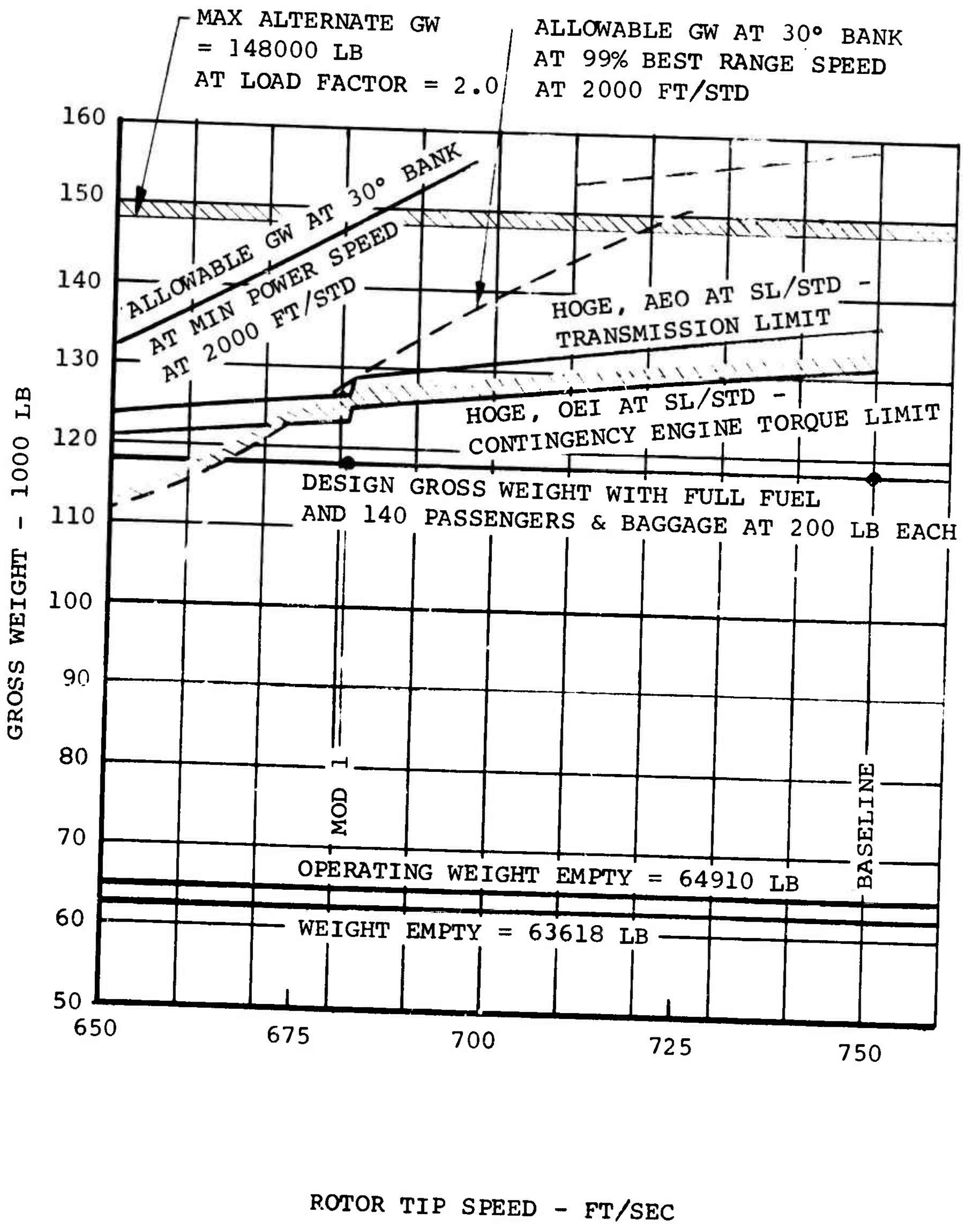


FIGURE 49 MODEL 301 TRANSPORT PERFORMANCE LIMITATIONS

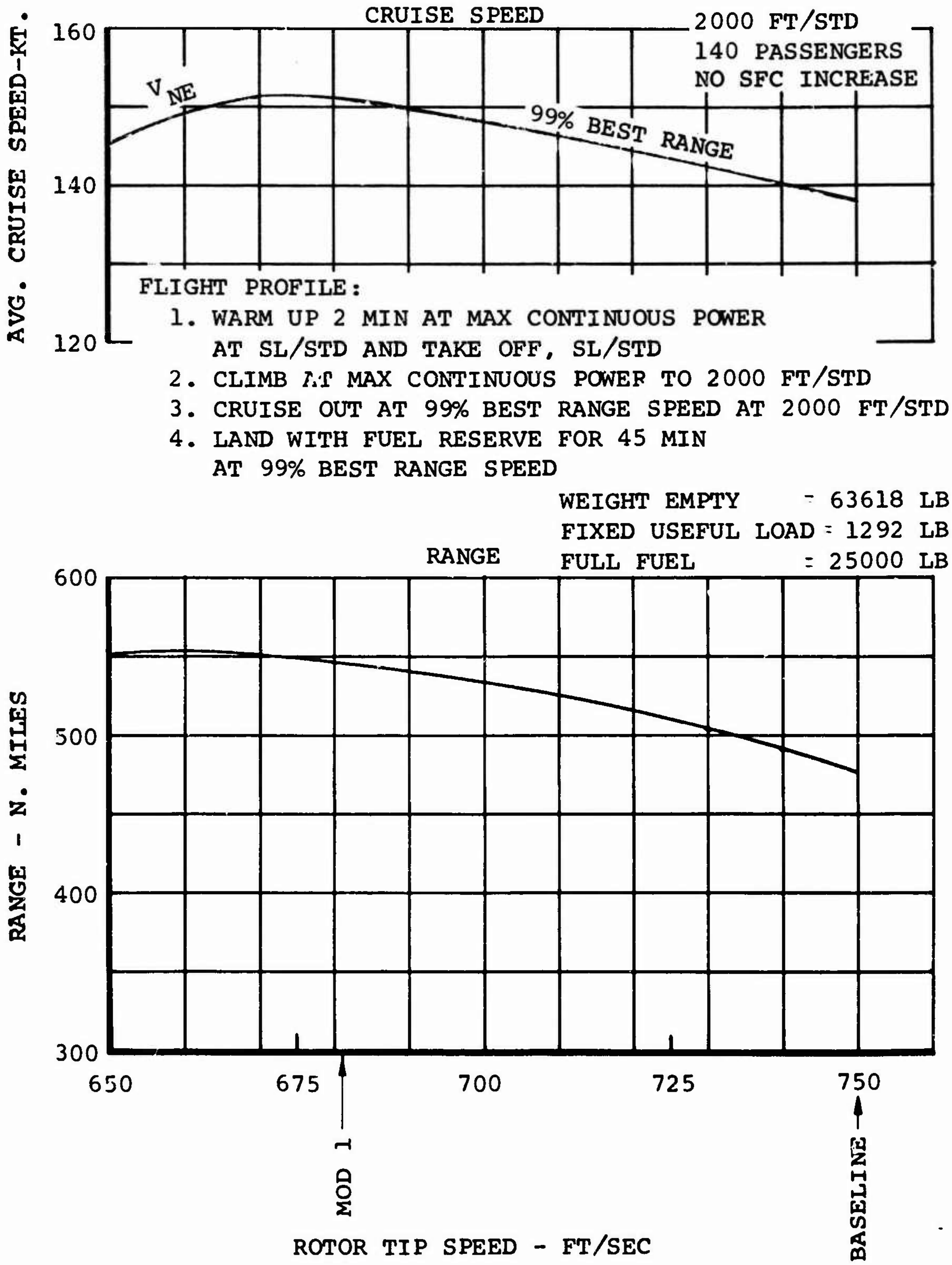


Figure 50 Effect of Rotor Speed Reduction on 301 Transport Cruise Speed and Range

speed of the Modification 1 configuration does not limit the transport payload capability. The direct operating costs in dollars per seat mile were utilized as calculated by the Reference 7 AIA costing program to compare configuration changes.

The Model 301 transport was considered to be a 'new' helicopter for this program with all nonrecurring costs spread over the entire production base. As for the other aircraft in the study, the production base was evaluated over a range of 50 to 1000 units to determine the effect which this variable has on operating costs.

Flyover costs and direct operating costs for the Model 301 transport are presented in Figures 51 and 52, respectively.

Changes in air mile costs resulting from Modification 1 changes to the transport version of the Model 301 produce only minimal changes in direct operating costs. As a 'new' aircraft, only those changes to the transport which incur additional material costs result in an increase in direct operating cost, since all nonrecurring is similar for both configurations. The additional material is in the wider chord blades, engine plenum acoustical linings, the retuned vibration absorbers in the cockpit and cabin floor and the fuel isolation system. These increased material costs, when added to all nonrecurring expenditures and spread over 140 seats for a distance of 100 miles result in a maximum change in DOC of 0.51 percent from the baseline.

As noted for the crane configuration, reductions in flyover EPNL below that defined by Modification 1 does not appear achievable as predicted by the methodology of Section III. A reduction in tip speed to 650 ft/sec produced only a 1 dB reduction in PNL. Further reductions in rotor noise for a rotor of this diameter (90 feet) are not apparent. Additional research is required to reduce the broadband component of large rotors.

Several cases of rotational and broadband noise evaluation showed that reduction in rotor speed alone without other rotor modifications resulted in no reduction in Perceived Noise Level for that rotor. This stemmed from an increasing average rotor lift coefficient as rotor speed was reduced having the effect of increasing broadband noise. The result was generally offsetting noise trends between rotational and broadband noise. Only when blade chord was increased, reducing average C_ℓ , did both rotational and broadband noise decrease.

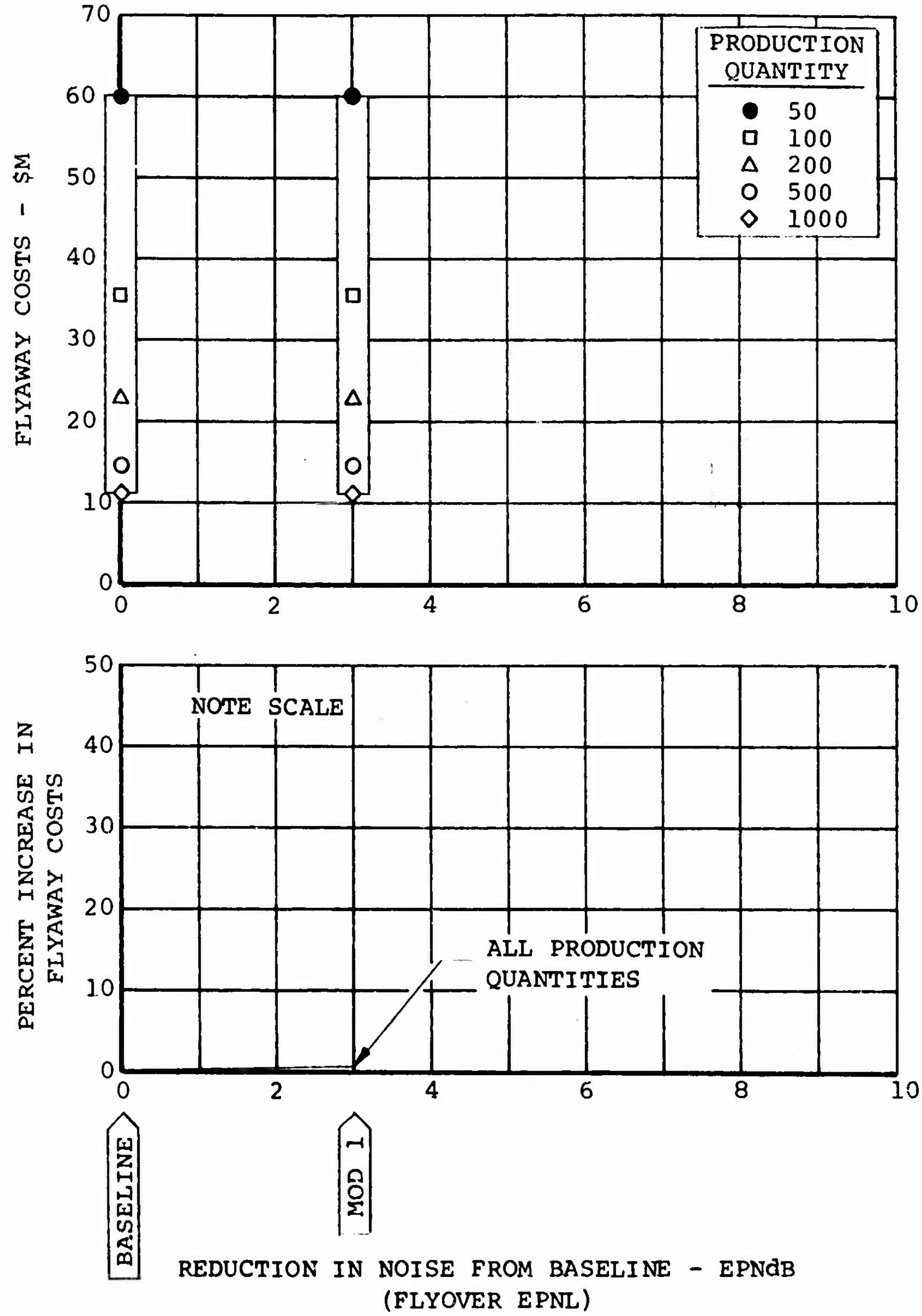


Figure 51. Effect of Configuration Changes on Flyaway Cost, Model 301 Transport

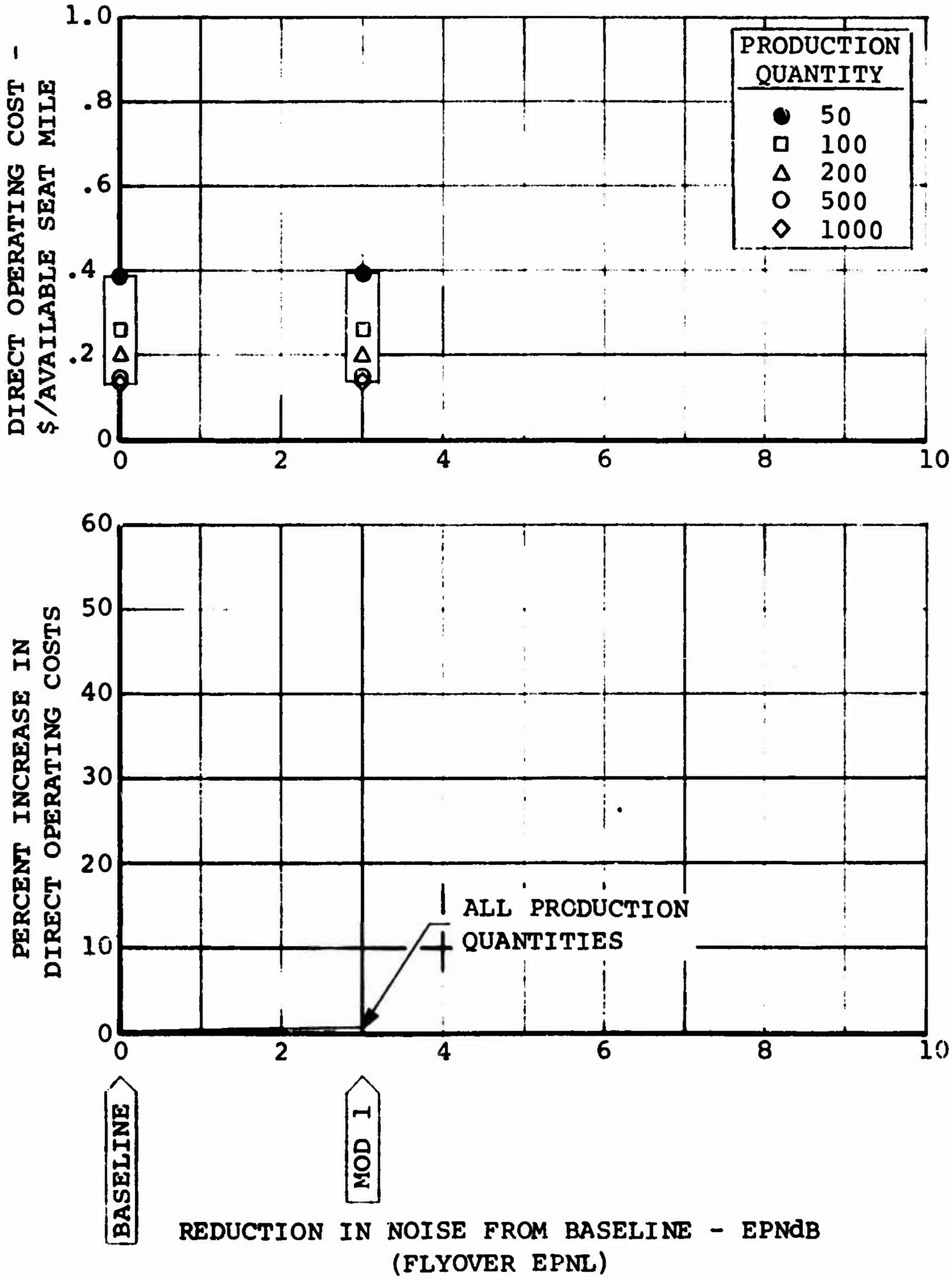


Figure 52. Effect of Configuration Changes on Direct Operating Cost, Model 301 Transport

VI. EFFECT OF NOISE STANDARDS ON ROTORCRAFT DESIGNED IN THE 1980's

Market Forecasts

Forecasting the requirements for commercial helicopters in the decade 1980-1989 requires knowledge of a number of wide-ranging factors. Assumptions must be made regarding: (1) military actions between key nations of the world, (2) inflation rates in industrial and oil-producing nations, (3) the value of the U. S. dollar with respect to other international currencies, (4) the rate of increase in the price of fuel, (5) the growth of certain rotorcraft technology areas (fuel consumption, aerodynamics, rotors, avionics), (6) the increased use of helicopters by major corporations as an element of their corporate fleets, and (7) the procurement practice of business and commercial helicopter operators with regard to replacement of helicopters currently in the corporate inventory. For this study, a forecast for commercial helicopter requirements for the period 1978-1987 prepared by Defense Marketing Systems, Inc. (11) was used to estimate the production rate of commercial helicopters.

The results of the Reference 2 study indicate that by far the largest number of units forecast is in the single turbine, under-6000-pound weight empty category. Dollar value of the larger rotorcraft remains relatively high, although total number of units to be produced is small in comparison with the smaller helicopters.

Although in the past helicopters designed for the civil market built heavily on military efforts this trend is changing for small and medium size helicopters. In a recent aerospace publication (Reference 11), the President of Sikorsky Aircraft, G. J. Tobias, presented a rationale which suggests that while helicopters with gross weights below 14,000 pounds may be developed with corporate funds to meet the civil market, civil helicopters of the larger sizes will continue to be derivatives of military models. Both categories of aircraft may require substantial additional testing, both wind tunnel and full scale to confirm noise reduction technology solutions where the helicopter is shown to be above the allowable certification levels. If, for example, tail rotor/main rotor interference dominates flyover noise, lateral offset of the tail rotor could be employed to reduce this component of the noise signature. Were this approach adopted for modifying an aircraft, wind tunnel testing would be required to confirm rotor performance and flying quality characteristics prior to the actual flight test and upon completion of a prototype vehicle, extensive flight testing would be initiated. Wind tunnel testing typically costs \$50,000 to \$70,000 per week of tunnel occupancy and flight testing may cost up to \$150,000 per week. These development and test costs would increase both the flyaway cost as well as the direct operating cost to the operator.

(11) Defense Marketing Systems, Inc., "World Helicopter Forecast to 1987", Published by DMS, Inc., 100 Northfield St., Greenwich, Conn. 06830.

Technology

It is not envisioned that there will be any breakthroughs which will lead to major noise reduction in the manner that the introduction of sound absorbing inlets and high bypass ratio engines achieved for subsonic jet airplanes. Helicopter noise reductions will probably be much more modest and result from a combination of several small changes.

Technology developments in the decade 1980-1989 which impact helicopter exterior noise will be related to the following areas;

Advanced airfoils
Reductions in drag of all forms
Composite materials
Fuel efficient engines
Noise reduction technology
Noise predictions.

The development of advanced airfoils has been paced by composite material research which is currently permitting the manufacture of complex airfoils. Development of lightweight materials will reduce total airframe weights and promote helicopters with higher forward speeds, decreasing block times and decreasing operating costs. The higher speeds will create higher advancing tip Mach numbers, however, which could result in higher levels of thickness noise. Achieving higher airspeeds without substantial noise increase is providing an impetus for the further development of transonic airfoils and blade planforms which will permit use of the higher speed rotor capabilities without accompanying increases in harmonic rotor noise.

Broadband rotor noise, which in some cases may determine the maximum value of Perceived Noise Level, is also adversely affected by high forward speed. Broadband sources, such as trailing edge noise may be reduced by modifications to trailing edge configuration made possible by composite material research. Some suggested modifications have been serration and porosity of the trailing edge as well as planform sweep near the tip region. To date, broadband noise has not been studied in the same depth as the harmonic components of rotor noise and is an area that requires greater understanding.

Reductions in rotor noise will bring about a requirement for turbine engine noise reduction on the larger helicopters. Engine noise will be a contributing component to flyover noise on approach and reductions in inlet noise will require nacelle linings similar to fixed wing transports, and the elimination of inlet guide vanes which currently produce strong tones in the acoustical signature. Exit velocities and mass flows of turboshaft engines are generally small and jet noise will not be a major noise source.

Drag reduction generally will result in reductions in broadband body noise. This source can contribute to the acoustical signature during takeoff and approach procedures. This source is not considered significant.

Transmission noise on some helicopters is a contributor to the flyover noise signature. This problem is being actively investigated for the primary purpose of reducing internal noise. Any positive results may also have beneficial effects on the external signature.

VII. CONCLUSIONS AND RECOMMENDATIONS

It is evident from the four aircraft investigated, representing six case histories of a wide range of helicopter gross weights and basic configurations, that every aircraft design affected by noise standards will be an individual case that does not submit to generalization. However, requirements to reduce helicopter noise have resulted in main and tail rotors which operate at lower tip speeds than their predecessors. This dictated that the modified rotors have higher solidity and that the drive systems were somewhat heavier. Rotors also required thin tips to accommodate higher forward speed capability. Airframes tended to be larger in order to provide greater rotor separation on tandems and main/ tail rotor separation on single rotor configurations.

In some cases, it appears that the rotor and power requirements for military helicopters may be more stringent than those required for civil operation. In those cases, some reduction in tip speed may be achieved without any physical changes to the rotor or drive systems. Once a change to an 'in-production' helicopter is required, however, the cost impact will be more severe than if the same capability had been included in the initial design.

The quantity of helicopters over which the costs of noise control can be spread is a very important factor, and appears to have a greater effect on helicopters which are already in production than on new designs. This has serious potential impact with respect to application of noise limits to derivative aircraft.

Noise levels during takeoff and approach were not investigated. They are considerably more complex to analyze from the aerodynamic as well as the acoustical aspect. Although some solutions for reducing noise in level flight are also applicable to other flight modes, there are some unique rotor-vortex interactions which have been observed on certain helicopters during takeoff and landing which may limit the noise reduction achieved in level flight. These regimes require further investigation. In addition, rotors with two blades were not investigated and the reduction of noise by reducing tip speed may have a greater impact on other characteristics of these aircraft than for helicopters with larger numbers of blades.

Flight test of unproven noise reduction techniques is required to verify some of the methodology suggested in this report. For example, reduction in tail rotor noise using advanced airfoils operating at low tip speeds and offsetting of tail rotors to achieve additional clearance between main and tail rotors discs are noise reduction methods that have been partially evaluated by model testing but need to be evaluated on full scale aircraft. Until these high technical risk methods have been adequately demonstrated, manufactureres are unlikely to incorporate them into their design.

APPENDIX A

ROTOR NOISE PREDICTION METHODOLOGY

The components of rotor noise calculated for the prediction of helicopter flyover acoustic signatures were (1) rotational, (2) broadband, (3) thickness, (4) compressibility, and (5) interaction noise. The first two of these methods had been previously programmed for machine computation and cases were run for all helicopters in the study.

Elements (3), (4) and (5) were calculated by hand from methods suggested by Pegg (Reference 3). Pegg reduced the computation complexity of the equations developed by several researchers in rotor acoustics. These elements were included, as appropriate, and summed with the rotational and broadband components to obtain estimates of the total flyover signature. The following section presents a synopsis of the equations adopted for use in this program.

Rotational Noise - The theory for this component of rotor noise was developed by Lowson and Ollerhead (12) and it forms the basis for the calculations of this element of rotor noise used in this program. Several assumptions were made to the original expression to permit a closed form solution:

$$C_n = \sum_{\lambda=o}^{\infty} K \cdot \frac{T}{Rr\ \lambda^k} \left\{ (10nM \sin\theta) J_1' - J_2' + \left(\frac{nM}{R} \cos\theta\right) J_3' \right\}$$

C_n	amplitude of nth sound harmonic at specified field point
λ	air loading harmonic number
K	constant
r	distance between rotor center and field point
n=mB	harmonic number x number of blades
M	rotational Mach number
R	radius of action of blade forces
θ	angle between disc plane and field point
J_i'	complex collection of Bessel functions of argument (nM cos θ)
$C_{\lambda T}, C_{\lambda D}, C_{\lambda C}$	thrust, drag, radial force harmonic coefficients
k	loading power law exponent
T	thrust

(12) Lowson, M. V., and Ollerhead, J. B., "Studies of Helicopter Rotor Noise", USAAVLABS TR 68-60, January 1969.

For this study, it was assumed that the thrust, drag and radial force components were randomized with respect to phase, that the ratio of the magnitude of the components ($C_{\lambda T}$, $C_{\lambda D}$, $C_{\lambda C}$) were 10:1:1, respectively, and that the harmonic airload power law constant (k) was 1.8 including the λ 0.5 term due to random phasing effects.

Broadband Noise

The broadband noise equation used for this program was based on the work of Lowson (13), Hubbard (14), Schlegel (15) and Munch (16). It was further modified to reflect an observed dependence on average lift coefficient. The spectrum peak frequency was calculated from

$$fp = -240 \log T + 0.746\, V_t + 786$$

The spectral content of broadband noise is shown in Figure A-1. One-third octave band sound pressure levels were then determined from the following equation based on rotor blades having constant chord, thickness and airfoil section along the radius:

$$SPL_{1/3} = 20 \log \frac{V_t^3}{r} + 10 \log A_b \,(\cos^2\theta + 0.1) + S_{1/3} + f(\bar{C}_\ell) - 53.3$$

where

SPL_j	sound pressure level in the jth 1/3 octave band
fp	peak frequency
T	thrust
V_t	tip speed
Ab	blade area
θ_1	angle between disc plane and field coordinate
r	distance to field coordinate
$S_{1/3}$	1/3 octave band correction from Fig. A-1
$\bar{C}_\ell$	average lift coefficient

(13) Lowson, M. V., "Thoughts on Broad Band Noise Radiation by a Helicopter", Wyle Laboratories WR 68-20, 1968.

(14) Hubbard, H. H., "Propeller Noise Charts for Transport Airplanes", NACA TN 2968.

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(16) Munch, C. L., "Prediction of V/STOL Noise for Applications to Community Noise Exposure", DOT-TSC-OST-73-19, May 1973.

Thickness Noise - Calculation of thickness noise was based on the theoretical analysis developed by Hawkings and Lowson (17). The following equation presents the harmonic sound pressure for thickness noise valid for hovering conditons:

$$P_{mB} = \frac{4}{\sqrt{2\pi}} M_t^2 \rho C_o^2 \left(\frac{R}{r}\right)\left(\frac{t}{c}\right) \int_1^\infty \frac{1}{\xi^4} \left(\frac{\sin nk\xi}{nk\xi} - \cos nk\xi\right) J_n\left(\frac{nM_t}{\xi} \cos\theta\right) d\xi$$

where:

P_{mB}	sound pressure level in harmonic mB
M_t	rotational tip Mach number
ρ	air density
C_o	speed of sound in air
R	rotor radius
r	distance between rotor center and field point
t	blade thickness
c	blade chord
ξ	$\frac{R_{TIP}}{R}$
n	mB
m	sound harmonic number
B	number of blades
k	$c/2R_t$, slenderness ratio
J_n	Bessel function of order n and argument $\left(\frac{nM_t}{\xi} \cos\theta\right)$

For estimating thickness noise levels, Pegg reduced the above expression to,

$$SPL_t = 40 \log M_t + 20 \log \frac{t}{c} + 20 \log B + 20 \log \frac{R_T}{r} + \Delta SPL_t - 0.9$$

where ΔSPL_t represents an evaluation of

$$\int_1^\infty \frac{1}{\xi^4} \left(\frac{\sin nk\xi}{nk\xi} - \cos nk\xi\right) J_n\left(\frac{n M_t}{\xi} \cos\theta\right) d\xi$$

for a matrix of values of M_t, θ and k.

(17) Hawkings, D. L., and Lowson, M. V., "Tone Noise of High Speed Rotors", Second Aero-Acoustics Conference, Hampton, Virginia, March 24-26, 1975, AIAA Paper 75-450.

Compressibility-Induced Profile Drag Noise - Prediction of compressibility noise is based on the work of Lowson and Ollerhead as modified by Arndt and Borgmann (Reference 18) who related the effect of compressibility drag on impulsive noise in the following expression,

$$P_{mB} = \frac{mB\overline{C}D_0}{4\pi^2\sqrt{2}} \frac{\Delta\psi}{\pi} \frac{R}{R_e} \frac{C}{r} \rho C_o^2 \sum_{j=-\infty}^{+\infty} (1-\frac{j}{mB})\beta j \; J_{(mB-j)}(mBM_e \sin\theta).$$

Pegg has derived a simplified form for the solution to this, assuming a drag divergence Mach number of $M_{dd}=0.8$.

$$SPL_{mB} = 20 \log \frac{R}{r} + 20 \log \left[(M_e-0.8) \frac{C}{R}\right] + \Delta SPL_c - 21.6$$

where

M_e	effective Mach number, $\frac{M_T}{1-M_f \cos\theta}$
ΔSPL_c	evaluation of the summation on the right side of the first equation
$\overline{Cd}_o$	profile drag coefficient
$\Delta\psi$	incremental azimuth angle where blade section $M > 0.8$.
β_j	Fourier coefficients in blade torque loading
j	summation index

Blade/Vortex Interaction - The component of interaction noise resulting from the intersection of trailed tip vortex filaments and rotor blades was estimated using a method proposed by Wright (Reference 19),

where $P_{mB} = (\frac{\Delta L}{L_o} E \rho_w) K_T \, mB \, \chi_s$

E	number of interactions per revolution
ρ_w	load solidity (fraction of the effective disk annulus occupied by the unsteady loading region
$\frac{\Delta L}{L_o}$	fractional steady load change per blade

(18) Arndt, R. E. and Borgman, D. C., "Noise Reduction from Helicopter Rotors Operating at High Tip Mach Number", American Helicopter Society, 26 Annual Forum, June 1970.

(19) Wright, S. E., "Discrete Radiation From Rotating Periodic Sources", Journal Sound and Vibration (1971) 17(4) 437-498.

K_T thrust constant

χ_s blade loading spectrum function,

$$= \frac{\sin\pi(ft_o-1)}{4(ft_o-1)} - \frac{\sin\pi(ft_o+1)}{4(ft_o+1)}$$

(for sine wave pulse profile)

ft_o $SE\rho_w$, (non-dimensional parameter)

s blade loading harmonic number

The simplified expression for interaction noise takes the form,

$$SPL_{MB} = 20 \log \frac{\cos\theta}{rC_o} + 20 \log \frac{\Delta L}{L_o} + 20 \log T\Omega + 20 \log (\chi_s MB\frac{\Delta\psi}{\psi_o}) + 120.6$$

where

θ angie between disc plane and observer

T rotor thrust

Ω rotational speed

$\Delta\psi$ azimuthal range of load excursion

ψ_o azimuth at intersection

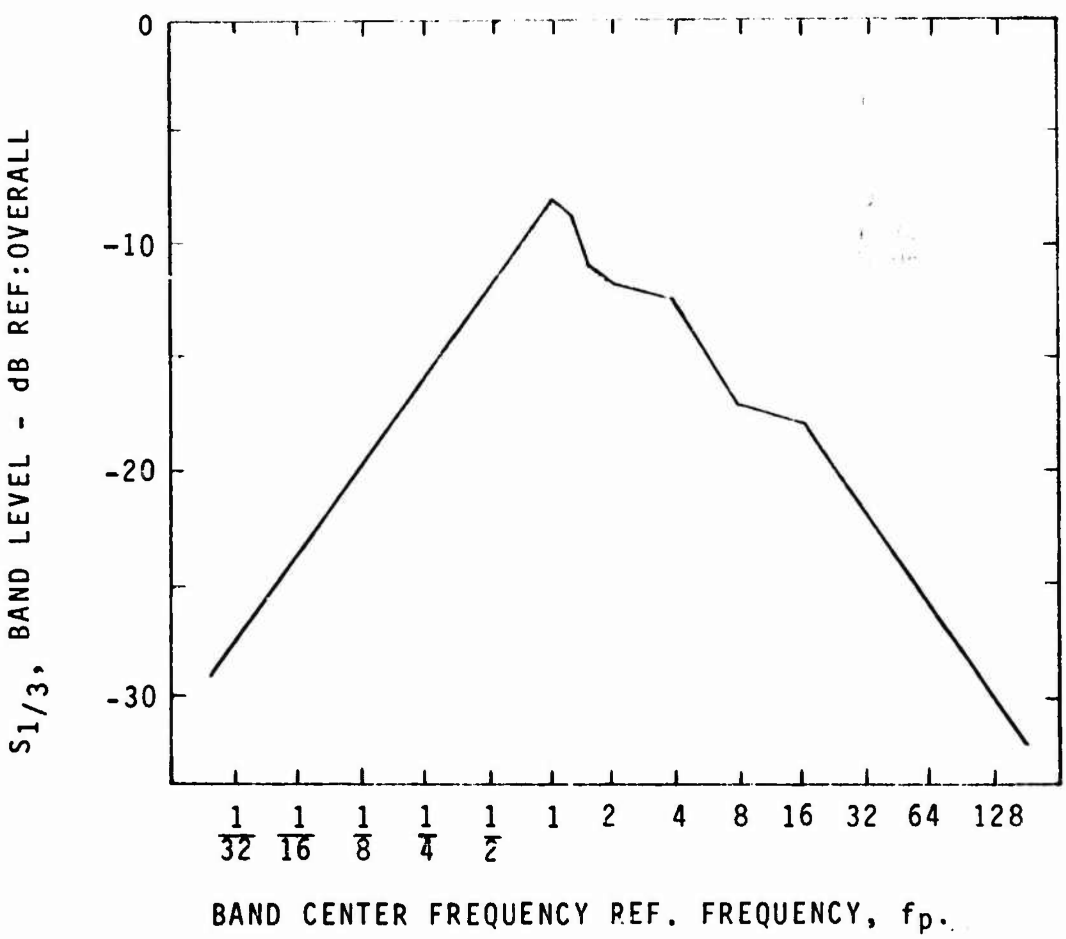

FIGURE A-1 ROTOR BROADBAND NOISE EMPIRICAL SPECTRUM

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