AD/A-005 049

FAN-IN-FUSELAGE ADVANCED ANTITORQUE SYSTEM

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Kaman Aerospace Corporation

Prepared for:

Army Air Mobility Research and Development Laboratory

November 1974

EUSTIS DIRECTORATE POSITION STATEMENT

This report has been reviewed by the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory and is considered to be technically sound. The results of the design study indicate that the application of a fan-in-fuselage antitorque system to an existing vehicle would be unsuccessful due to increased weight, unfavorable center-of-gravity change, and control complexity. However, if used on a new design, the penalties might be minimized.

The technical monitor for this contract was Mr. F. A. Raitch, Technology Applications Division.

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The information contained in this report is a result of study, analysis, and design conducted to determine the performance of a helicopter with a fan-in-fuselage antitorque and yaw control system, as compared with the performance of the same basic helicopter with a tail rotor.

A literature search and analyses were conducted to assure that all feasible techniques for airflow modulation, switching, and turning were considered, and to determine which techniques are the most efficient. Fan pitch change,

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biased contraction, and the use of thick cascade vanes were found to be the most efficient techniques for modulation, switching and turning, respectively.

An inventory tail rotor helicopter was selected as a baseline helicopter, and a preliminary design modification of this helicopter to a fan-in-fuselage configuration was developed. Performance of the fan-in-fuselage configuration was compared to the performance of the baseline tail rotor helicopter. This comparison was made for a mission derived from data on actual service operations.

The interrelationships of exit area (which is analogous to tail rotor area), duct loss, tail boom length, fan power, gross weight, and total power required in the fan-in-fuselage modification were investigated.

The fan-in-fuselage configuration provides marked improvements in flight and ground safety, dynamic component life and reliability, and vulnerability, and can be used for suppression of exhaust gas infrared radiation. The advantages are gained with a moderate performance penalty. Endurance is decreased by less than 6 percent; some reduction in this penalty may be achieved with further design refinement. The reduced airspeed and ceilings still meet requirements of the mission, which was derived from actual service experience. Control and stability characteristics, as determined by analysis, are equivalent to those characteristics in the baseline tail rotor helicopter.

PREFACE

This study was conducted by Kaman Aerospace Corporation, with the assistance of Hamilton Standard Division, United Aircraft Corporation, and Dr. Barnes W. McCormick, Jr., as subcontractors, under the terms of Contract DAAJ02-73-C-0033, Project 1F162204AA44. USAAMRDL Technical Monitor was Mr. F. Raitch. The task was begun on 9 February 1973 and completed on 31 January 1974.

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Although aerodynamic performance of the tail rotor configuration is relatively good, this configuration has serious disadvantages in several areas:

Safety - The Army accident rate due to tail rotor malfunctions has been 1.47 per 100,000 hours, and the tail rotor strike rate has been 1.37[1]. The tail rotor is a serious hazard to ground personnel.

Maintainability and Reliability - Life cycle cost for the tail rotor system of one utility helicopter is reported to be $10.64 per flight hour, and for one attack helicopter, $17.85 per flight hour[2].

Vulnerability - The tail rotor is vulnerable to small arms fire and cannot be armored.

Detectability - The tail rotor contributes significantly to both radar cross section and noise.

Two other antitorque systems, each of which reduces or eliminates these disadvantages, have been investigated[3,4]. In one of these (Figure 1) a shrouded fan is installed in the vertical fin. This configuration is safer than the tail rotor; ground personnel cannot walk into it, although a careless person could put his hand or arm into it, and the danger of damage from tree branches is reduced. Since the fan, like the tail rotor, is subject to nonaxial flow but is not articulated, blade root failure loads are high; there is no improvement in reliability. Since the tip speed is higher[4], high frequency noise will be increased. The vulnerable area is reduced by 17%, but armoring is still not possible (as seen in side view).

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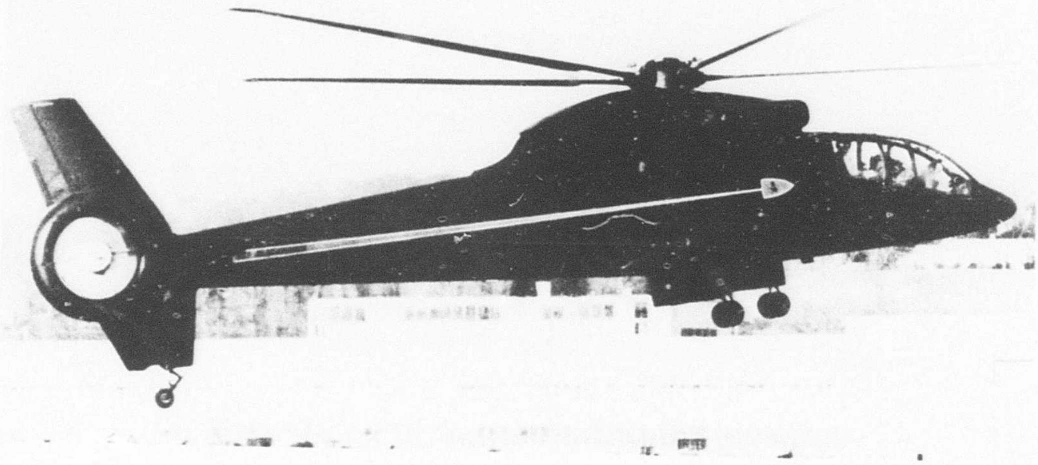
[1]Letter, USAAS, Ft. Rucker, Ala., To Kaman Aerospace Corporation, Bloomfield, Conn., dtd 5 February 1964.

[2]Knudsen, George E., and Carr, Patricia V., DATA ANALYSIS OF THE UH-1/AH-1 TAIL ROTOR SYSTEM, Bell Aircraft Company, Fort Worth, Texas, USAAMRDL TR 74-11.

[3]Velazquez, J. L., ADVANCED ANTITORQUE CONCEPTS STUDY, Lockheed-California Company, USAARMDL Technical Report 71-44, Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, August 1971, AD 731493.

[4]Grumm, Arthur W., and Herrick, Groves E., ADVANCED ANTITORQUE CONCEPTS STUDY, Sikorsky Aircraft Division of United Aircraft Corporation, Stratford, Conn., USAAMRDL Technical Report 71-23, Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, July 1971, AD 729860.



SIKORSKY S-67 MODIFIED FOR FAN-IN-FIN
EVALUATION TESTING

AEROSPATIALE SA-341 (OPERATIONAL)

Figure 1. Fan-in-Fin Helicopters.

In a second alternative, sometimes referred to as the fan-in-tailcone but here called the fan-in-fuselage configuration, a fan installed in the aft end of the fuselage or the forward end of the tail boom drives air to the end of the tail boom, where it is turned 90 degrees to the left or the right to provide the required antitorque and yaw control thrust. Early examples of this configuration are the Hiller J-5 and the British Cierva W-9, built in the mid-40's (Figure 2).

The fan-in-fuselage configuration reduces or eliminates all of the cited disadvantages of the tail rotor configuration:

Safety - All moving parts are enclosed in fixed structure. The probability is that only an impact which wiped out the entire tail boom would make the controls inoperative.

Maintainability and Reliability - Inflow to the fan, with proper inlet design, is axial in all flight conditions. Fatigue loads are low.

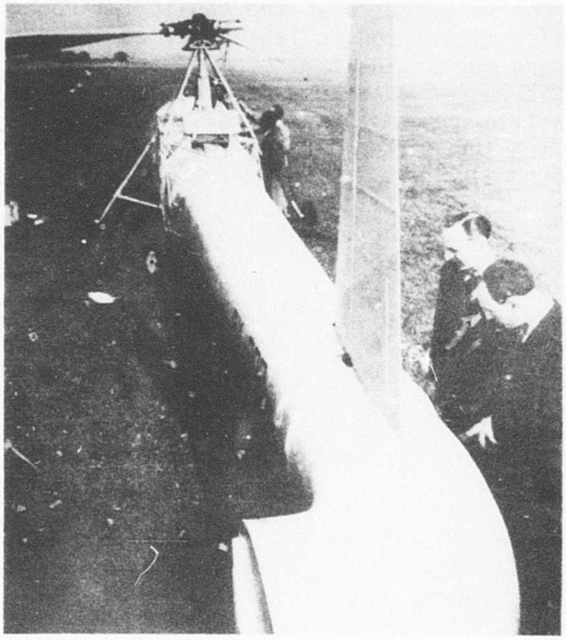
Vulnerability - Fan vulnerable area in the Y-Z plane is very low, and the fan can be armored against small arms fire which does not have a large longitudinal component. The vulnerability of other system elements is low.

Detectability - Since the fan is enclosed, acoustic insulation can be provided to control whatever noise it may produce, and no rotating parts are visible to radar.

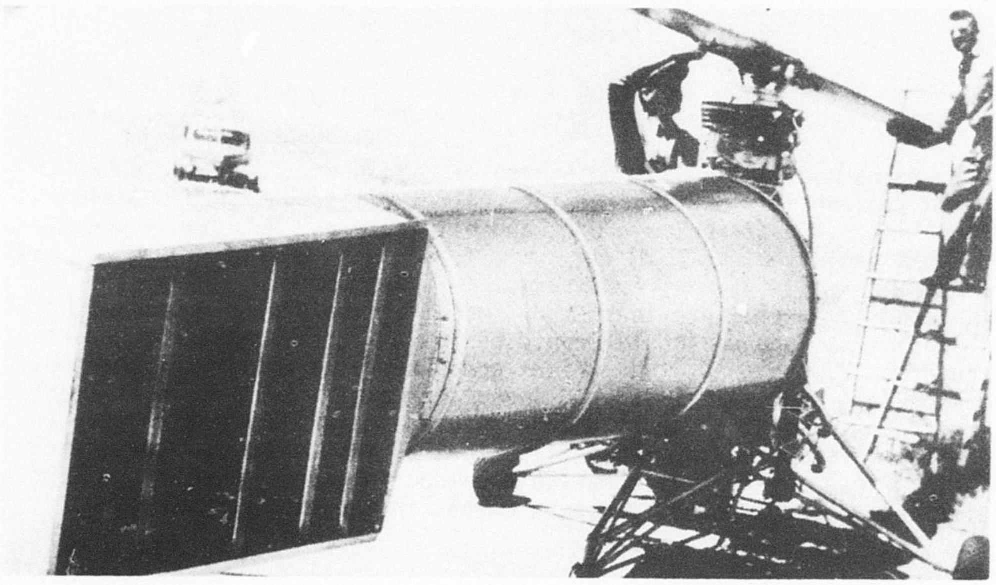
A design feature not specified under the contract can give the fan-in-fuselage one additional advantage:

Infra-Red (IR) Radiation Control - Mixing of the engine exhaust with the airflow required for control will reduce exhaust plume temperature to below 300°F, and eliminate any detectable plume IR radiation.

The conclusion of a previous study[4] was that the design gross weight of a fan-in-fuselage helicopter would be 15.8% higher, and the power required would be 47.6% higher, than weight and power of a tail rotor helicopter designed for the same mission. These increases resulted, in at least a large part, from the use of a fan-in-fuselage exit area which gave an equivalent disc loading seven times that of the tail rotor considered. The fan-in-fuselage configuration does not inherently require so high a disc loading (301 pounds per square foot in the case referred to).



British Cierva W-9



Hiller J-5

Figure 2. Early Fan-in-Fuselage Helicopters.

The purpose of this study has been to determine optimum techniques for airflow modulation, switching, and turning; to select an inventory tail rotor helicopter (subsequently known as the baseline helicopter) with well-established actual performance characteristics; to design a modification of this helicopter to a fan-in-fuselage configuration; and to compare the aerodynamic and control and stability performance of the fan-in-fuselage (modified) helicopter with that of the tail rotor (baseline) helicopter.

For purposes of aerodynamic performance comparison, a mission corresponding to actual service experience was derived. Changes required to make the fan-in-fuselage configuration match the speed, endurance, and ceiling of the baseline helicopter were not considered. Such changes would have required changes to the basic dynamic system, and thus, in the writers' judgment, would have exceeded a practical degree of modification.

A secondary purpose has been to determine if any advantage could be gained by using the fan-in-fuselage system for forward thrust augmentation.

GENERAL DESCRIPTION OF SYSTEM

The fan-in-fuselage antitorque and yaw control system functions, as do the fan-in-fin and tail rotor systems, through the thrust reaction to an air jet directed laterally from the end of a tail boom. In powered flight, the required thrust is unidirectional and varies in magnitude as a function of rotor power, altitude, and yawing moment. In autorotational flight, when rotor torque is very low, the required thrust may be either to the left or to the right and varies in magnitude as a function of the yaw moment required.

The basic elements of the system are:

1. Air inlet, aft end of fuselage or forward end of tail boom.
2. Fan or blower, aft of inlet.
3. Duct.
4. Airflow modulation device(s).
5. Airflow switching device, left/right.
6. Airflow turning device, 90 degree, left and right.
7. Air exits, left and right.

Rotor torque is given by the basic expression

$$M_r = 550 \quad P_r/2\pi N \tag{1}$$

where P_r is rotor power required, and N is rotor speed, in revolutions per second. Antitorque thrust is then

$$T_t = 550\ P_r/2\pi NR \tag{2}$$

in which R is the perpendicular distance from the centroid of the fan-in-fuselage duct exit jet to the center of gravity (cg) of the helicopter.

Maximum yaw control thrust is established to give the yawing displacement required at the end of one second under Paragraph 3.3.5 of Specification Mil-L-8501A, given gross weight equal to W:

$$\theta = 330/\sqrt[3]{W + 1000}\ \text{(degrees)} \tag{3}$$

Assuming constant acceleration

$$\theta/57.3 = \alpha/2 = T_c R/2I_y \tag{4}$$

where

α = angular acceleration, radians per second

T_c = maximum yaw control thrust required, pounds

I_y = helicopter yawing moment of inertia about the cg, slug-feet^3/sec

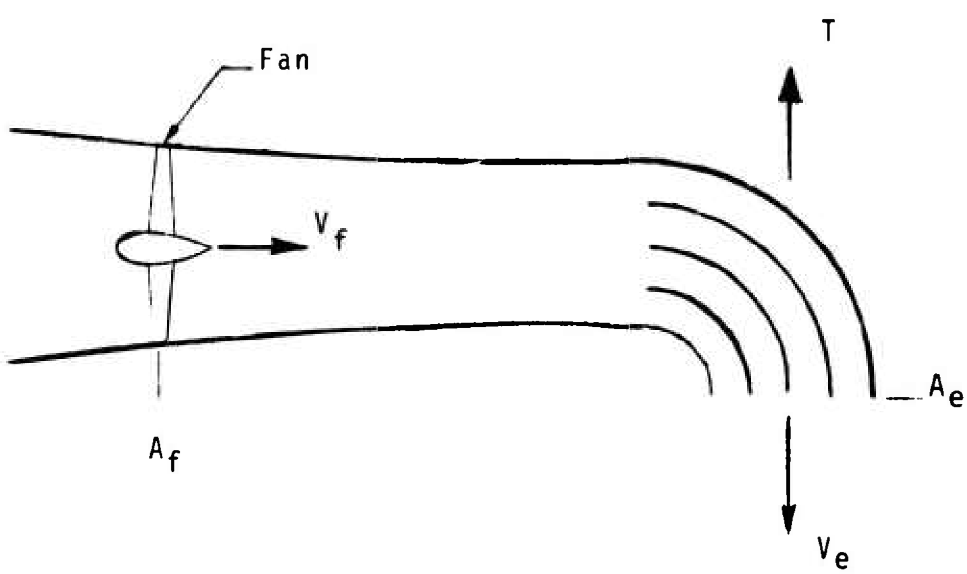
Then

$$T_c = 11.52\ I_y/R\ \sqrt[3]{W + 1000} \tag{5}$$

And maximum lateral thrust required is

$$T_m = T_t + T_c \tag{6}$$

Fan power in hover is determined from basic momentum and energy relationships, as follows:



Let

A_n = cross-sectional area, ft^2, at point n

k_n = duct loss factor at point n

n_f = fan efficiency

m/t = mass flow rate, slugs per second (sl/sec)

V_f = air flow velocity at fan, ft/sec

V_e = outlet (jet) velocity, ft/sec

A_f = cross-sectional area at fan, sq ft

A_e = cross-sectional area at exit, sq ft

P_e = energy output, HP

P_f = fan power required, HP

T = lateral thrust, lb

V_n = airflow velocity at point n, ft/sec

ρ = air density, slugs/ft^3

From the momentum change

$$T = (m/t)V_e \tag{7}$$

$$= \rho A_e V_e^2 \tag{8}$$

And for a required thrust

$$V_e = \sqrt{T/\rho A_e} \tag{9}$$

The energy of the jet flow is

$$P_e/550 = 1/2(m/t)V_e^2 \tag{10}$$

$$= 1/2 \rho A_e V_e^2 \tag{11}$$

$$= T_e^{1.5}/2\sqrt{\rho A_e} \tag{12}$$

$$P_e = T_e^{1.5}/1100\sqrt{\rho A_e} \tag{13}$$

Validity of the foregoing relationships depends on fulfillment of two conditions:

1. Exit flow must be in a pure lateral direction, with no convergence or divergence. Fulfillment of this requirement will depend on the detail design of the turning vanes and the exit.

2. Detail design of the inlet lip must be such that there is no stagnation point or "dead" air in the duct.

It is seen that power required for a given thrust decreases as the "disc loading" - that is, the ratio of thrust to exit area decreases.

Duct losses are determined by summing discrete pressure losses, p_n, which appear as increases to the fan load, p_f. Empirical loss factors, k_n (see Airflow Control Elements, page 18 and subsequent), are applied to the local dynamic pressures, q_n. Total pressure loss is:

$$p_e = \sum_1^n k_n q_n \tag{14}$$

$$= 1/2\, \rho \sum_1^n k_n V_n^2 \tag{15}$$

$$= 1/2\, \rho V_e^2 \sum_1^n k_n \left(\frac{A_n}{A_e}\right)^2 \tag{16}$$

Fan power required to overcome this loss is

$$P_e/550 = p_e A_f V_f \tag{17}$$

$$= p_e A_e V_e \tag{18}$$

$$= 1/2\, \rho A_e V_e^3 \sum_1^n k_n \left(\frac{A_n}{A_e}\right)^2 \tag{19}$$

Total fan power, if fan efficiency is η_f, is then

$$P_f = (T^{1.5}/1100\, \eta_f \sqrt{\rho A_e}\,)\left[1 + \sum_1^n k_n \left(\frac{A_n}{A_e}\right)^2\right] \tag{20}$$

If we let

$$K = \sum_{1}^{n} k_n \left(\frac{A_n}{A_e}\right)^2 \tag{21}$$

Then

$$K = \frac{p_e}{q_c} \tag{22}$$

And

$$P_e = (T^{1.5}/1100\ n_f \sqrt{\rho A_e})(1 + K) \tag{23}$$

It is not intended in this report to discuss fan efficiency; but it should be noted in passing that use of a low pressure rise (consistent with the desirability of low exit disc loading), of low flow - through velocities, and of a small tip clearance are all highly desirable.

Fan power for a given rotor power thus varies inversely as $R^{1.5}$ (R being the distance from the cg to the center of the air exit area), and as $\sqrt{\rho A_e}$, while rotor power, in hover at constant altitude, varies directly as gross weight. For a given R, a smaller A_e gives a lower gross weight, but also a higher fan power required and a larger duct loss.

In forward flight, antitorque and yaw control may be provided using either airflow reaction thrust, or a fin and rudder. Power required in either case is discussed below, under "Airflow Control Elements", page 18.

FORWARD THRUST

It would be feasible to use the fan-in-fuselage with an exit directed aft to provide thrust in forward flight, but, for pure helicopters, it would be neither practical nor advantageous.

A number of conceptual arrangements for providing forward thrust were considered. Any such arrangement greatly increases control complexity and weight, with unfavorable effect on center of gravity.

Power required in such an arrangement is considerably higher than that required when thrust is obtained from the rotor, so long as the rotor is not operating in partial stall. In the case of the OH-6 (FIF), drag at 110 kts is 246 lb, and rotor power to overcome this drag is 87.4 HP, assuming a rotor efficiency of 95%. Power required to provide the same thrust from the fan-in-fuselage, given a 9.2-sq-ft exit, would be 112.4 HP.

Such an approach could only be justified as a (possibly inefficient) means of increasing speed beyond the pure helicopter regime.

AIRFLOW CONTROL ELEMENTS

Available technical literature (See Bibliography) was reviewed to determine loss factors for the various turning, modulating, and switching techniques. Appendix A lists documents which include possibly pertinent data and indicates the categories of such data. The sources of data actually used are separately cited in the body of this report.

Designs considered were functionally categorized as turning, modulating, and switching. Turning devices (Figure 3) include:

1. Duct elbows

   a. No vanes
   b. Thin vanes
   c. Thick vanes

2. Airfoils

   a. Pivoted
   b. Mechanical flap
   c. Jet flap
   d. Blown flap
   e. Boundary layer control
   f. Circulation control
   g. Elastic chord

3. Thrust reversers and deflectors

The minimum loss factor for a 90° turn occurs in an elliptical duct with an elbow, and may be as low as .04 for a radius/thickness ratio of 4, with Reynolds Number (RN) = 600,000[5]. This shape and r/t ratio are not appropriate to the application, though, as will be apparent through examination of the actual design.

Loss factors for other designs for which data is available are shown in Figure 4. As is shown, thick cascade vanes are the most efficient, with K = .05 at RN = 180,000. The actual RN proved to be 600,000 approximately. Reduction in K for RN > 180,000 is very small.

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[5] Aerospace Applied Thermodynamics Manual, Society of Automotive Engineers, Inc., New York, N.Y., Rev. January 1962.

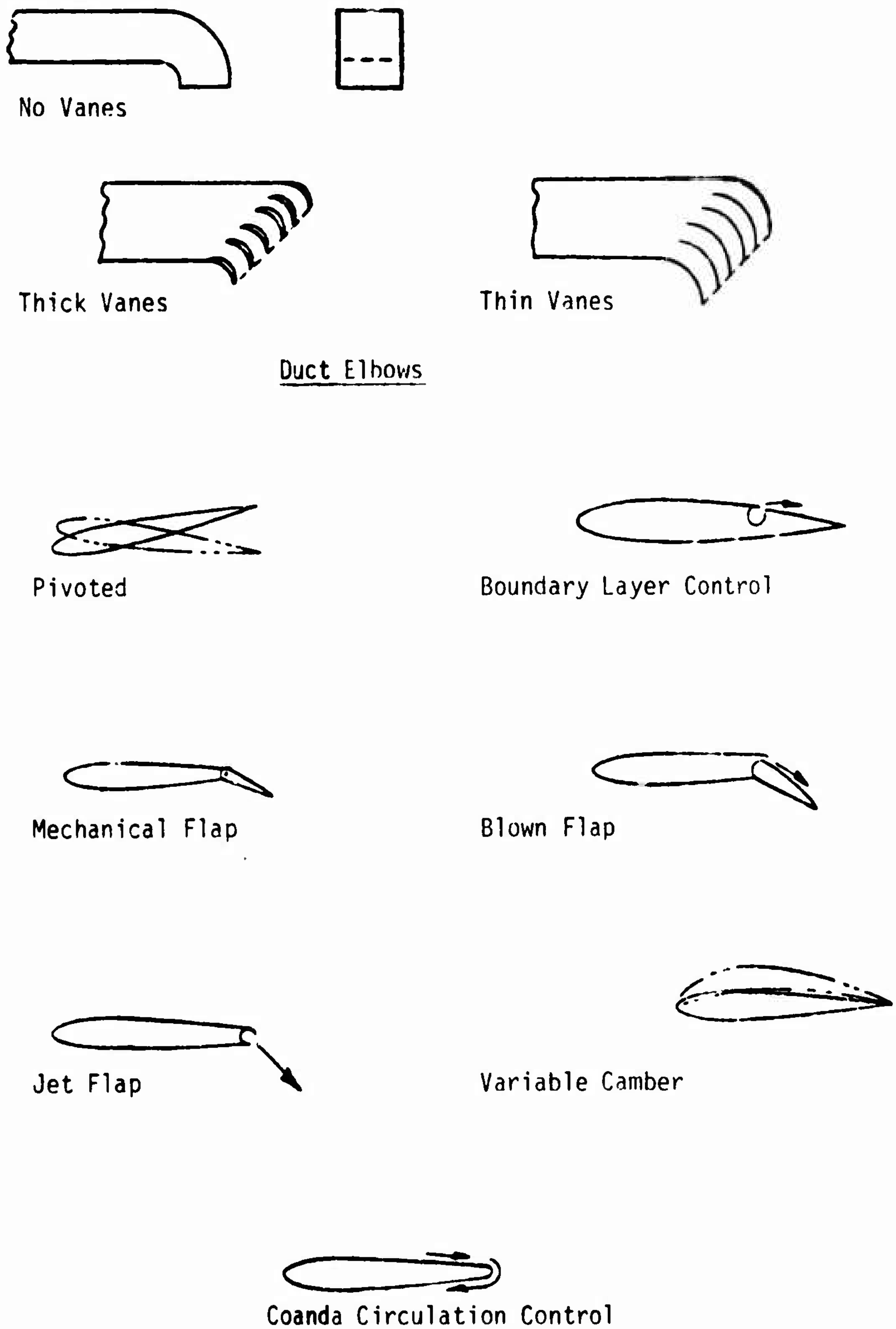

Figure 3. Turning Techniques.

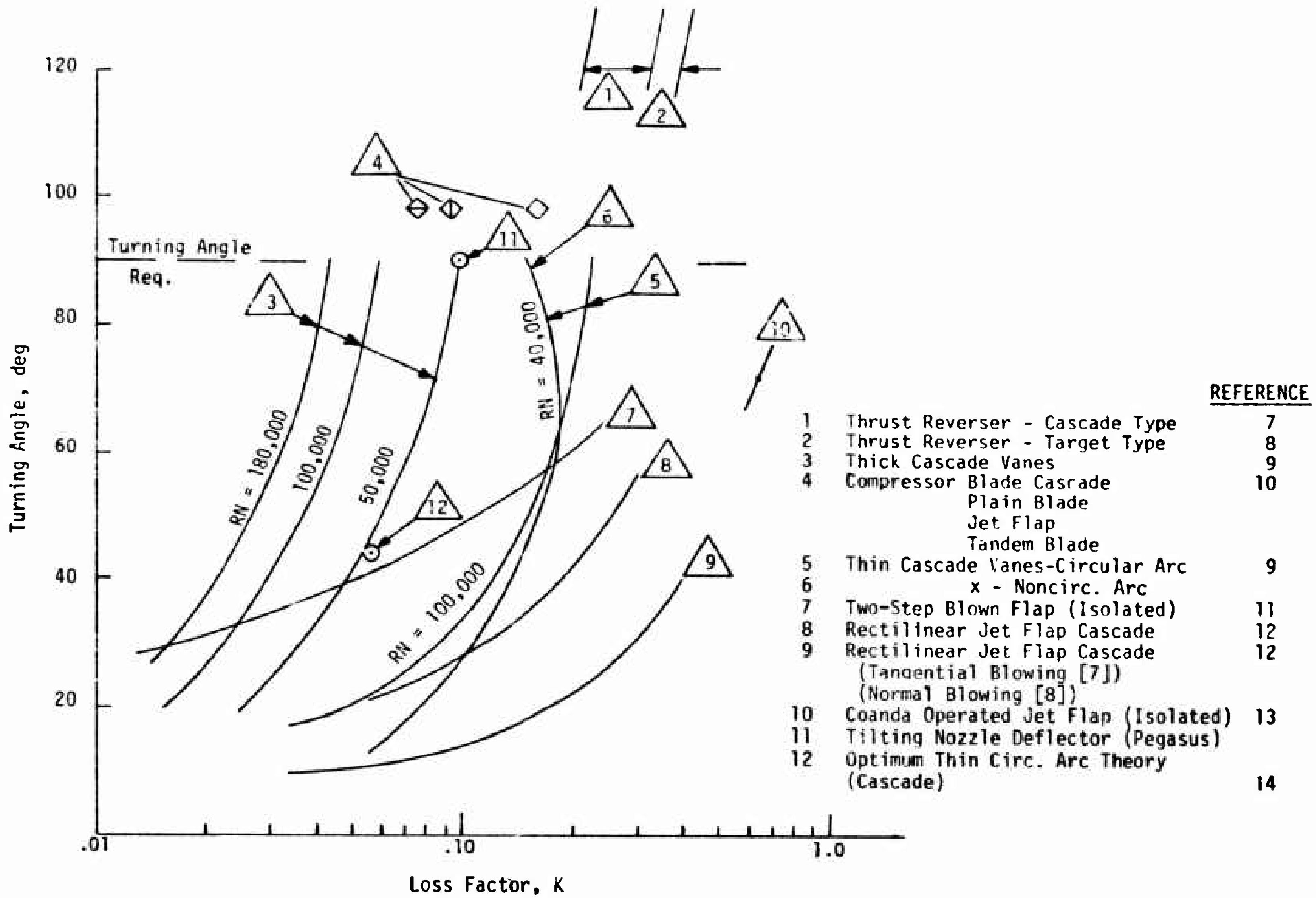

Figure 4. Turning Devices - Variation of Loss Factor With Turning Angle.

Designs not shown in Figure 4 can be appraised only in a qualitative manner. A pure airfoil, although efficient, is severely limited in its turning angle capability; and we do not know how it would perform in a duct. The circulation control blade, although effective as a high lift device, is less efficient than a simple airfoil[6] (or Figure 8 of reference). Airfoils using fluidic augmenting techniques in general are assumed to have efficiencies which, at best, are not significantly better than that of the two-step blown flap as shown in Figure 4.

Airflow can be modulated at its source (by varying fan blade pitch), by throttling, or by dumping unwanted flow overboard (Figure 5). The last approach is obviously inefficient. The efficiency of a variable pitch fan is discussed below (Fan Design, page 38). Loss factors for various throttling techniques are shown in Figure 6; the most efficient being the flapper valve. Use of a jet curtain as the equivalent of a mechanical vane, with possible recovery of the jet momentum, was investigated[15]. The efficiency of this technique is given in Table 1.

TABLE 1. EFFICIENCY OF JET-THROTTLED AIRFLOW AT A CONSTANT FLOW RATE

Contraction Ratio	Efficiency
.792	.216
.709	.202
.625	.181
.542	.164

Switching techniques, as shown in Figure 7, are biased contraction, differential splitting, and induced flow diversion. Pressure losses in the differential splitter are low when the total flow is in one direction; the only loss is that due to rapid expansion of the fraction of the flow that enters a partially open channel. On the other hand, part of the total flow is being dumped except when total flow is in one direction only. This would increase total power required in the ratio of $(T_t + T_c)/T_t$; in the case actually used, power increased by twenty five percent. Further, the differential splitter is larger for comparable exit areas than the other possible designs.

[6] Williams, Robert M., and Rogers, Ernest O., DESIGN CONSIDERATIONS OF CIRCULATION CONTROL ROTORS, Aviation and Surface Effects Department, Naval Ship Research and Development Center, Bethesda, Maryland, 28th Annual National Forum, American Helicopter Society, Washington, D.C., May 1972.

[15] McCormick, B.W., Jr., AN EXPERIMENTAL STUDY TO DETERMINE THE FEASIBILITY OF DEFLECTING A DUCTED FLOW BY MEANS OF A SECONDARY JET, State College, Pa. (Unpublished).

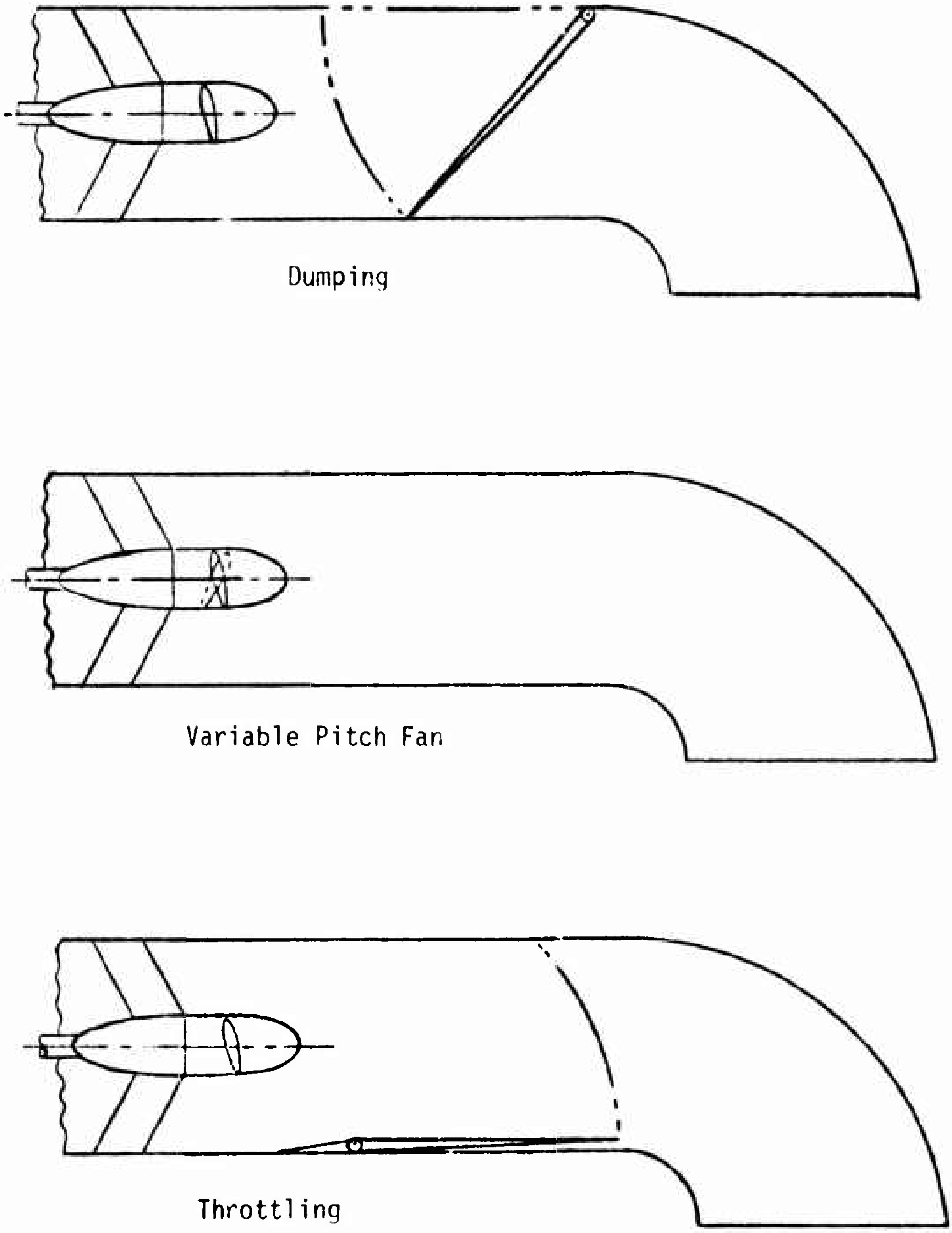

Figure 5. Modulation Techniques.

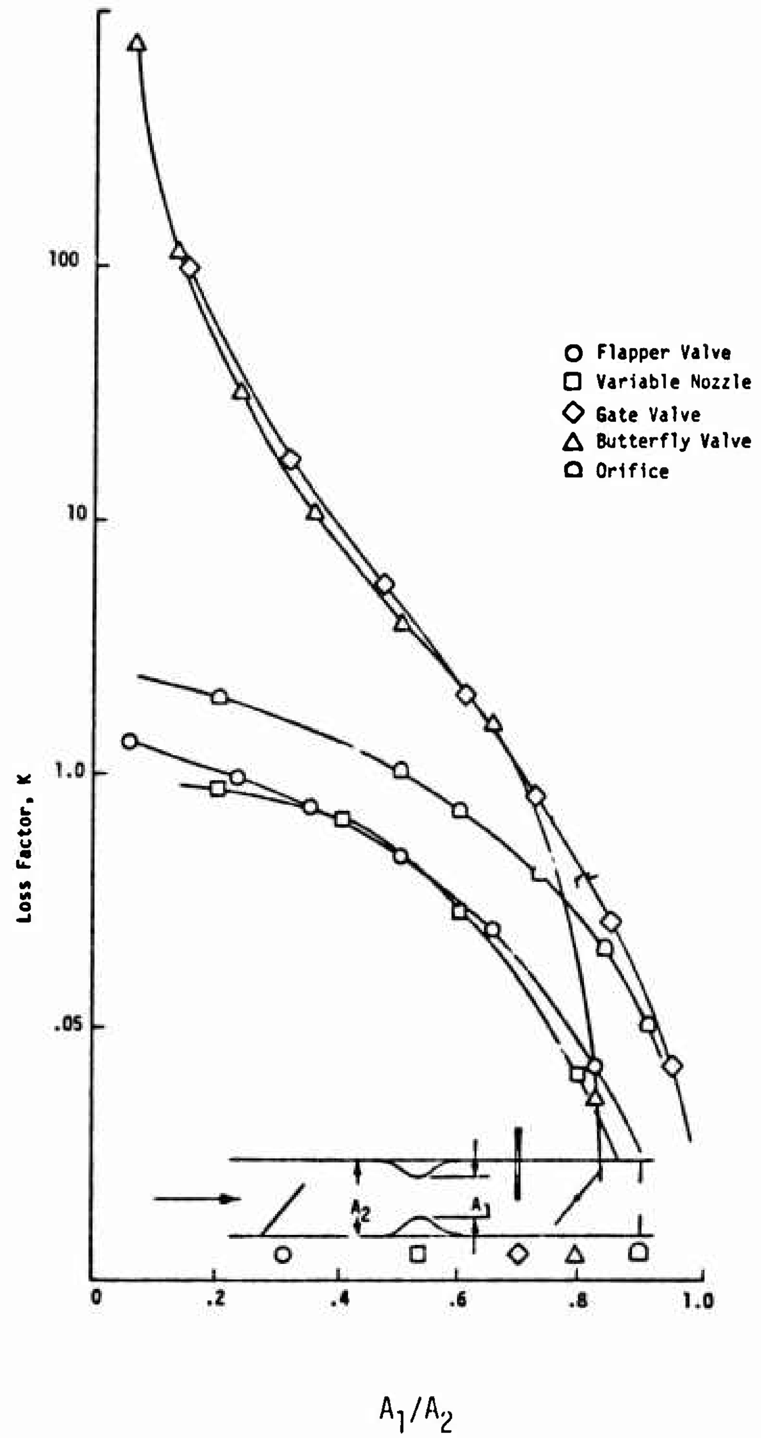

Figure 6. Throttling Devices - Variation of Loss Factor With A_1/A_2 (Reference 7).

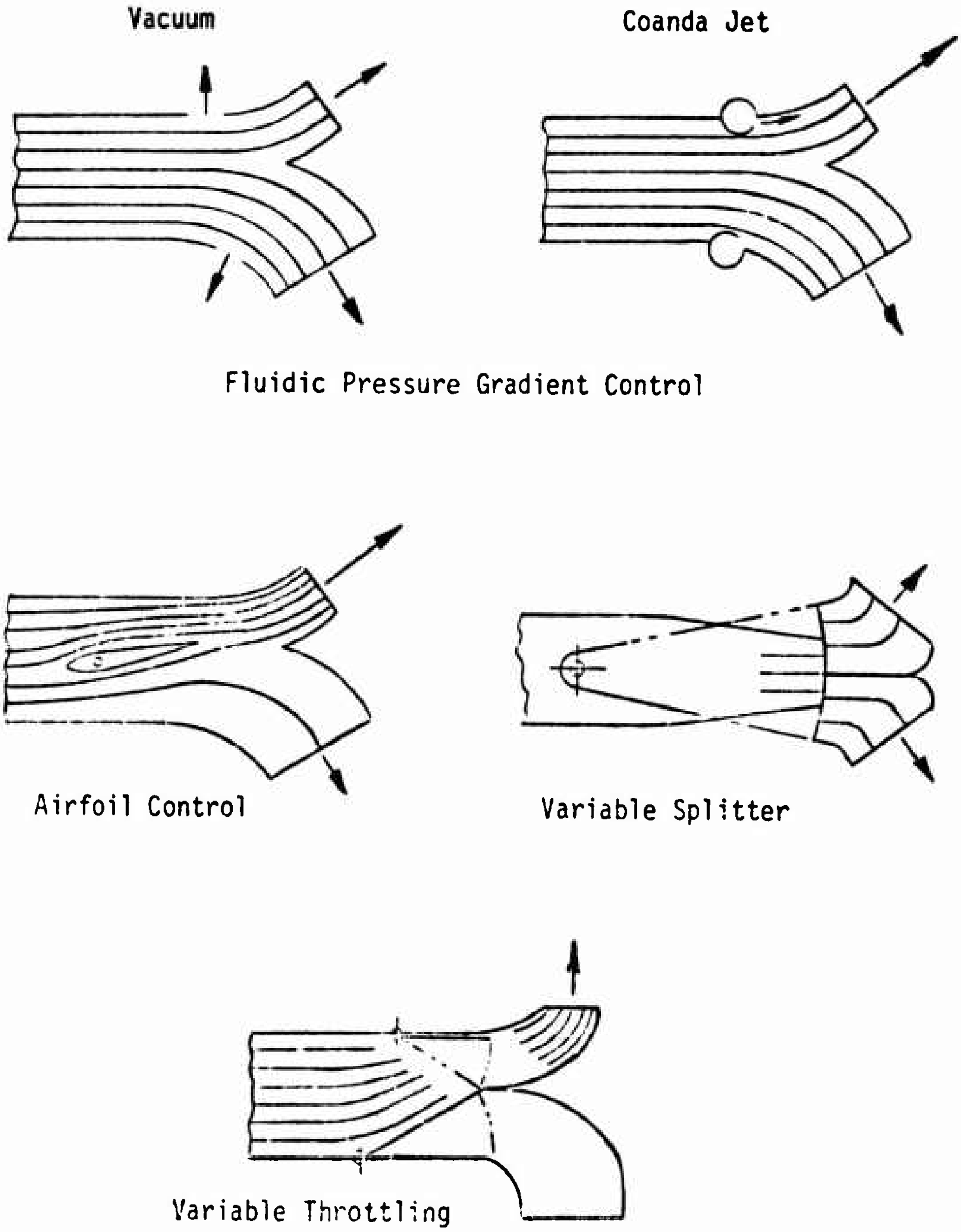

Figure 7. Switching Techniques.

The losses in biased contraction are small. With a 30 ramp angle, K = .05[5], and will be smaller if the ramp angle can be reduced.

Switching may be induced through either mechanical or fluidic techniques. The potential efficiency of a mechanical technique, in which an airfoil upstream of the Y diverts the airflow to the left or to the right, is not known, although it could be experimentally determined. Fluidic switching, Figure 7, is commonly used in servo-control systems, but not in any application involving the mass flow rates (about two slugs per second) and Reynolds Numbers (600,000 or over) required in the fan-in-fuselage. The only data[16] known to the writer indicates that the efficiency of a fluidic switching device, similar to one shown in Figure 7, is about 60%. The mass flow rate in this case is 0.12 slug per second; about 6% of that required here. An informal study by Minneapolis Honeywell, under the direction of Dr. Raymond Rose, concluded that efficiency would be about 60% for a symmetrical Y, but only about 20% for the asymmetrical Y required here[17].

The final selection of turning techniques was thick cascade vanes, and of biased contraction for switching. The choice for modulation was deferred, pending control response investigation. Duct loss factor was estimated as 0.20 for purposes of initial design investigation.

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[16] Campagnuola, Carl, et al, A STUDY OF TWO EXPERIMENTAL FLUIDIC GAS DIVERTER VALVES, THIRD CRANFIELD FLUIDIC CONFERENCE, Turin, Italy, Paper Cl; Army Materiel Command, Harry Diamond Laboratories, Washington, D.C., May 1968.

[17] Letter, Dr. R. E. Rose, Honeywell, St. Paul, Minnesota, to G.W. Carson, Kaman Aerospace Corporation, Bloomfield, Conn., 20 April 1973.

BASELINE HELICOPTER SELECTION

In order to assure valid evaluation of fan-in-fuselage performance as compared to tail rotor performance, an existing helicopter was selected for preliminary design modification to the fan-in-fuselage configuration.

The requirement for data availability led to restriction of the candidate list to in-service U. S. military helicopters, while consideration of costs (in the event of a future experimental prototype program) indicated the advisability of using a small or medium helicopter. The following were considered: OH-6A, OH-58A, UH-1D and SH-3A. Several helicopters in the same general gross weight range were deleted from consideration for anticipated availability reasons.

Maximum fan power required for each of these helicopters was then estimated on a preliminary basis. Exit area was defined as the area of a circle tangent to the top line of the tail boom, and to a 5° flare angle ground clearance line, centered at the tail rotor station line. Fan radius was taken as equal to the maximum half breadth, less 4 inches of wall thickness allowance. Since relative values only were desired, fan efficiency was taken (in this step only) as 1.00. The basis of calculation is given in Equations (1) through (23).

The OH-6A offered the prospect of a minumum ratio of fan power to total power (see Table 2). Antitorque power required is basically low because of a relatively high main rotor speed, and the OH-6A geometry permits a relatively large exit area, given exit limits as defined above. Further investigation led to the conclusion that there would be no serious problem in modifying the OH-6A, if modification were later required.
Further, the OH-6A is one of the two smallest candidate helicopters, and modification costs would be relatively low.

The OH-6A was therefore selected as the baseline helicopter.

TABLE 2. BASELINE HELICOPTER SELECTION DATA

	OH-6A	OH-58A	UH-1D	SH-3A
Exit Diameter, ft	3.5	2.60	4.7	7.00
Fan Diameter, ft	3.5	3.75	4.7	9.33
Exit Area, A_e, ft	9.23	4.32	17.35	38.50
Fan Area, A_f, ft	9.23	11.04	17.35	68.50
A. Sea Level, Standard Day, Max Gross Weight				
Gross Weight, W lb	2700	3000	9500	20,000
Yaw Moment of Inertia, I_y slug-ft^3	884	1505	10795	47,300
Center of Gravity to Exit Distance, R ft	15.83	19.54	26.58	36.67
Yaw Control Thrust, Max, T_c lb	41	56	213	538
Rotor Speed, N rpm	483	354	324	203
Rotor Power Required, P_r HP	272	252	1090	2285
Antitorque Thrust Required, T_T lb	187	191	665	1613
Total Lateral Thrust, Max, T lb	228	247	878	2151
Exit Velocity, V_e fps	99.8	140	146	153
Fan Power (Less Losses), P_f HP	20.7	31.5	116.5	298
Fan Power/Rotor Power	.076	.125	.107	.130
B. 4000 Ft, 95°, Gross Weight as Limited by Power Available (ρ = .00192)				
Gross Weight, W lb	2190	2280	7070	15,770
Yaw Moment of Inertia, I_y slug-ft^3	15.83	19.54	26.58	36.67
Center of Gravity to Exit Distance, R ft	717	1144	8034	37,296
Yaw Control Thrust, Max, T_c lb	35	45	173	459
Rotor Speed, N rpm	483	354	324	203
Rotor Power Required, P_r HP	223	211	780	1765
Antitorque Thrust Required, T_T lb	153	160	476	1246
Total Lateral Thrust, Max, T lb	188	205	649	1705
Exit Velocity, V_e fps	101	142	140	152
Fan Power (Less Losses), P_f HP	17.3	26.5	83.1	236
Fan Power/Rotor Power	.078	.126	.106	.134

FAN-IN-FUSELAGE PRELIMINARY DESIGN

AIRFLOW CONTROL SYSTEM

The selection of thick cascade vanes for turning and biased constriction for switching, left open the selection of a modulation technique. From a power required standpoint, changing fan pitch is the most efficient means of modulation. Power required comparisons are given in Table 3.

Modulation through fan pitch change presents no problem in application to slow-response control changes - specifically, power changes resulting from changes in gross weight, speed, and entry into climb or descent. Such modulation is, though, accompanied by a lag of about 0.6 second in the thrust response at the exit, if linear programming is used. Such a lag will give unsatisfactory characteristics in yaw control. Use of the "quickening" technique, discussed below under "Control and Stability", page 61, will reduce this lag to acceptable limits. The alternative, use of throttling at the aft end of the duct, requires maintenance of the fan at a constant thrust level which, in autorotation, would increase the sink speed by about 160 fpm. Modulation by varying fan pitch is thus the only acceptable technique.

Since the thrust required in hover is unilateral, while that required in autorotation alternates symmetrically, a minimum of two exits is required, that on the right being for yaw control flow only, while that on the left must be sized for antitorque and maximum yaw control combined. Control design considerations led to the use of three: a main antitorque exit and two yaw control exits. In hover and at low speeds, the left yaw control exit is used with the main antitorque exit, for maximum area and minimum power; while in autorotation the main antitorque exit is closed, and the symmetrical yaw control exits are used. See Figure 8.

In flight at 60 knots and over, sufficient moment about the vertical axis through the center of gravity can be developed with a rudder (Figure 9). Power required was analyzed for two control methods for this regime:

1. Antitorque and yaw moments provided by the fan-in-fuselage airflow system.

2. Antitorque and yaw moments provided by a rudder; fan-in-fuselage system is blocked, with fan at "flat" pitch.

As shown in Figure 10, the second of these approaches requires the least power at speeds over 60 knots; this approach was adopted.

TABLE 3. FAN POWER REQUIRED USING VARIOUS MODULATION TECHNIQUES

GW, lb	Pedal Position	Thrust, lb	Fan Power Required, HP			
			Pitch Ctr.(1)	Pitch Ctr.(2)	Dumping	Throttling
2400	Max. Left	180	19.5	19.5	22.5	19.5
2400	Neutral	140	19.5	14.0	22.5	15.5
2400	Max. Right	100	19.5	8.3	22.5	12.0
1800	Max. Left	140	12.3	14.0	12.8	15.5
1800	Neutral	100	12.3	8.2	12.8	12.0
1800	Max. Right	60	12.3	4.0	12.8	8.0

(1) Pitch controlled for weight changes only

(2) Pitch controlled for weight changes and yaw control

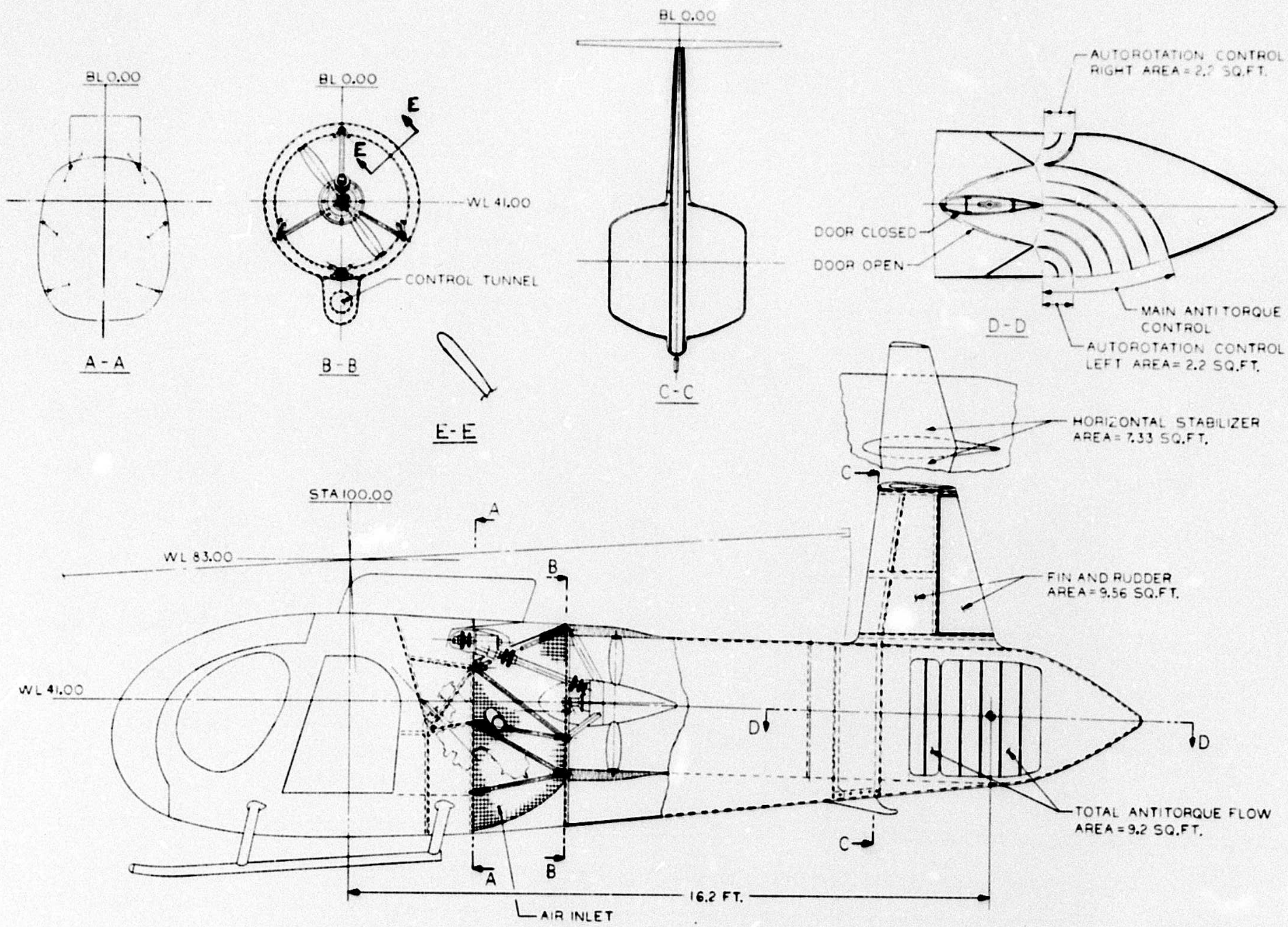


Figure 8. Fan-in-Fuselage Modification of the OH-6A - General Arrangement.

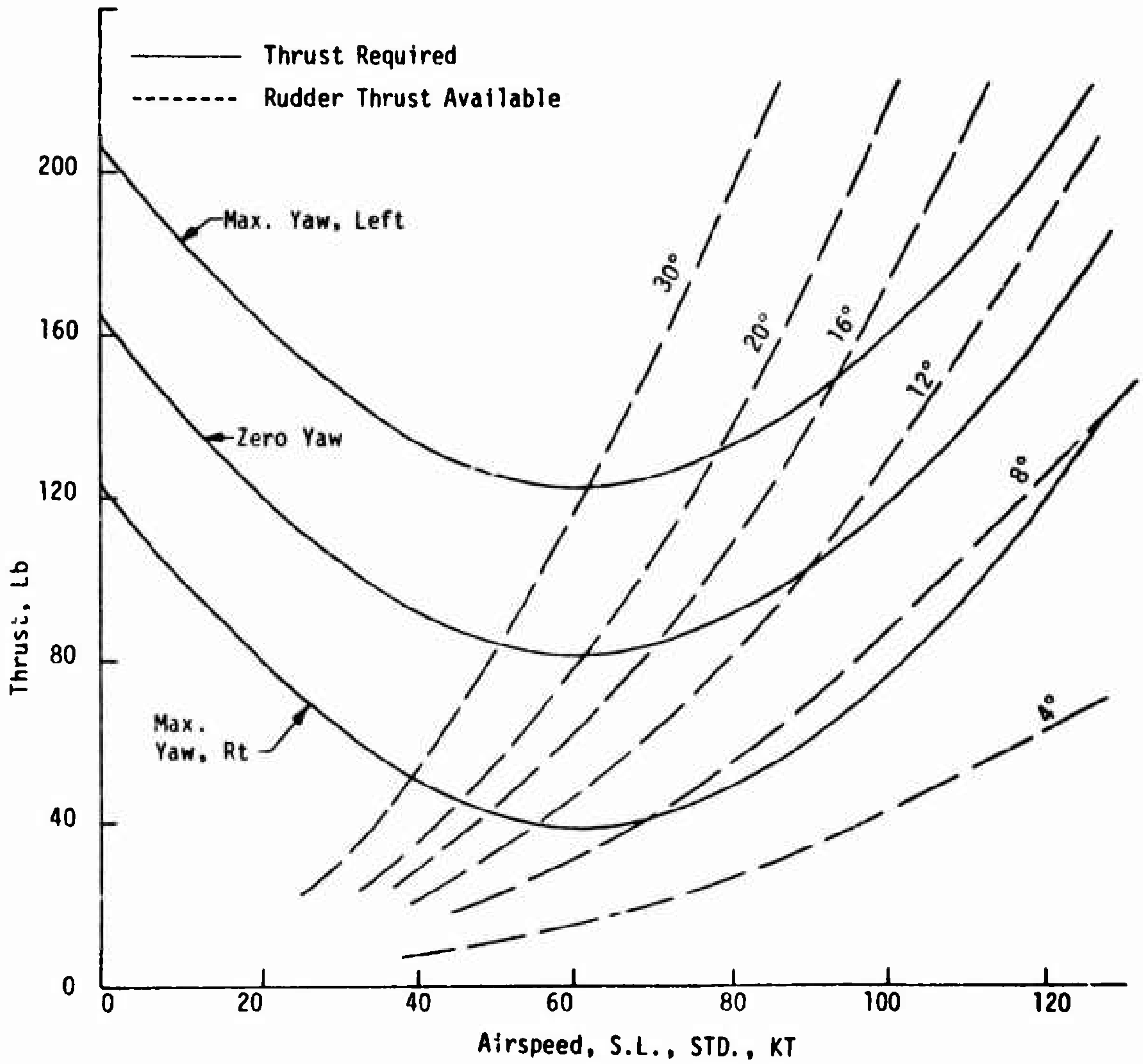
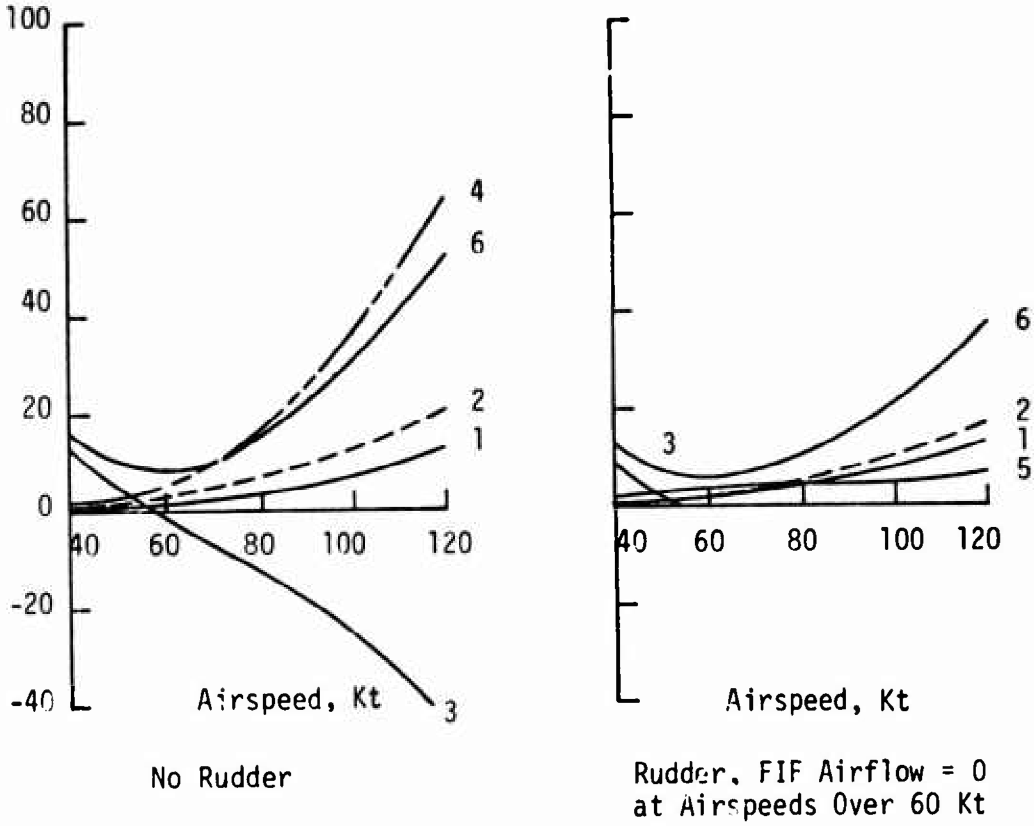


Figure 9. Variation of Exit Thrust Required and Rudder Thrust Available With Airspeed and Rudder Deflection.



POWER INCREMENTS:

1. External Skin Drag, Duct
2. Exit Base Drag, Duct
3. Shaft Power, Fan
4. Momentum Drag
5. Fin-Rudder Drag
6. Total

Figure 10. Differential Power Increments for Antitorque Thrust at Higher Airspeeds, With and Without Rudder.

The modes of operation of the controls for the various flight regimes and transition are specified in Table 4. Mechanical controls suggested to effect the required operations are shown schematically in Figures 11 through 13, excepting for the rudder control, which is a linear control from the pedal takeoff. Two actuators only - one operating from a torque signal, preferably collective stick position, the other from an airspeed signal - are required. They are shown for clarity in each of the schematics.

STRUCTURE

New structure is required to accommodate the air inlet, to support the fan, empennage, tail skid, control vanes and airflow turning vanes, and to support or constitute the fan housing and airflow duct.

Since the fan, duct, and air exit must be relatively large for minimum fan power (see Trade-Off Study, page 50), the tail boom has a much larger cross section than that of the baseline OH-6A. Loads are no higher, except for the small increase due to increased weight. This creates a problem if a standard semimonocoque structure is used. The lateral moment (ultimate) at the base of the boom due to maximum thrust is approximately 36,000 in.-lb. Assuming a 1.70-ft radius and a .020-in. thick skin, and assuming that the skin is stable, the maximum stress is 617 psi. It is thus clear that an aluminum semimonocoque structure would be highly inefficient. A skin is required which is self stabilized and is light; and a low stress allowable is no detriment. A plastic or fiberglass-skinned sandwich with a foam core is ideal for the application.

The use of innately stable sandwich skin has the further advantage that no separate duct or fan housing is required. The inner surface of the sandwich is, itself, the duct, and has what is probably the lowest attainable friction coefficient. Also, it is highly repairable.

The materials selected are:

Outer skin	2-ply #181 E-glass
Inner skin	2-ply #120 E-glass
Core	1/4-in. R-400 Goodrich Rigid Cell (polyvinal chloride) or equivalent

See Reference 18 for further data on this construction.

[18] Mayerjak, Robert J., and Smyth, William A., INVESTIGATION OF ADVANCED STRUCTURAL CONCEPTS FOR FUSELAGE, Kaman Aerospace Corporation, Bloomfield, Conn., USAAMRDL Technical Report 73-72, Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, October 1973, AD 773597.

TABLE 4. CONTROL FUNCTIONS/FLIGHT CONDITION

FLIGHT REGIME	FAN PITCH	CONTROL VANE MAIN ANTITORQUE	CONTROL VANE LEFT AUTOROTATIONAL	CONTROL VANE RIGHT AUTOROTATIONAL	RUDDER
Autorotation	Same	Closed	Varies from full closed (neutral pedal) to full open (full left pedal)	Varies from full closed (neutral pedal) to full open (full right pedal)	Travel Proportional to pedal travel, all regimes
Transition 2 seconds	Same	Closing	Closed (neutral pedal); left pedal authority increasing	Closed (neutral pedal); right pedal authority increasing	
Hover to 45 kts (Base point)	Varies as function of engine torque and of absolute value of pedal position	Full open	Open	Closed	
Transition 50 to 60 kts	Authority reduced to zero (both inputs); pitch reduced to zero	Closing	Closing	Closed	
60 to 120 kts	Constant, flat pitch	Closed	Closed	Closed	

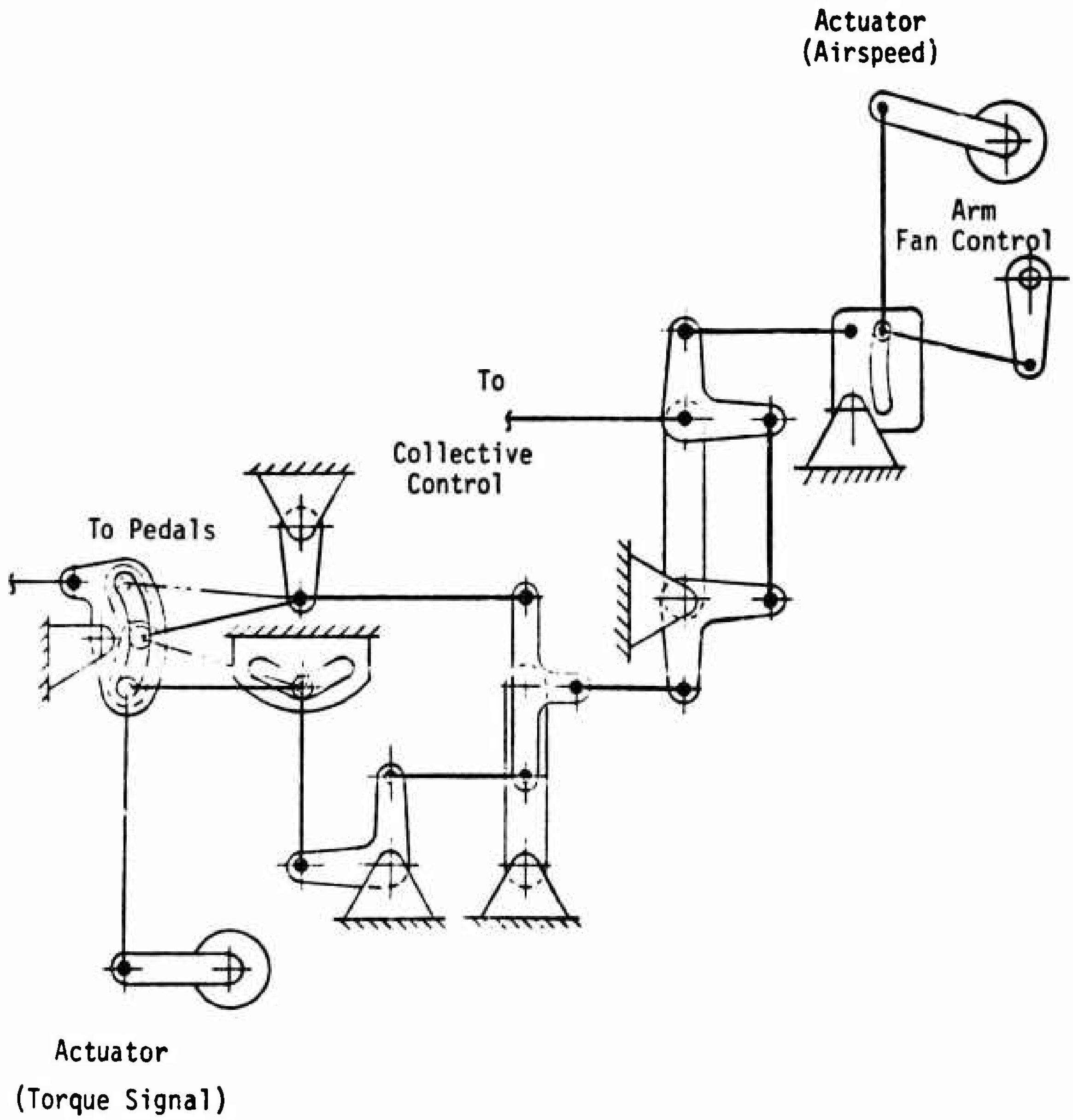


Figure 11. Schematic, Fan Pitch Control.

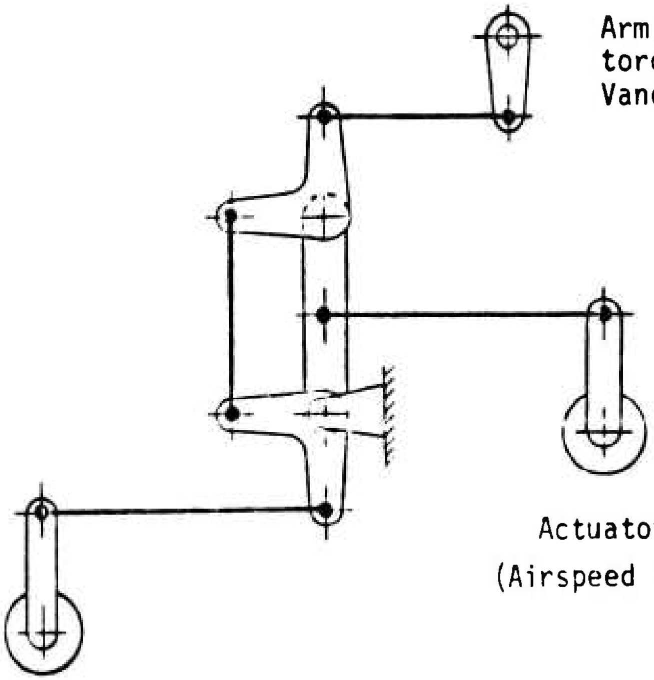


Figure 12. Schematic, Antitorque Control Vane Control.

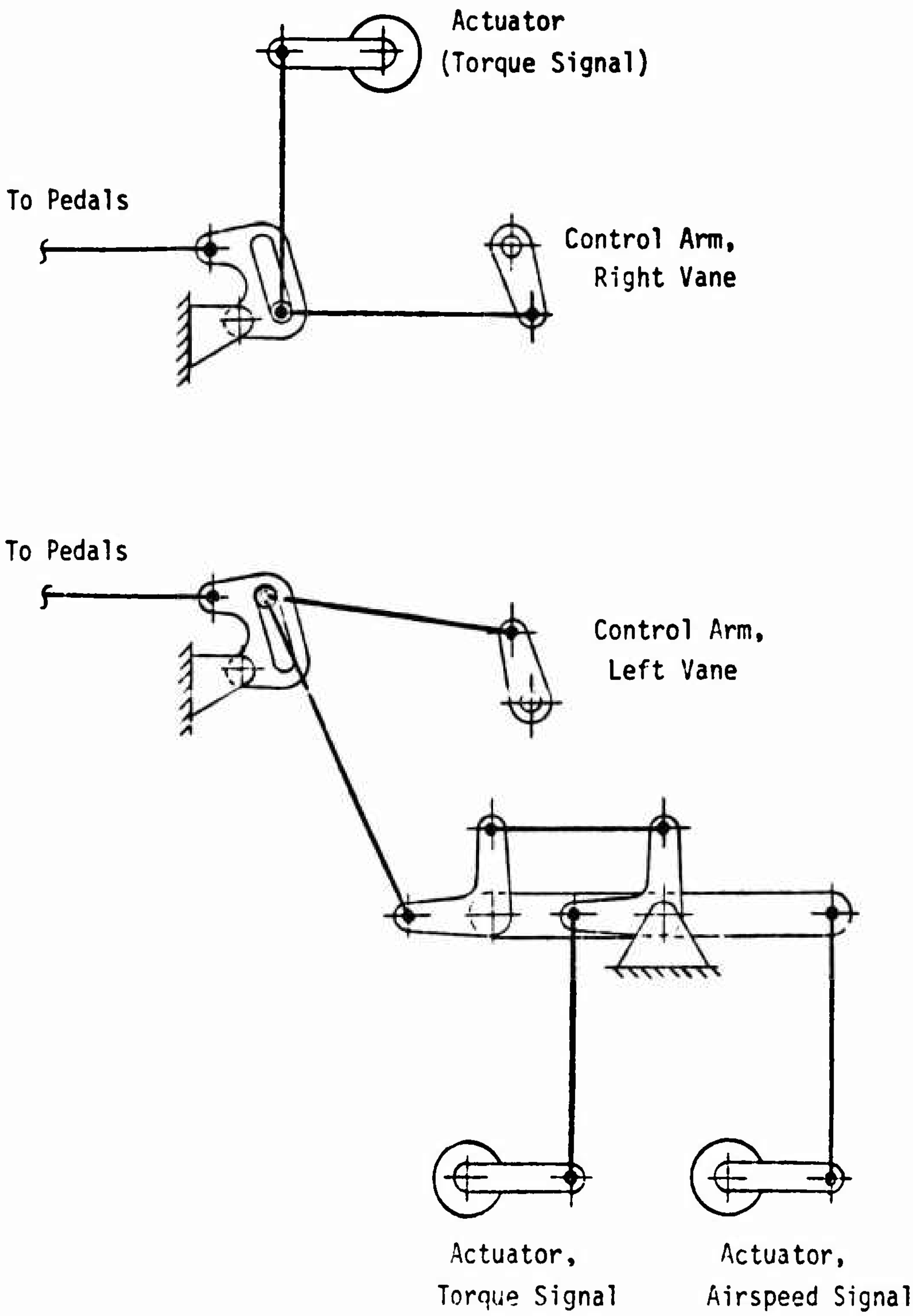


Figure 13. Schematic, Autorotational Control Vanes.

Since the tail boom is a monocoque structure, a distributed load transfer would in some ways be more efficient, but the fan support must be through three discrete points, and a distributed load transfer structure would reduce the inlet area. The boom was therefore attached to the fuselage using a truss, similar to a radial engine mount, of welded aluminum tubing. See Figure 8.

FAN DESIGN

Fan requirements were specified on the basis of preliminary analysis as:

Fan diameter	3.50 ft
Exit area	9 sq ft (hover)
Thrust	204 lb (hover, maximum yaw)
Exit area	1.8 ft (autorotation)
Thrust	40 lb (autorotation, max. yaw)
Pressure loss coeff.	0.20 (referred to exit velocity)
Air density	.002378 slug per cu ft

Aerodynamic studies investigated the influence of design lift coefficient, hub/tip diameter ratio, total activity factor, and tip speed on fan horsepower for the two design conditions. The study is summarized by data presented in Figures 14 through 22. Significant conclusions which can be drawn from these data are:

- The minimum design point power is not significantly influenced by the major fan parameters over the range investigated.
- The performance requirements of both hover and autorotation can be met through the use of variable pitch.

Based on the aerodynamic studies, a fan having the following characteristics was selected for the conceptual mechanical design:

Number of Blades	2
Total Activity Factor	200
Integrated Design Lift Coefficient	0.7
Tip Speed	650 fps
Hub/Tip Diameter Ratio	0.45

Efficiency (air power output/fan power required) at the original design point is 97.3%. Subsequent changes in the design concepts reduced the thrust required for steady hover from 204 to 140 pounds, but increased the duct loss factor, K, from 0.20 to 0.234. It was not possible to redesign the fan to the final requirements; but it was assumed that the original design point efficiency could be met for the final design point.

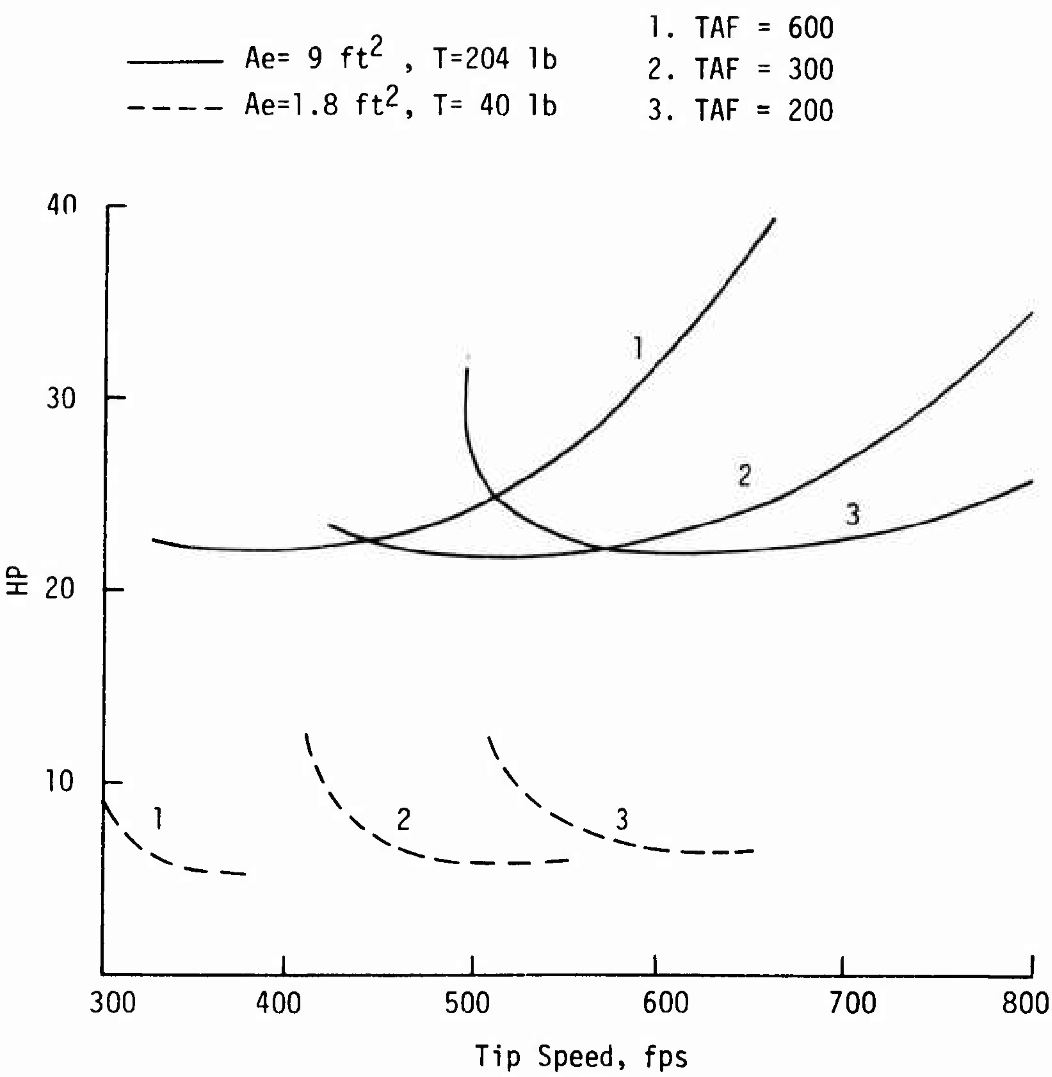


Figure 14. Fan Power Variation With Tip Speed and Total Activity Factor for C_l = .7 and D_h/D_f = .45.

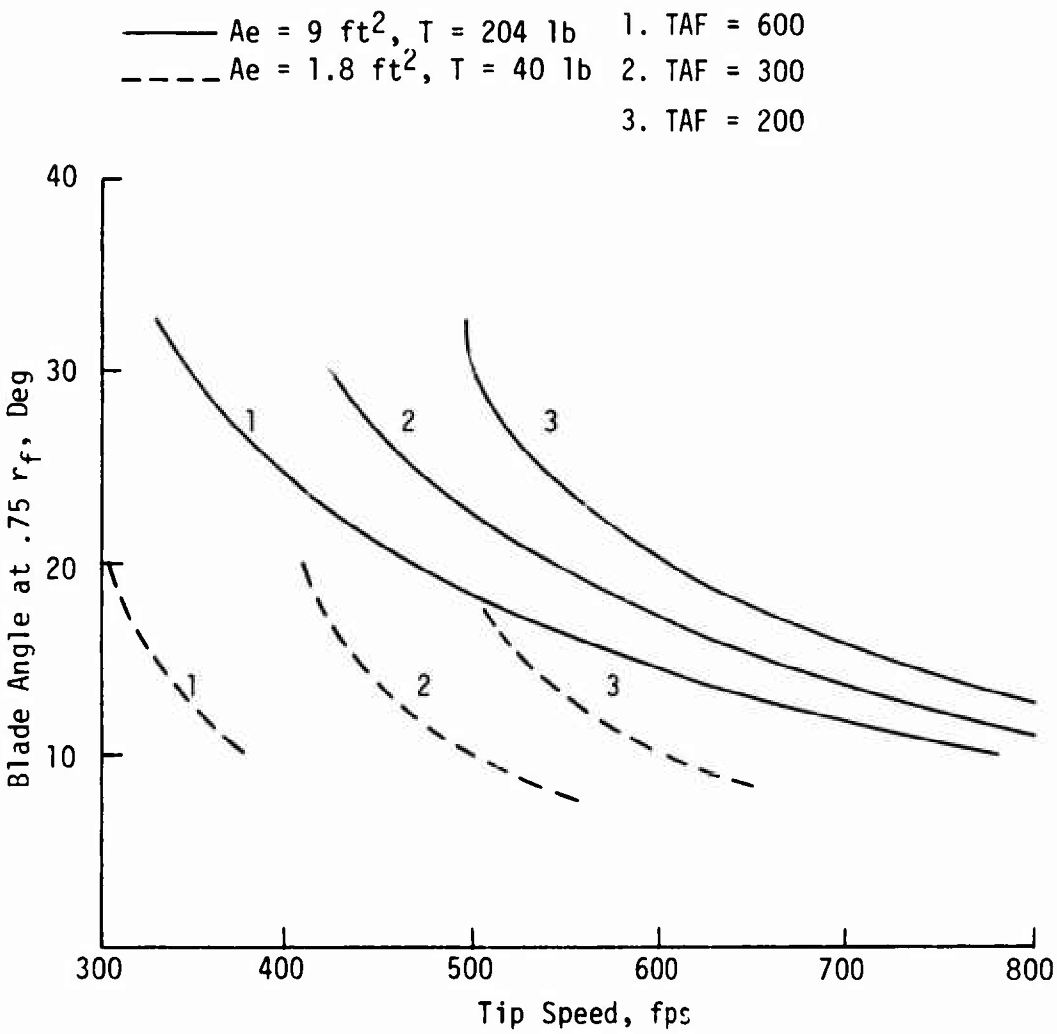


Figure 15. Blade Angle Variation With Tip Speed and Total Activity Factor for C_l = .7 and D_h/D_f = .45.

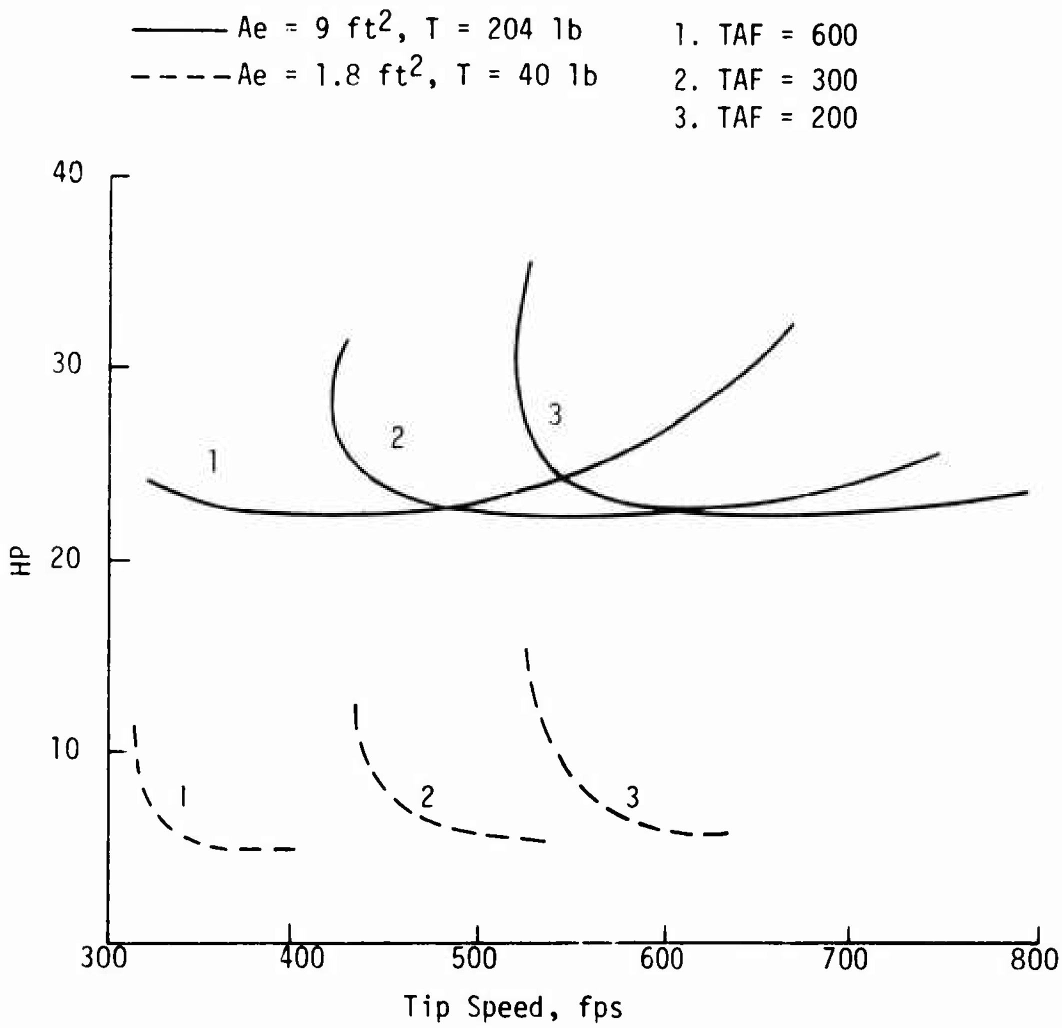


Figure 16. Fan Power Variation With Tip Speed and Total Activity Factor for C_1 = .7 and D_h/D_f = .30.

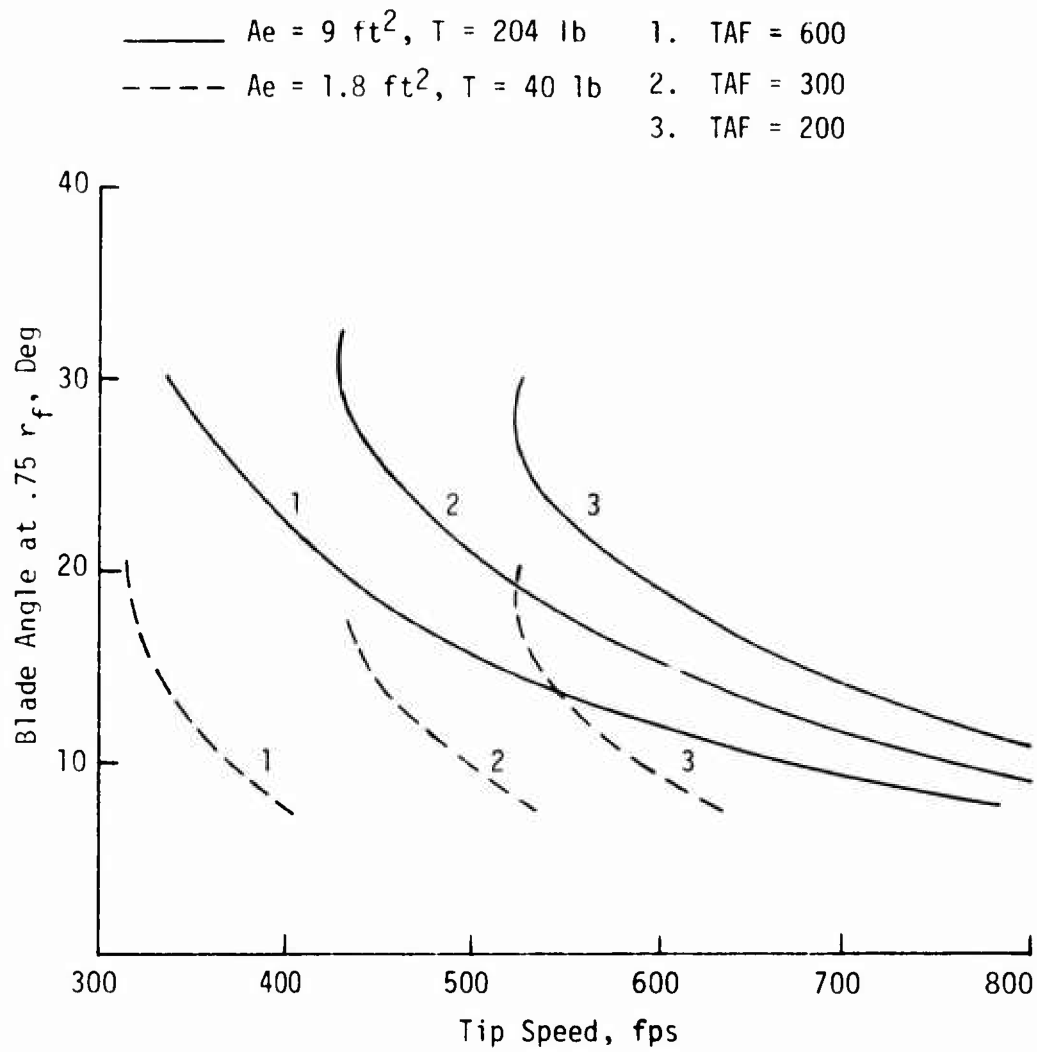


Figure 17. Blade Angle Variation With Tip Speed and Total Activity Factor for C_1 = .7 and D_h/D_f = .30.

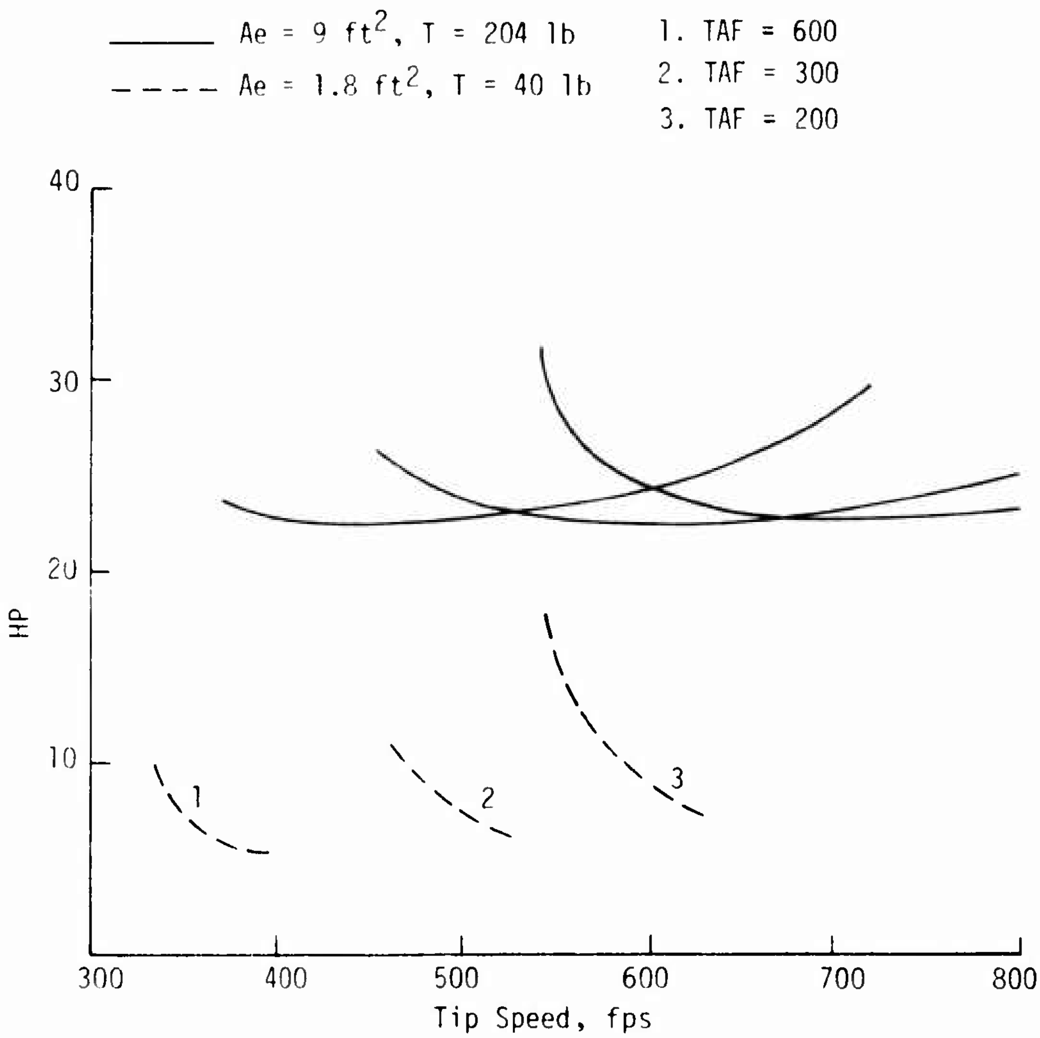


Figure 18. Fan Power Variation With Tip Speed and Total Activity Factor for C_1 = .5 and D_h/D_f = .3.

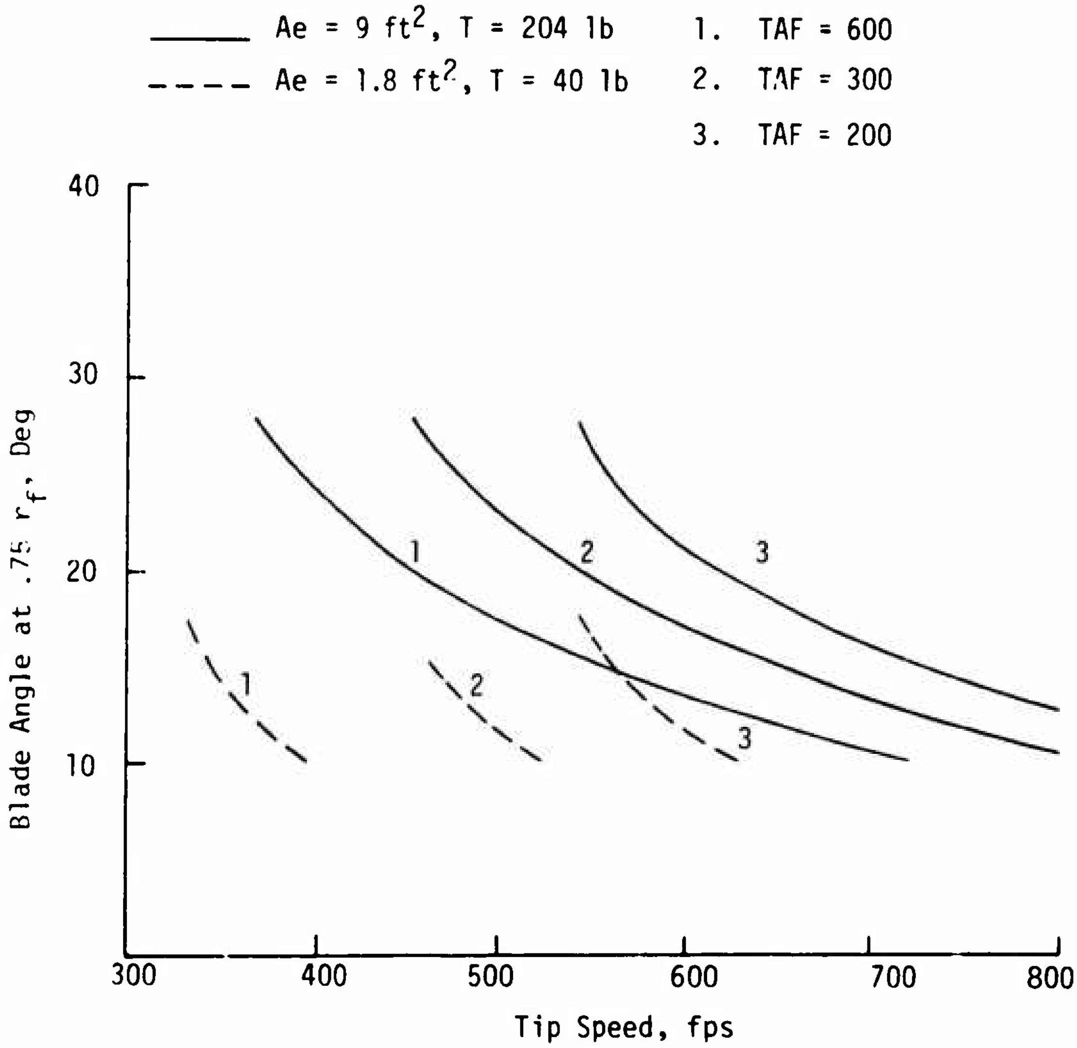


Figure 19. Blade Angle Variation With Tip Speed and Total Activity Factor for C_l = .5 and D_h/D_f = .3.

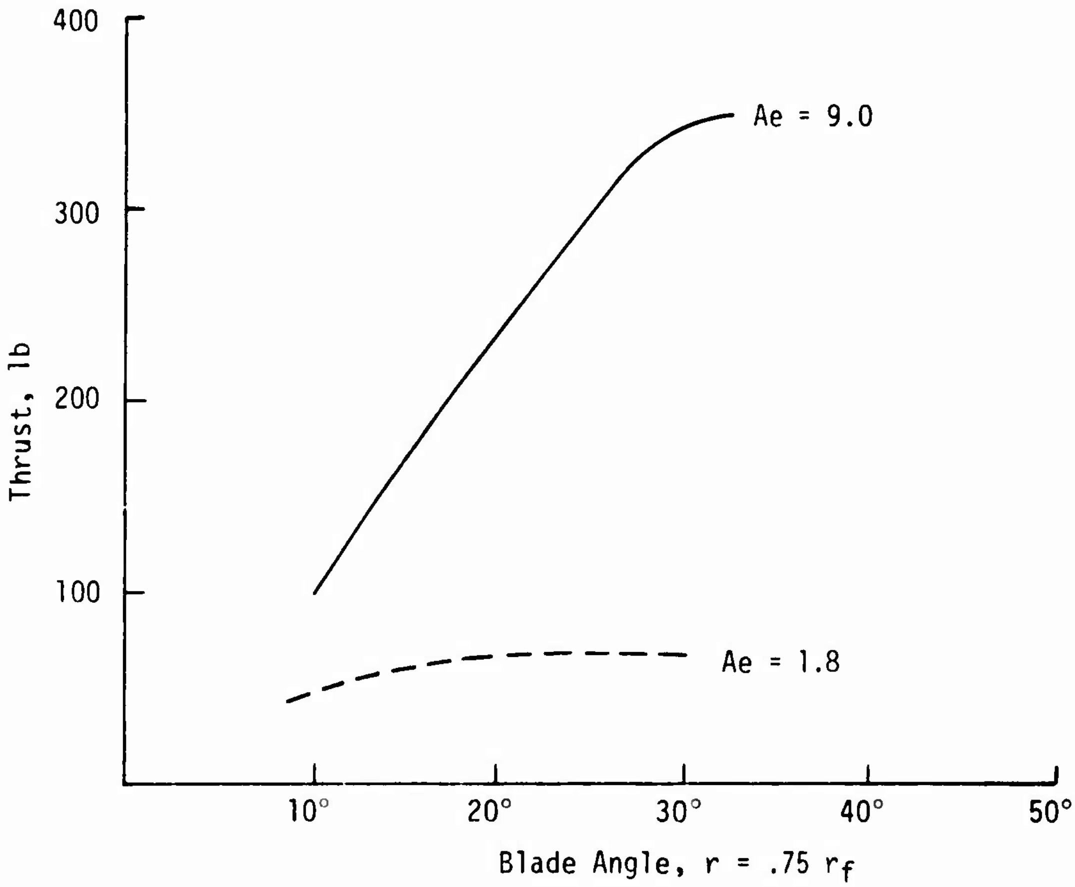


Figure 20. Thrust Variation With Blade Angle.

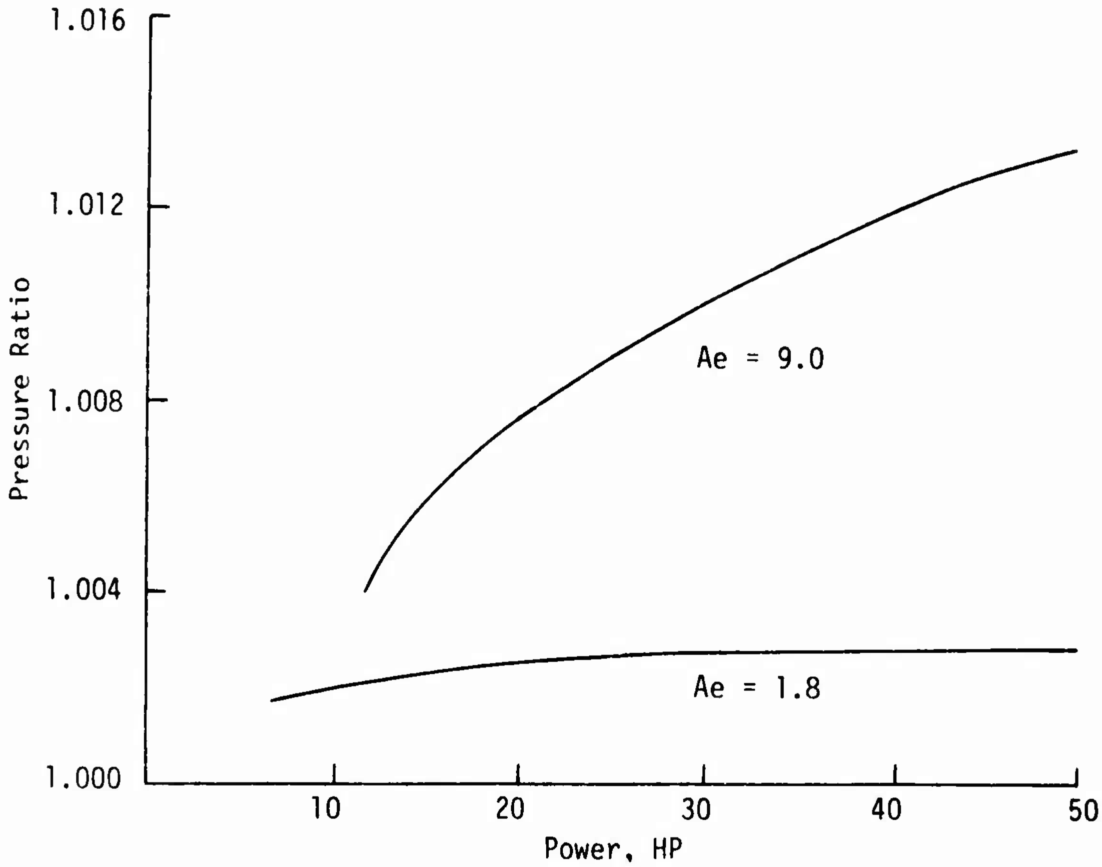


Figure 21. Fan Shaft Power Variation With Pressure Ratio.

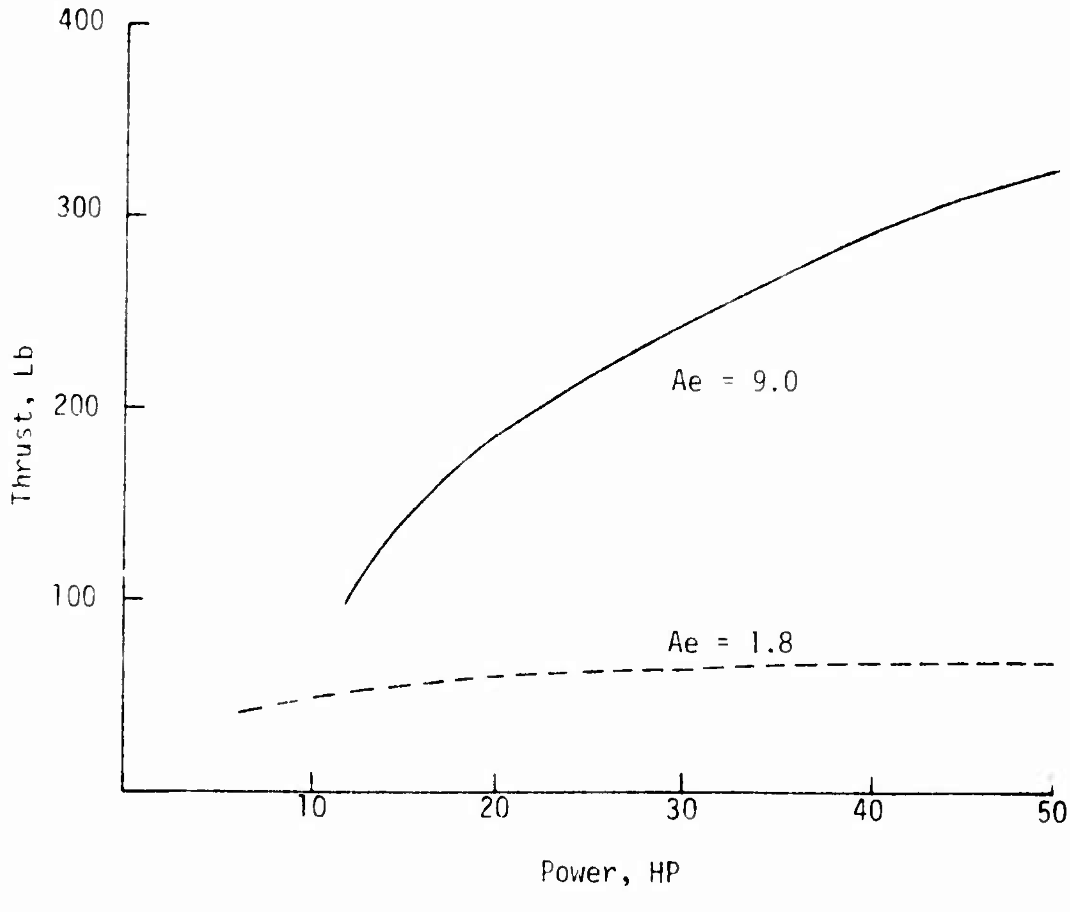


Figure 22. Variation of Fan Power With Thrust

Design study parameters leading to this design are given in Figures 14 through 19. Fan performance is given in Figures 20 through 22, and the design is shown in Figure 23.

The design incorporates an integral gearbox, mechanical pitch change mechanism, fan disc and centerbody. The design employs conventional lightweight structures and material, but does not incorporate the advanced lightweight technology, i.e., composite blades and titanium structures, which have been used in larger fans, because it was believed that the additional weight saving did not justify the cost. Further studies would be needed to identify the weight savings and costs associated with advanced technology.

The fan is driven through 1.2:1 speed increasing gearing mounted in the fan support housing. A flanged drive positioned at a 15° angle relative to the fan axis provides the interface with power shafting from the main transmission. The gearing is splash lubricated, using oil which meets the requirements of specification Mil-L-7807.

The fan disc is supported by ball and roller bearings contained in a magnesium housing which also houses the gearing. This housing incorporates provisions for attaching to three struts which transmit the fan loads to aircraft structure.

The fan disc is a two-piece steel-structure which supports solid aluminum blades on split race roller bearings. The solid aluminum blade incorporates a collar which prevents the blade from moving inward when the fan is not rotating; this collar also provides a sealing surface for the blade retention seal.

The pitch change mechanism is an all mechanical device. It incorporates a "no-back" which is a mechanical brake that prohibits blade movement except as produced by the input control. In the event of an input mechanism failure, the blades would remain fixed at the last selected angle. Force to change pitch is applied to a bell crank which translates a nonrotating member on the downstream side of the "no-back". Pitch change loads are carried through a bearing set to trunnions on the blades via a translating yoke which rotates at fan speed.

A lightweight fiberglass centerbody provides a smooth aerodynamic contour for air exiting the fan. The hub/tip diameter ratio for the fan with the centerbody shown in Figure 23 is 0.286. While this centerbody is smaller than that used for the performance calculations, it has no significant effect on the data presented by curves, Figures 20 through 22.

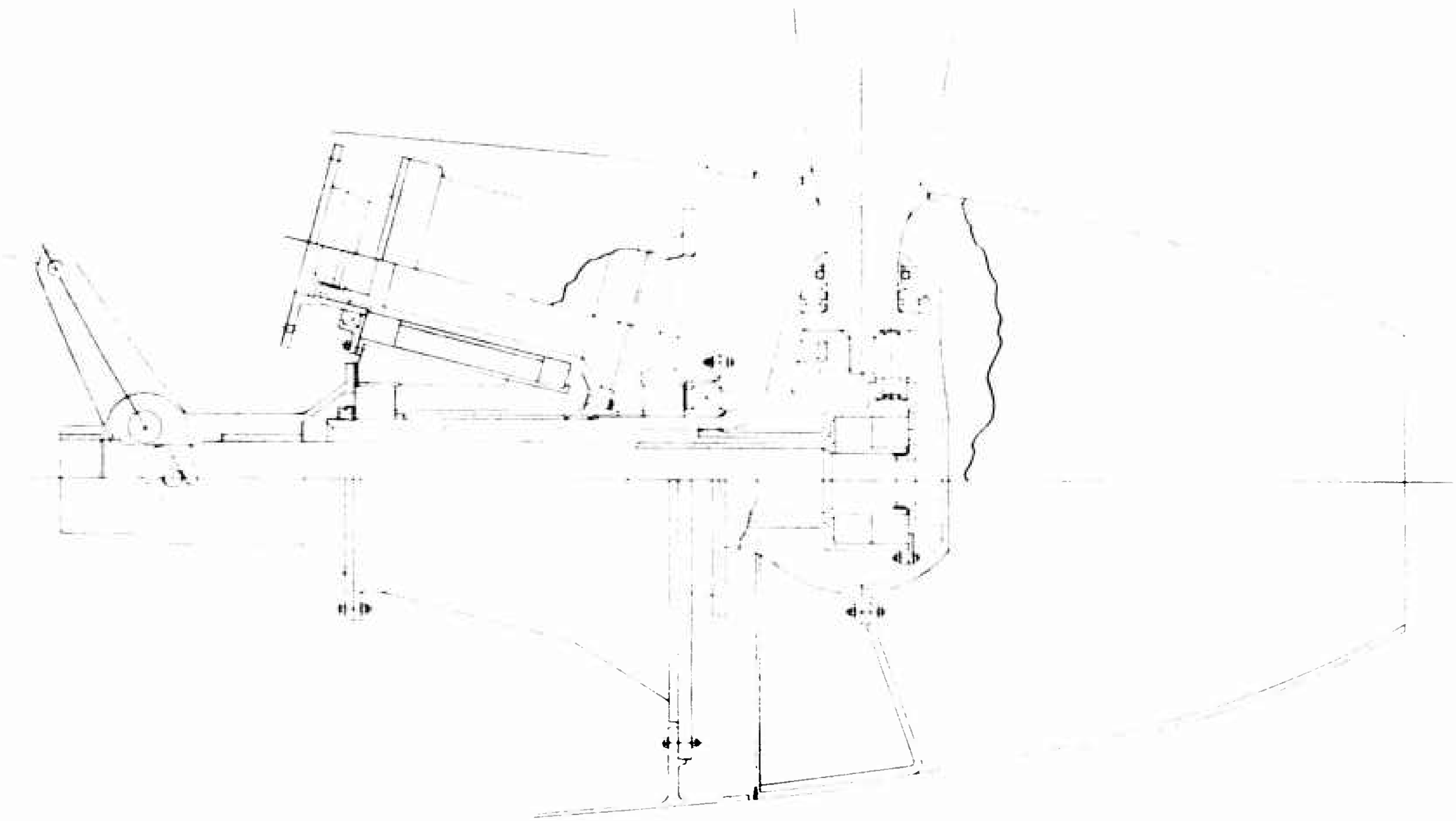


Figure 23. Fan Gearbox and Hub Detail Design Concept.

HOVER POWER TRADE-OFF STUDY

A conceptual design of the helicopter, as modified, was prepared and used as a base point for trade-off and other design studies.

Although, for a given thrust output, minimum fan power is obtained by making the fan and exit areas as large as possible, it does not necessarily follow that such areas will result in minimum helicopter power required. Thrust is reduced as the tail boom is lengthened; weight is reduced at constant boom length with a smaller exit area, and it is not immediately apparent what combination of dimensions is optimum.

The weight of the conceptual design, in which a 9-foot area and a distance of 16 feet from the cg to the exit area was used, provided a basis for the establishment of equations expressing weight as a function of variable exit areas and torque arms. Fan area was held constant. Fan power was determined from the basic momentum relationship (Equation (23)). Fan power determined was therefore unconservative where exit velocities significantly exceeded those of the baseline configuration. Nevertheless, the analysis shows that, up to a 30-ft moment arm to the exit, the net effect of increasing moment arm and decreasing exit area is to reduce total power required (Figure 24).

GENERAL ARRANGEMENT

The general arrangement of the selected preliminary design is given in Figure 18. In Figure 25, the profiles of the modified and unmodified OH-6A helicopters are shown for comparison.

Tail boom length was held as close as possible to that of the OH-6A in order to obtain a valid performance comparison. Stiffness and weight considerations made it desirable to minimize loads in the air exit area. The empennage and tail skid are consequently attached to a vertical beam, or "island", on the centerline just forward of the exit area; use of this location relieves the open section at the exit area of all torsional and significant vertical bending loads.

Maintenance of fan tip clearance to a maximum of 0.08 in. is critical to the high efficiency of the fan. The design calls for the forward end of the boom shell, which serves as the fan housing, to be made with a 3-in. (average) core, and with a soft plastic insert at the fan blade tip area. The shell, spliced shortly aft of the fan, will be cured with the inside diameter of the insert left undersize. The forward shell section will then be jigged for matching of the inside diameter, and of fan and tail boom mount attaching holes.

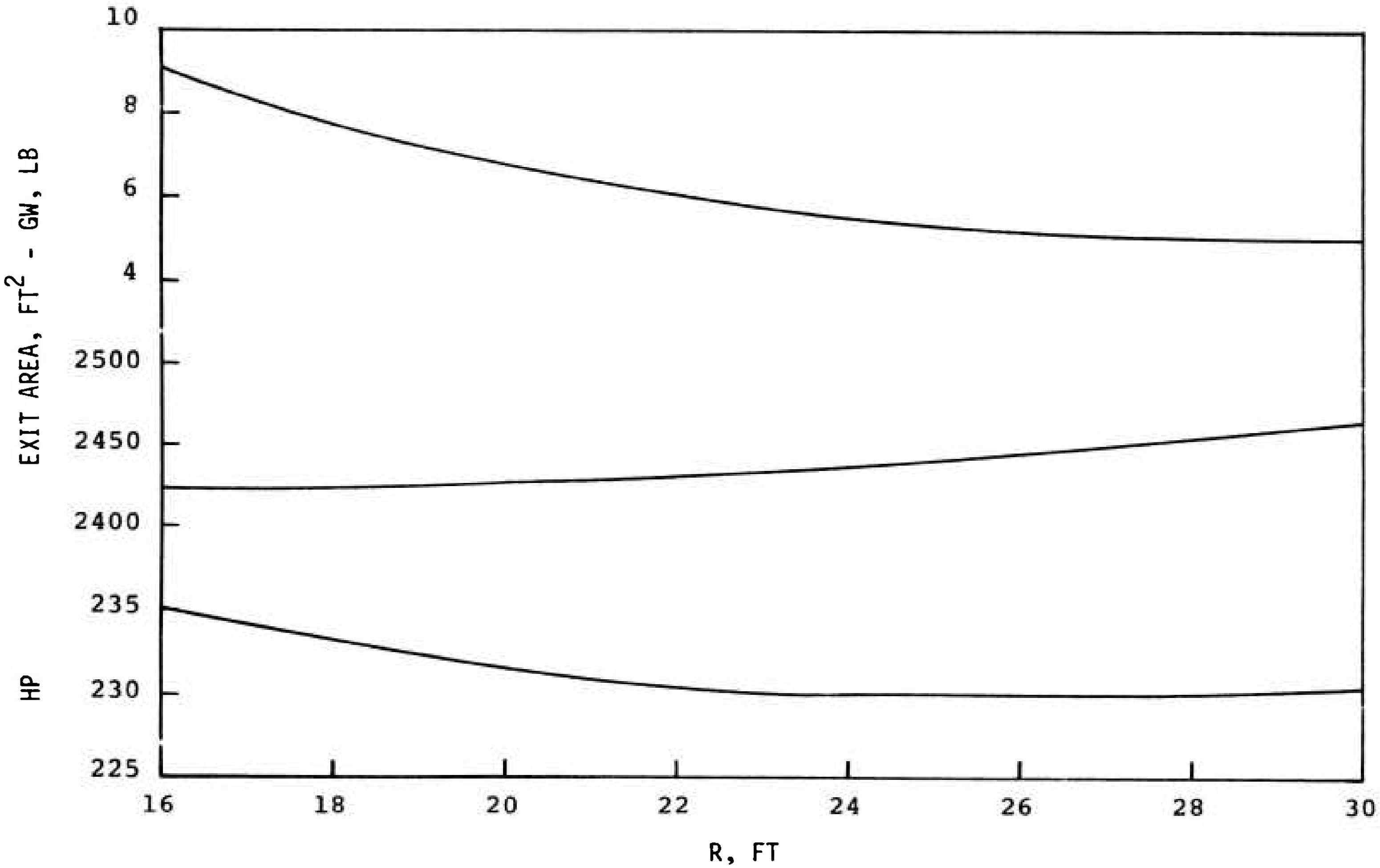


Figure 24. Fan-in-Fuselage Power, Weight, and Exit Area Variation With CG to Exit Distance.

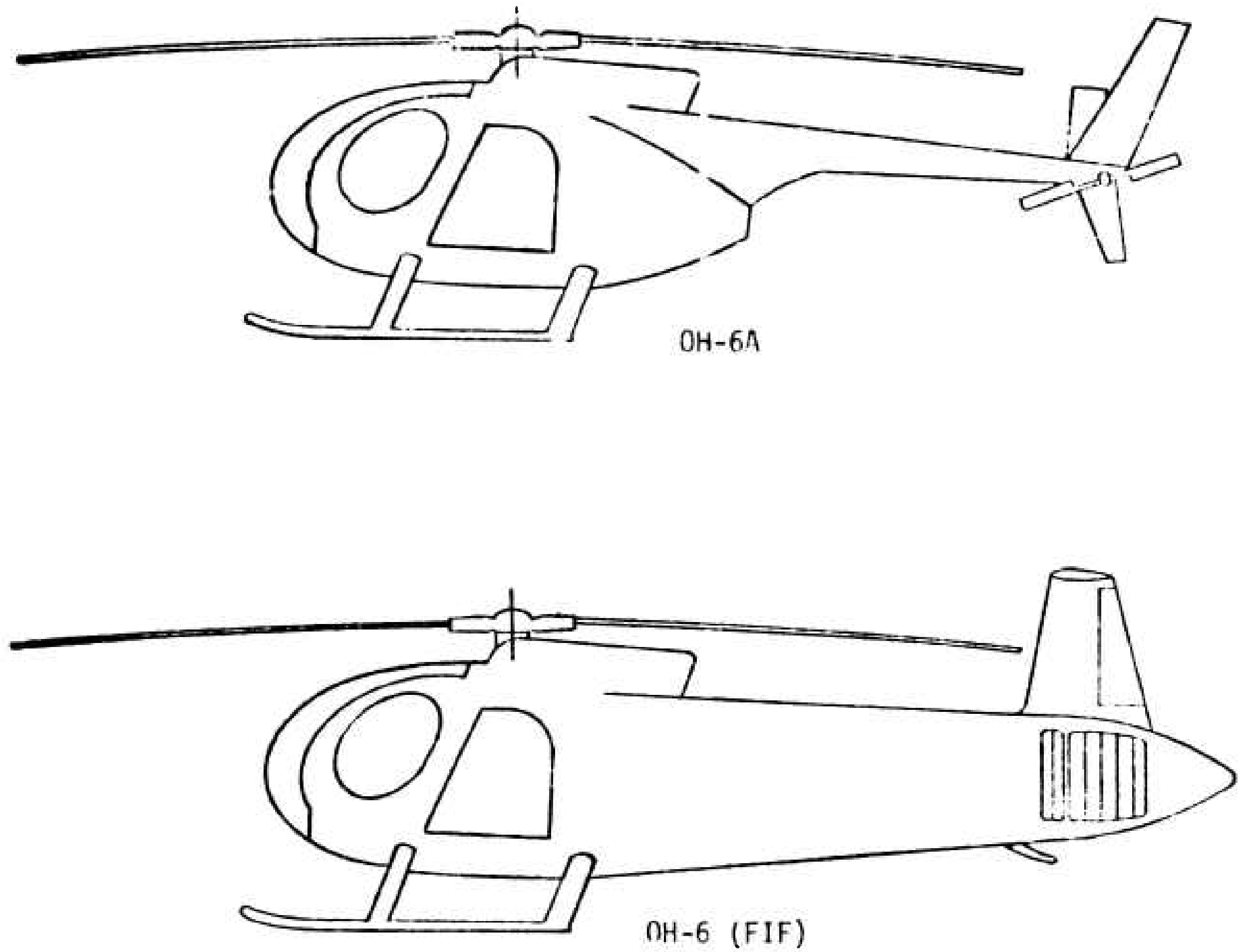


Figure 25. OH-6(FIF) Profile Compared to OH-6A Profile.

The air inlet screen may not be necessary. Inlet air velocity at takeoff gross weight is only 35 fps, fan solidity is only .163, and the blades are of solid aluminum. The probability of significant foreign object damage is small. The screen has, conservatively, been included; but elimination of its weight and drag would reduce total power required in hover by about 2 horsepower.

The fan is driven from the tail rotor output of the main transmission, through two bevel gear sets (one of which is included in the fan assembly). No clutch is used, although the fan requires 5 horsepower at zero airflow, the flow programmed for speeds over 60 knots. A clutch would weigh about 15 pounds; mission fuel saving would be only 6 pounds.

WEIGHT AND BALANCE

Both weight and balance are of major concern in the fan-in-fuselage design. The tailboom must enclose a large duct area for aerodynamic efficiency. and is therefore heavier than a tail rotor tailboom. In this particular design, the offset from the centerline of the tail rotor drive takeoff in the main transmission to the centerline of the fan makes another gearbox necessary. In addition, airflow control and turning vanes are required at the aft end of the boom, and a larger fin area is required because the yaw moment of inertia is higher. The gross weight is therefore increased by 7 percent, approximately.

Since most of the weight increase is in the tail area, a large center-of-gravity shift would occur if no compensating changes were made. In a new design, proper balance would be provided in the basic arrangement. In this modification design, it is proposed to shift a large part of the forward section forward, adding splice structure just forward of the rotor. The center of gravity is thus located on the rotor centerline. An additional weight penalty of 10.4 pounds is thus incurred, and the cabin size is increased by 5.8 inches.

DESCRIPTION	WEIGHT (lb)	HORIZONTAL ARM (ft)	MOM(ft-lb)
Basic Aircraft Weight	(1154)	105.98	(122300)
Oil	7	130.00	910
Crew (2)	400	73.50	29400
Fuel (615 Gal.)	400	97.70	39080
Cargo	300	100.00	30000
Takeoff - No Fan Instl	(2261)	98.05	(221690)
Added:			
	(222.3)	210.40	(46774)
Fan Instl	32.0	178.00	5696
Gearbox	12.0	140.00	1680
Drive Shaft	2.7	158.60	427
Air Inlet Screen	11.4	150.80	1719
Tailboom Supt. Struct.	10.4	151.00	1570
Fan Shroud Struct. & Closeout	15.7	175.80	2765
Ducting	32.2	244.60	7867
Center Post	7.8	260.50	2032
Tail Skid	2.3	258.00	593
Fin Rudder	9.6	272.10	2623
Horiz. Stabilizer	7.3	270.50	1975
Air Control Vanes - Main	10.4	253.00	2631
Air Control Vanes - Auto	3.8	264.00	1003
Cascade Vanes - Main	14.9	290.00	4321
Cascade Vanes - Auto	3.9	268.00	1045
Tail Fairing	4.6	320.60	1475
Lower Fairing	8.1	202.10	1637
Fan Support Struct.	6.5	172.00	1118
Finish	6.7	220.50	1477
Controls - Aft	10.0	246.00	2460
Controls - Fwd	10.0	66.00	660
Remove:	(-76.7)	237.00	(-18181)
Tail Group - Blades & Hub	20.9	284.00	-5957
Aft Cabin Structure	-14.7	-186.00	-2734
Tail Boom	-13.4	240.00	-3216
Drive Shaft	-6.0	220.00	-1320
Tail Gearbox	-13.0	282.00	-3666
Engine Doors	-8.7	148.00	-1288
Takeoff - Fan Instal In	(2406.6	104.00	(250283)
Fuselage Splice (5.8 In.)	10.4	100.00	1040
Move Fuselage Fwd 5.8 In.			-9626
Takeoff With 5.8 Splice	(2417.0)	100.00	(241697)

TABLE 5. ESTIMATED WEIGHT

INTERNAL PRESSURE LOSSES AND POWER REQUIRED

The fan-in-fuselage airflow path is broken down in order to determine total pressure loss, as shown in Table 6, which lists local cross sectional areas, local velocities, loss factors, and pressure losses. The basis of the loss factors used are appended as notes following the tabulation.

TABLE 6. INTERNAL PRESSURE LOSSES

Section (ft^2)	Area	Velocity (ft/sec)	P_d (psf)	K	P_dK (psf)	Note No.
Exit Pd	9.2	81.3	7.85	-	-	-
Screen	30.0	24.9	.739	.40	.296	1
Inlet Turn	18.8	39.7	1.875	.09	.169	2
Inlet Lip	9.6	77.9	7.218	.03	.216	3
Duct	9.6	77.9	7.218	.03	.216	4
Contraction	9.2	81.3	7.859	.04	.314	5
Turn	9.2	81.3	7.859	.05	.392	Fig. 3
Sub-Total					1.531	
Allowance, 20%					.306	
Total Loss					1.837	

Notes:

1. Screen - Screen is .0625 in. dia wire, 1/2 in. mesh solidity is thus 0.25. Flow rate is 747 cfs. Then, from Figure 3-33[9], K = 0.40
2. Inlet Turn - Effective area is taken as the mean between the inlet screen and the net fan areas. Loss is estimated to be equivalent to that in a round duct for a 30° turn, r/d = 2.0, and K = .09[9].
3. Inlet Lip - For r/d = 4 in./40 in. = 0.1, K = .03. See Figure 1A-39[9].
4. Duct - Length = 8.78 ft, Dia = 3.5 ft, L/D = 2.5, Airflow = 81.3 x 9.2 x .00238 x 32.2 = 57.3 lb/sec, w/d = 1.36, From Figure 1A-8[5]: 4f = .01 and K = 4fl/d = .025
5. Contraction - Ref. 9, Para 3.3.2. leads to a value of K = .01. Conservatively, K is taken as 0.04.

[9] Committee A-9 Membership, SAE, AEROSPACE APPLIED THERMODYNAMICS MANUAL, Society of Automotive Engineers, Inc., Aero-Space Environmental Systems, 485 Lexington Ave., New York 17, N.Y., February 1960, January 1962.

Loss factor is then

$$K = 1.837/7.85 = .234$$

Thrust required (see Equation (2)) is:

$$T_t = 33000\ P_r/2\pi \times 483 \times 16.2 = .671\ P_r$$

and fan power (Equation (23)) is:

$$P_f = T^{1.5}[1 + K]/1100\ n_f \sqrt{\rho A_e}$$

$$= (.671\ P_r)^{1.5}[1 \times 0.234]/1100 \times 0.95 \times \sqrt{.002378 \times 9.2}$$

$$= .0044\ P_r^{1.5}$$

EXTERNAL DRAG

Fuselage Drag - The fan-in-fuselage concept embodies a fuselage with a greater fineness ratio than the basic OH-6. While the form drag is less, the friction drag is increased due to the larger wetted area. The difference in the parasite drag is estimated from Reference 19:

	Basic OH-6	Fan-In-Fuselage
Projected Frontal Area, ft^2	22	22
Fuselage Length, ft	12.4	25.7
Fineness Ratio	2.33	4.86
Parasite Drag Coefficient	.053	.060
Equivalent Drag Area, ft^2	1.17	1.32

Tail Rotor Drag - Approximate values of tail rotor drag were determined from a reference plot of rotor h-force versus thrust coefficient, solidity and advance ratio. Thrust values were determined for the OH-6 speed range from main rotor power required at a gross weight = 2600 lb for sea level standard day. The equivalent flat plate drag area is shown in Table 7.

[19] Perkins, C. D., and Hage, K. E., AIRPLANE PERFORMANCE, STABILITY AND CONTROL, New York, New York, 1949.

TABLE 7. FORWARD FLIGHT DRAG COMPARISON

6W-2600 lb, SL Std	OH-6A				OH-6 (FIF)			
Advance Ratio =	.08	.16	.24	.32	.08	.16	.24	.32
A_f (fuselage, inlet screen, and exit values), ft^2	1.17	1.17	1.17	1.17	2.02	2.02	2.02	2.02
A_f (tail rotor H-force), ft^2	.48	.19	.15	.25	0	0	0	0
A_f (fins and rudder), ft^2	.05	.05	.05	.05	2.10	1.15	0.38	0.33
ΔA_f, Total	1.70	1.41	1.37	1.47	4.12	3.17	2.40	2.35
OH-6A equivalent flat plate area, ft^2	5.0	5.0	5.0	5.0				
Residual OH-6A drag, ft^2	3.3	3.59	3.63	3.53	3.3	3.59	3.63	3.53
Fan-in-fuselage drag (residual OH-6A drag plus ΔA_f, total), ft^2					7.42	6.76	6.03	5.88
Drag increase, OH-6 (FIF), ft^2					2.42	1.76	1.03	.88

Vertical Fin - Drag of the basic OH-6 vertical fin was estimated for zero angle of attack and a reference area of 5.64 ft^2. Drag of the fan-in-fuselage vertical fin depends on the rudder deflection required to overcome main rotor torque. A reference plot of fin-rudder lift and drag versus rudder deflection at zero angle of attack and aspect ratio of 3.0 was used to estimate the drag of both fins.

Horizontal Fin - The fin area and shape for both the basic OH-6 and the Fan-in-Fuselage concept are assumed to be the same. Any differences in fin drag due to location and position are expected to be negligible.

Fan Inlet Screen - In forward flight, the fan will be programmed to idle at or near a zero flow condition with antitorque force supplied by the fin-rudder. The drag force for a flush inlet with no inflow is essentially zero. The external drag then becomes only a function of the added screen roughness for the screen area portion of the fuselage. A typical skin friction drag coefficient = .003 is doubled and applied to the screen area as an estimate of the drag associated with the inlet screen. Then

$$Af = 2 \times .003\ (34\ ft^2) = .204\ ft^2$$

Exit Vanes - An estimate of the drag penalty due to the exit turning vanes was obtained by assuming that the fuselage boat tail area affected by the vanes is a blunt trailing edge with a drag coefficient of 0.10.

Projected boat tail area = 5 ft^2

Exit vane equiv. flat plate area = 5 x .10 = .5 ft^2

Summation of the drag differences noted above are shown in Table 7 which describes the estimated Fan-in-Fuselage forward flight equivalent flat plate area.

AERODYNAMIC PERFORMANCE

A comparison of the OH-6 Fan-in-Fuselage with the standard OH-6A was made for a representative mission. Performance, discussed in full in Appendix C is summarized here.

The mission derived from a report on OH-6A operations in Vietnam[20] was:

[20]Giessler, F. Joseph, et al, FLIGHT LOADS INVESTIGATION OF OH-6A HELICOPTER OPERATING IN SOUTHEAST ASIA, Technology, Inc., October 1971, AD 738202.

Average Flight Condition

Altitude	1000 ft
Temperature	90°F
(Density Altitude	3000 ft)

Profile

Segment	Flight Mode	% Time	KTAS
Ascent	Climb, 300 fpm	12.0	90
Maneuver	Hover	25.5	0
Maneuver	Level Flight	25.5	40
Cruise	Level Flight	25	105
Descent	Descent, 300 fpm	12	110

Takeoff gross weight was 2600 pounds for the OH-6 (modified), and 2454 pounds for the OH-6A. These weights include a 493-pound payload.

For this mission, average fuel flows and endurances are:

	OH-6A	OH-6 (FIF)
Average fuel flow	140	149
Endurance	2.57	2.42

The modification thus reduces mission endurance by 6%.

At the nominal takeoff gross weight, sea level, standard day, comparison of performance is:

	OH-6A	OH-6 (FIF)
Gross Weight	2261	2417
OGE Hover Ceiling	9400	7300
Maximum Speed at Maximum Continuous Torque	128	120

Total horsepower required in hover at takeoff gross weight, S.L., standard day, is broken down as follows:

	OH-6A	Difference	OH-6 (Modified)
Total	220	15	235
Main Rotor	193	17	211
Drag Increase		(2)	
Weight Increase		(15)	
Antitorque	17	-3	13
Accessories	6	0	6
Gear Losses	4	1	5

Use of the ducted fan has actually reduced antitorque power. Rotor power, increased because of higher weight and vertical drag, more than accounts for the difference in total power.

Fan power could be reduced by elimination of the inlet screen, which would save two horsepower. It is doubtful if any weight reduction can be anticipated.

The breakdown of total power at 110 knots is:

	OH-6A	Difference	OH-6 (FIF)
Gross Weight	2261	146	2417
Drag, Aft Fuselage	18	13	31
Drag, Fin & Rudder	1	4	5
Tail Rotor H-Force	3	-3	0
Lift/Other Drag	132	7	139
Main Rotor Power	154	21	175
Tail Rotor/Fan Power	3	2	5
Accessory Power	6	0	6
Gear Loss	3	1	4
Total Power	166	24	190

The causes of the increases, and possible measures (if any) to reduce them, are:

1. Drag, Aft Fuselage. Although the fineness ratio of the fan-in-fin modification is better than that of the original OH-6A, the wetted skin area is considerably greater, the open inlet and exit areas add about 11 hp to drag power, and the rudder adds another four.

   A reduction in inlet screen area would increase fan losses in the hover low speed regime more than it would reduce drag power at high speed. It may be possible to louver the inlet, for a 3 hp saving. At 10 lb/hp, anything less than 30 pounds of additional weight might be justified. It would also be feasible to add a door to the exit areas; drag power would thus be reduced by 8 hp, with some increase in weight and complexity.

2. Drag, Fin and Rudder. The power required to react torque using the fan is greater than the power consumed in using the rudder.

3. Fan Power, Zero Airflow. No improvement anticipated.

4. Gear Loss. A function of total power required.

To sum up, it may be possible to reduce hover power by 3 hp and power at 110 knots by as much as 11 hp. Mission endurance would then be increased by about 0.05 hour.

CONTROL AND STABILITY PERFORMANCE

Control and stability characteristics of the modified OH-6 meet the requirements of MIL-H-8501A without stability augmentation and compare favorably in general to those of the OH-6A.

Control and stability characteristics, discussed fully in Appendix C, are summarized as follows:

Hover and Low-Speed Flight

1. Yaw response. Yaw damping cannot be analytically determined. Qualitatively, it is apparent that the greatly increased boom profile and the larger vertical tail surface will provide more non-viscous damping than the OH-6A. Yawing displacement of the air exit will change the relative velocity of the air jet and thus provide viscous damping, as the tail rotor does. It is judged that yaw damping of the OH-6 (FIF) is thus roughly equal to that of the OH-6A, but testing is required to verify this judgment. Response lag of the OH-6 (FIF) will be 0.6 second, considerably greater than that of the OH-6A, if output is programmed as a linear function of pedal input. This lag would not be a flight safety problem, and MIL-H-8501A does not establish any response lag limits. It would, though, be unacceptable to pilots. Nonlinear programming, specifically, the "quickening" technique must be used to reduce the effective lag to an acceptable value, comparable to that of the OH-6A. Testing will be required to establish the proper transfer functions.

2. Directional Control Power. Control sensitivity of the OH-6 (FIF) exceeds the MIL-H-8501A requirement, but is considerably less than that of the OH-6A (406 ft-lb/in. pedal versus 478). The lower sensitivity may be desirable, but if not, fan power limits are adequate to increase sensitivity by a simple control ratio change.

3. Lateral/Directional Dynamic Stability. Stability in this mode is as good as, and possibly better than, that of the OH-6A.

Cruising and High-Speed Flight

1. Directional Control. Control power is equal to that of the OH-6A.

2. Lateral/Directional Stability. Stability in this mode is expected to prove equal to that of the OH-6A.

3. Longitudinal Stability. As good as, and probably better than, that of the OH-6A.

CONCLUSIONS

GENERAL

The improvements in safety, maintainability, vulnerability, and the advantage of exhaust gas IR suppression to be gained through use of the fan-in-fuselage antitorque system can be realized with a relatively small decrease in endurance. The effect of this configuration on mission airspeed and hover ceiling requirements depends on installed power; in the case of the OH-6A, the baseline helicopter airspeed and hover ceilings, although reduced by the modification, satisfy the actual mission envelope as reported in Reference 20.

The fan-in-fuselage, as an antitorque system, requires less fan power than the tail rotor antitorque system. Tail boom weight is increased to obtain a favorable "disc" loading, and total power required is, therefore, higher. Drag is increased because of the air inlet screen, the exit openings, and use of the rudder in forward flight. The drag increment would be reduced by closing the exit openings in forward flight.

Control and stability were found to be equivalent to that of the baseline, tail rotor, helicopter. Safety is markedly improved by replacement of the tail rotor by the fully enclosed fan, and the use of low inlet velocity and moderate exit velocity virtually eliminates hazard to ground personnel. The aft end of the tail boom, with its simple, submerged airflow control vanes, is relatively immune to damage caused by contact with trees and other external objects. Vulnerability is reduced. The fan is shielded by other helicopter components in the forward quadrant, and peripheral armor can easily be provided. Vulnerable area of the aft controls is small. Noise generated by the fan is amenable to simple control measures, and the fan itself is not visible to radar.

The fan requires no flapping or lead-lag hinges, and operates in straight inflow at all times, and is judged to be more reliable and have longer life than a tail rotor. The flight control system is somewhat more complex than that of a tail rotor design. It is expected that the net effect of the fan-in-fuselage changes will be to improve both reliability and maintainability.

Use of the fan-in-fuselage for exhaust IR suppression was not investigated, although the relatively large mass flow required for antitorque control makes such use an attractive possibility. Utilization of fan-induced airflow for forward thrust is neither practical nor efficient. The main rotor, having a much larger area, is more efficient in providing forward thrust (provided it is not in the stall region), and a requirement to direct the fan induced flow aft as well as left and right would make the controls unacceptably complex.

AERODYNAMIC PERFORMANCE

Optimization of aerodynamic hover performance with a fixed tail boom length, depends primarily on a proper balance between exit area and gross weight. Fan power required decreases as the exit area is increased, but weight and rotor power increase with exit area. Antitorque thrust required will also increase as a function of rotor power and, therefore, of gross weight; but up to some point the reduction in fan power with increasing exit area outweighs the increase in rotor power with the exit area increase. In the case of the OH-6 (FIF), minimum total power was obtained with an exit area of 9 square feet, 72% of the OH-6A tail rotor area. The area could not, incidentally, have been made larger because of main rotor and flare angle clearance limitations.

Tail boom length was kept close to overall OH-6A length. An increase in the distance from the cg to the exit from 16 to 23 feet would reduce total power required by over two percent; a 4-foot increase would give a 1.4 percent reduction. Thrust required decreases, less fan power is required even with a smaller exit and, because the exit area can be decreased, there is practically no weight increase. Increase in distance to a tail rotor is less effective because the weight and cg effect is greater.

Total power required in hover is increased by 6.8%. This increase is due primarily to the gross weight increase of 6.4% caused by the large, constant cross section, tail boom. A small part of the increase is due to increased vertical drag, also caused by the large tail boom.

The fan-in-fuselage design is actually more efficient as an antitorque system than the tail rotor, requiring 14 as compared to 17 horsepower. Duct and turning losses account for 2.7 of the 14 horsepower. The duct/turning loss factor is thus seen to be relatively unimportant, as compared to the gross weight increase.

Drag is increased by 68 percent at 110 knots. The increase is due to the inlet screen, the exit openings, and the use of a rudder for antitorque control at airspeeds over 60 knots. Closure of the exit areas would improve drag, with a 2- to 3-percent reduction in power required at 110 knots, and a corresponding increase in endurance.

The full performance comparison is as follows:

	OH-6A	OH-6 (FIF)
Gross Weight	2261	2417
OGE Hover Ceiling	9400	7300
V_{max}	128	120
Endurance	2.57	2.42

RECOMMENDATIONS

The advantages of the fan-in-fuselage configuration in safety, dynamic component life and reliability, vulnerability, and its potential for exhaust IR suppression may, for many missions, more than offset the somewhat smaller endurance of the fan-in-fuselage. The endurance reduction is the only absolute penalty; the relatively lower ceiling may, as in the case of the OH-6A, still satisfy the mission requirements as determined by experience. In the case of the OH-6A, and possibly in other helicopters, the decrease in ceiling can be offset by installation of a later, completely interchangeable engine model with a higher power rating. Continuation of fan-in-fuselage investigation and development is, therefore, strongly recommended.

The following further investigation is recommended:

A. Wind tunnel investigation. The drag estimates made in this study can be confirmed by wind tunnel testing of a scale model. This investigation can include evaluation of the improvement which can be realized by closing off the exits in forward flight, over 60 knots, and by eliminating the inlet screen.

B. Experimental control response and duct loss evaluation. A full-scale model of the tail boom with fan, with the inlet area geometrically simulated, and with the yaw moment of inertia of the full helicopter simulated, can be tested on a rotating stand to confirm the analytically determined control power and response rate. This test may lead to improvements in the inlet and turning vane design.

C. Prototype flight test. Following satisfactory conclusions from wind tunnel investigation and experimental control response and duct loss evaluation, an OH-6A can be modified for full flight evaluation of the configuration.

Consideration of the full control system required for a pure fan-in-fuselage configuration also indicates possible advantages in a hybrid configuration study which was not within the scope of the contract.

Bidirectional lateral thrust is not required in powered flight. It is only required in autorotation for yaw control, and yaw control thrust requirements are transient and at a maximum are only 25 percent, approximately, the magnitude of antitorque thrust required.

If the fan-in-fuselage were not required to provide yaw control thrust, the system could be much simplified, and its losses (both duct and external drag) and weight would be reduced. These gains could be realized by the addition of a small fan in the fin. Weight of the fan and its drive would be roughly offset by savings in other parts of the total system. Mission endurance may be improved. The hybrid would yield, to a high degree, all the advantages of the pure fan-in-fuselage. Safety would be marginally lower, since the fan-in-the-fin is somewhat vulnerable to damage from protruding branches, but would be less vulnerable than in the case of the pure fan-in-fin, where the fan area must be considerably larger. The absolute advantage of exhaust gas IR suppression, not gained with pure fan-in-fin, would be realized.

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APPENDIX A
LITERATURE SEARCH RECORD OF APPLICABLE REFERENCES

Bibliography Reference No.	Reference Ident.	Reference Area of Interest: Turning Device	Loss Factor	Ducted Fans	Remarks
1.	AD447995		x		Section 3.9 Pressure losses applicable. Friction factors of Fig. 3.9.2.1a good for high Reyn. No.s. Ref. primarily for hydraulic (tubing) design
2.	SAE Manual	x	x	x	Section 1 Eng. Fund. fric. factors pg. A7, vane losses pg. A40-42 axial fan design consideration pg.H-23, valves pg. H-93
3.	AD723823		x		Inlet lip losses, internal duct losses, diffusion losses. Data mostly involved with supersonic flow. Section 2
4.	AD723824	x			Internal airflow relating to nozzles, ejectors. thrust reversers in section 8. High bypass Turbofan cowl data good section 10.
5.	Duct Hndbk	x	x	x	Data on inlet, duct and exit configuration losses Chap.III Axial fan design Chapt. 4 pg.7-183 including blade cascade data.
6.	A70-21323	x			Demonstrates that linearized jet flap theory used with momentum theory can predict cascade flow deflection K's
8.	A70-14890	x	x		Cascade data on ability of a jet flap blade and tandem blades to prevent separation when highly loaded giving defl. and press. loss
9.	A70-28087	x			Qualitative discussion on vectoring ADAM III air 90°. Contouring inner turn radius in lieu of vanes improved press. distribution across duct.
18.	A71-29453	x			Jet flap cascade perf. data showing that tangential blowing relying on Coanda air turning at rear radius of section is twice as effective as jet flap normal blowing. Shows flow ratios and deflection angles to 50°

Bibliography Reference No.	Reference Ident.	Reference Area of Interest: Turning Device	Loss Factor	Ducted Fans	Remarks
22	N72-14001	x	x	x	Sikorsky comparison of fan-in-fin, fan in tailcone and tail rotor. Describes system press. losses and est. pwr. requirements.
23	A72-16485	x			Good data on circular are vanes for turning $=\Delta B$ 45°
24	AD-731493	x	x	x	Lockheed comparison of fan-in-fin, fan in tailcone and tail rotor. Performance and wt. data on Lockheed 28C and advanced design.
25	A69-19573	x	x		Same info as Ref. 19 on ADAM III concept
26	ASME No. 58 A-4			x	Good ducted fan design theory for optimum fan sizing for both inlet guide vane fans and fans with exit straighteners.
27	JAE Apr. 62			x	Good info on ducted fan power relationships relative to duct losses in hover and forward flight.
29	Vert.R-80			x	Good info on ducted fan design relative to thrust/Hp for variations in hub to tip ratios and exit diffusion, etc.
30	TND-3122			x	Fan (4') in nacelle wind tunnel tests with varying blade with performance shown at zero advance ratio.
33	Hiller ARD 232			x	Survey of ducted fan literature and state of the art through 1959 - Mostly short ducts and props for VTOL
36	AD823150	x	x		Wind tunnel tests of a wing section with LE flap blowing and TE flap blowing. Lift & drag & flow coeff. for various config.

Bibliography Reference No.	Reference Ident.	Reference Area of Interest: Turning Device	Loss Factor	Ducted Fans	Remarks
40.	A68-33291 (AIAA)	x			Describes efficiency of fluidic diverter (control vents to atmos.) and fluidic diverter (backloading using values in each port).
41	AD476094	x			Shows that coanda deflection for the rig used depends on pri. press. ratios less than 2.0. Est. of turn eff. in terms of thrust.
42	CR-1341	x	x		Compares a cascade of tandem blades with plain blades under high load conditions. Also reported in Ref. 17.
44	UTIA No. 49	x			Subsonic tests of jet sheet parameters controlling flow over a 90° curved surface quadrant. No secondary flow considerations.
45	UTIA No. 51	x			Supersonic tests similar to ref. 80 showing in terms of force ratio for the parameters varied during test.
46	UTIA No. 79				Lift and drag of a primary coanda jet used to entrain secondary air while turning 90°, i.e., an ejector system showing augmentation.
47	TN-3904	x	x		Prop slipstream deflection over a two-step flap with blowing at 1st step. Blowing power vs. deflection shown.

APPENDIX B
OH-6 (FIF) AERODYNAMIC PERFORMANCE

This concept uses a buried fan to create thrust needed for yaw control. The fan is mounted axially under the hollow tail cone. Controllable vanes at the end of the tail section determines the magnitude and direction of the sidewards thrust produced. This system takes the place of the conventional tail rotor system.

OH-6A was selected as the baseline aircraft to study the effect of this concept on performance and weight. OH-6A is a well tested and operationally proven helicopter for which a good performance base exists. In this study the referenced OH-6A operators manual was used for performance basis.

The OH-6A is an all-metal, single-engine helicopter with a conventional main rotor tail rotor arrangement. The engine in the operational aircraft is the Allison T63-A-5A engine.

Principal dimensions of the baseline aircraft (OH-6A) and the OH-6 (FIF) are:

Aircraft dimensions	OH-6A	OH-6 (FIF)
length (main rotor extended)	30ft 3.7 in.	36ft 1 in.
length (main rotor folded)	22ft 9.5 in.	28ft 7 in.
width (fuselage)	4ft 6.4 in.	same
width (landing gear tread)	6ft 9.2 in.	same
height (to top of upper vertical stabilizer	8ft 6 in.	8ft 10 in.
height (to center of rotor)	8ft 1.5 in.	same
main rotor diameter	26ft 4 in.	same
tail rotor diameter	4ft 3 in.	not applicable

Aircraft Weights (lb)	OH-6A	H-6/FIF
basic weight	1154	1310
fuel capacity (61.5 gallons of JP-4)	400	same
max. certified (FAA)	2400	not applicable
max. structural	2700	not applicable

Power (hp)		
engine rated takeoff power (SL15°C)	317	same
engine rated max. continuous power (SL, 15°C)	270	same

Power	OH-6A	OH-6 (FIF)
transmission 5 minute limit @ 103%N_z	260	same
transmission max. cont. limit @ 103%N_z	221	same
Operating conditions and limits		
rotor speed (103%N_z), rpm	483	same
tip speed (103%N_z), fps	666	same
airspeed, structural limit KCAS	130	-
airspeed, stall with margin applied	flt man. chart	same
Yaw control device parameters		
diameter - tail rotor/fan	4ft 3 in.	3ft 6 in.
solidity	.12	.081
rotational speed (100% N_z), rpm	3030	3547
tip speed, (103%N_z), fps	694.4	650

The mission of the OH-6A helicopter is observation, target, acquisition, reconnaissance and command control. Mission performance parameters have been obtained in Vietnam from three specially instrumented OH-6A helicopters under actual combat conditions. Data is comprised of rpm, speed, weight altitude and other recordings taken over a period of 200 flight hours as reported in Reference 20. This data was used to define the mission profile for analysis of the FIF concept.

The combat data was grouped so as to allow a realistic comparison of the OH-6 (FIF) with the OH-6A to be made. The data is summarized as follows:

Average flight conditions

altitude 1000 ft
temperature 90°F
(density altitude 3000 ft)

OH-6A mission profile

segment	flight mode	% time	KTAS	GW
ascent	climb at 300 fpm	12	90	2600
maneuver	hover	25.5	0	2400
maneuver	low-speed level flight	25.5	40	2400
steady cruise	high-speed level flight	25	105	2400
descent	descent at 300 fpm	12	110	2200

The above profile represents a grouping of flight mode, airspeed and gross weight so that all presented flight conditions of the Vietnam statistics would be represented in proper proportions. The weights, of course, apply to the OH-6A standard configuration. The weights will change when the basic weight of the aircraft is changed as is the case with the OH-6 (FIF.)

The power required for the OH-6A at 460 rpm ($100\%N_Z$) is shown in the Operators' Manual. The statistical OH-6A operation experience indicates that the aircraft was rarely operated at this rotor speed and that the average rotor speed appeared at the posted maximum rotor speed limit of 483 rpm ($103\%N_Z$). The Operators' Manual data was correspondingly adjusted for the increased rpm using the standard rotor performance relationship. The resulting hover total power required is shown in Figure B-1. Forward flight power required is shown in Figure B-2.

For the fan-in-fuselage configuration, the tail rotor is removed, the tail cone is enlarged and an internal fan is installed inside the tail cone. These changes result in weight, power and drag changes. From Figure 16, the net weight increase due to the fan-in-fuselage installation is 154 pounds.

In hover at a given main rotor power there is a net power required difference due to replacing the tail rotor by the internal fan. The tail rotor power required was estimated by a computer analysis to be about 9% of main rotor power. The main rotor power is related to the total power as follows:

$$P = (P_r + P_t + 6) / .98$$

where 6 hp is the estimated accessories power and .98 is the estimated drive system mechanical efficiency. OH-6A main rotor power estimated using the above relationship is shown in Figure B-1.

In addition to the change in antitorque power at any given main rotor power there is also a shift in main rotor power and lift relationship. This shift is due to an estimated increase in vertical drag by 1% of lift. This increase is small relative to the increase of the tail cone, mainly because the increase in Reynolds number of the airflow over the tail cone. The high Reynolds numbers cause the airflow to become super-critical, resulting in halving of the drag coefficients. The decrease in lift for the OH-6(FIF) configuration is shown in Figure B-1.

The total power for the OH-6(FIF), as shown in Figure B-1, includes the same accessories power and mechanical efficiency as that of the OH-6A. The tail rotor power however, is replaced by fan power, estimated here as:

$$P_f/\tau = .0044\,(P_r/\tau)^{1.5}$$

An illustration of relative power levels of the standard and the FIF concepts is given in Table B-1.

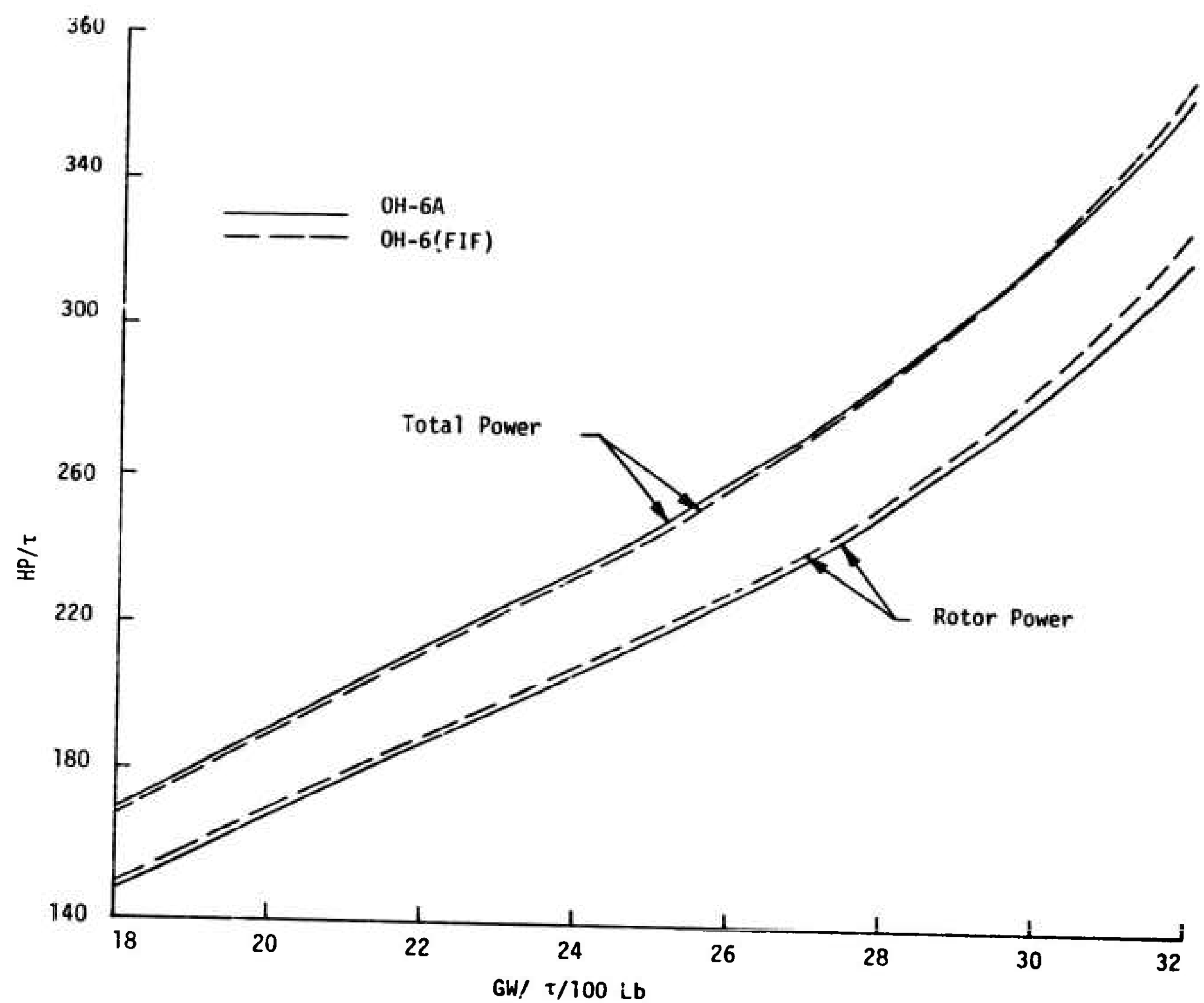

Figure B-1. Variation of Hover (OGE) Power Required With Gross Weight.

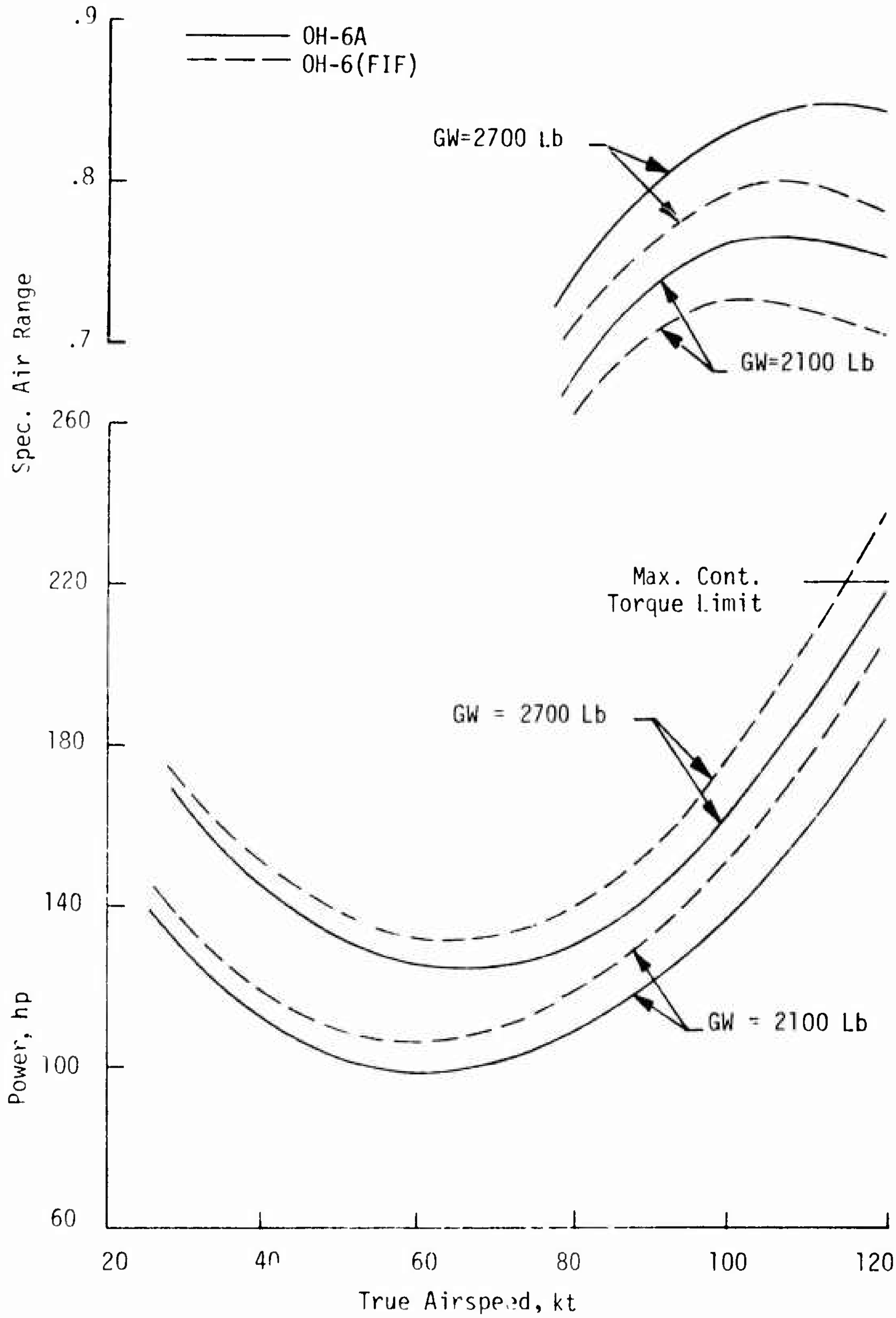

Figure B-2. Variation of Power and Specific Air Range With Airspeed.

TABLE B-1. COMPARISON OF HOVER POWER REQUIRED (OGE)

	OH-6A	H-6/FIF
Gross Weight, lb	2261	2417
Main rotor power increase due to increased vertical drag, hp		2
Main rotor power increase due to gross weight increase, hp		16
main rotor power, hp	193	211
antitorque power, hp	17	13
accessories power, hp	6	6
gear losses, hp	4	5
Total power, hp	220	235

In forward flight, yaw control of the H-6/FIF is provided by the rudder. The fan is operated at a no-thrust level and absorbs power for windmilling only. Tail rotor and tail rotor H-force power are deducted.

The biggest increase of forward flight power required of the H-6/FIF over the standard OH-6A configuration stems from the drag increase of the enlarged aft fuselage. The corresponding power required differences are calculated using a propulsive efficiency of 0.9. The total effect of these changes on power required is seen in Figure B-2 at two representative gross weights.

The individual changes are tabulated in Table B-2 at a typical cruise speed, close to the speed for maximum range. The values apply for sea level standard day conditions.

TABLE B-2. COMPARISON OF POWER REQUIRED AT 110 KNOTS

	OH-6A	OH-6(FIF)
Gross Weight, lb	2261	2417
Items by which the propulsive power requirements differ between the two configurations		
aft fuselage drag, hp	18	31
vertical fin and rudder drag, hp	1	5
tail rotor H-force, hp	3	0
Power increases due to increased gross weight, hp	-	7
Main rotor power, hp	154	175
tail rotor power/fan windmilling power, hp	3	5
accessories power, hp	6	6
gear losses, hp	3	4
Total shaft power, hp	166	190

PERFORMANCE PENALTIES

Hover ceiling capability at a given gross weight is not affected by the fan-in-fuselage installation. This is illustrated by Figure B-3, for standard day and 95° day conditions. The hover ceilings are applicable for the OH-6A equipped with 317 hp T63-A-5 engine. Installation of an uprated T63 engine would increase the ceiling, but the performance difference between the OH-6A and the OH-6(FIF) would be about the same.

In forward flight the increased drag reduces maximum speed whenever stall limits are not reached first. Figure B-4 shows maximum speed as limited by maximum continuous transmission torque limit at 103% rotor speed. The OH-6 (FIF) is seen to have about 6 knot lower airspeed as the result of the increased power required. Also shown is a stall boundary at 103% rotor speed and with a margin applied. The curves are for sea level standard day. The proximity of the stall curve to the transmission limited curve suggests that at altitude the stall limited airspeeds are below the transmission limited airspeeds for either concept.

Range capability is also reduced by the increased drag of the fan-in-fuselage installation.

Figure B-2 shows this in terms of specific air range at sea level standard day conditions. The range is seen to be reduced by about 5-1/2% and the speed for maximum range is reduced by about 6 knots.

Net effects of power, drag and weight changes are illustrated in Table B-3. The first column shows performance of the OH-6A with a typical payload, two crew men and full fuel. Previously discussed hover ceiling and maximum speed are listed. The fuel flow shown is for the generalized mission based on Vietnam combat experience. Thus, it is not at the gross weight listed in the table, but is based instead on a prorated gross weight schedule. The detail profile was defined earlier. The average mission time reflects the preceding average fuel flow for the tabulated fuel load, less 10% reserve.

The second and third columns show performance for the OH-6 (FIF), but at the OH-6A gross weight. The increased empty weight of the fan installation is compensated by off-loading payload or fuel. Hover ceiling is seen to be unaffected by the fan installation, but the air speed is reduced by the increased drag of the aft fuselage. Average fuel flow is increased somewhat, but the average mission time is severely affected only when fuel is off-loaded to compensate for empty weight increase.

Final column shows performance at the original OH-6A useful load. The gross weight is increased by the fan installation weight increment. Hover ceiling is reduced due to the gross weight increase by 2100 feet on a standard day. (Reduction is 1400 feet at 95% as seen from Figure B-3.)

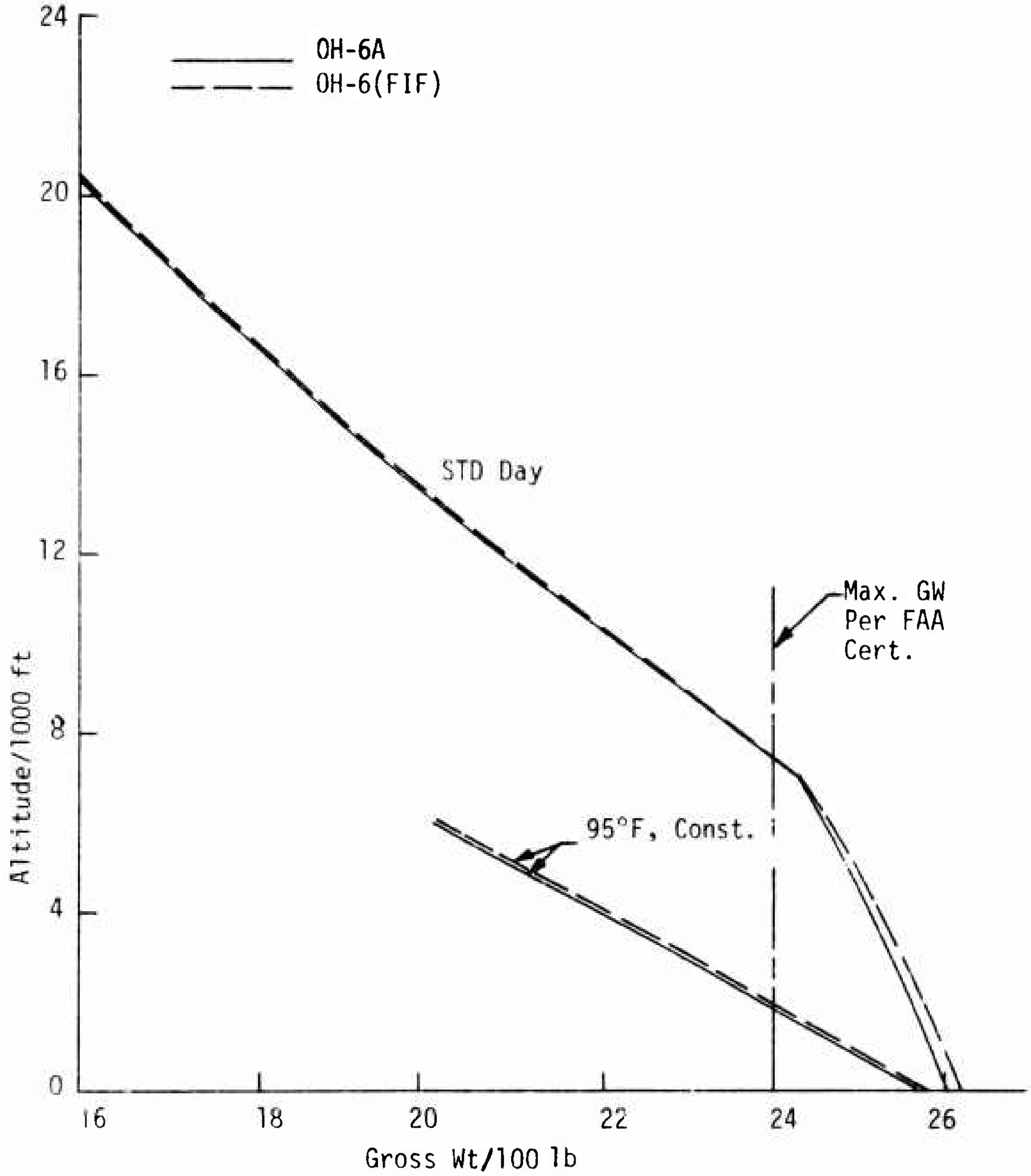

Figure B-3. Hover Ceilings, OGE.

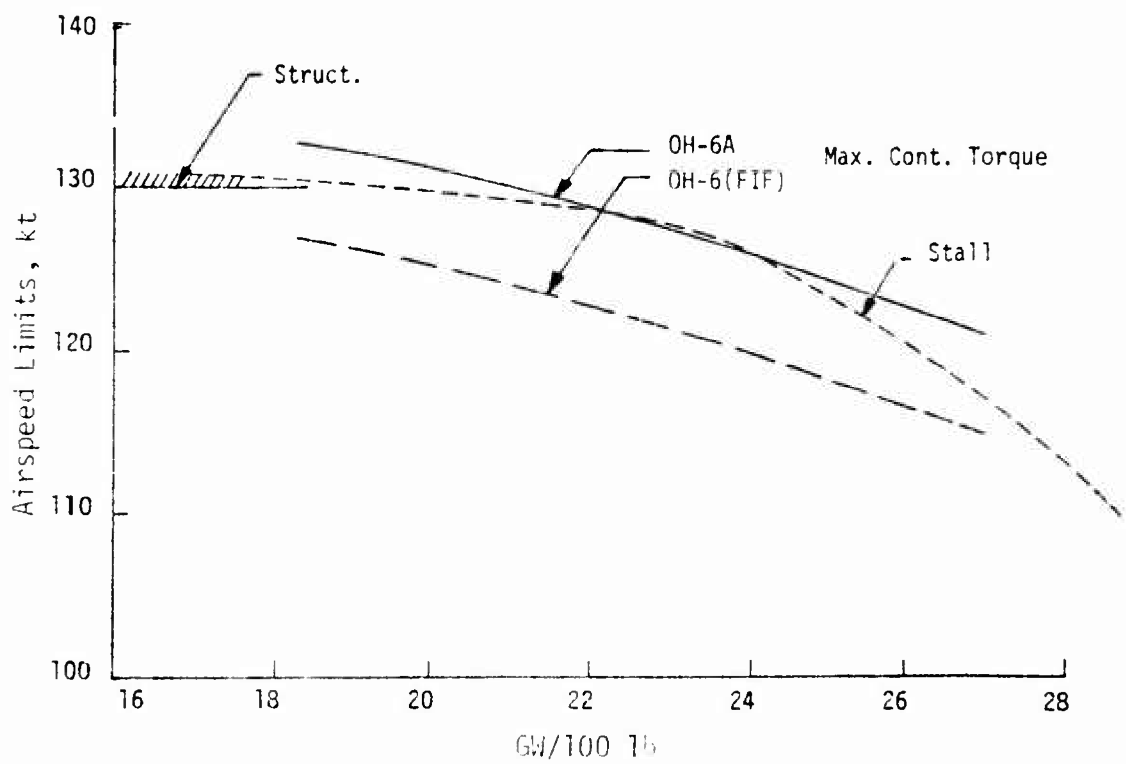

Figure B-4. Airspeed Limits, OH-6A and OH-6(FIF).

The average mission fuel flow and time are penalized by about 6% as compared to the original OH-6A performance.

TABLE B-3. EFFECT OF FAN INSTALLATION ON PERFORMANCE

		OH-6A with internal fan		
		Constant Gross Weight		Increased Gross Weight
	Operational OH-6A With 300-lb Payload and Full Fuel	Reduced Payload	Reduced Fuel Load	With 300 lb Payload and Full Fuel
Basic Weights	1154	1300	1300	1310
Oil	7	7	7	7
Crew	400	400	400	400
Fuel	400	400	354	400
Payload	300	154	300	300
Gross Weight	2261	2261	2261	2417
At Above T.O.G.W				
OGE Hover Ceiling Std. day (ft)	9400	9400	9400	7300
Maximum speed Max. Cont. Torque SL, Std. Day (kt)	128	122	122	120
At Typical Vietnam combat conditions (1000 ft 90°F, GW & Airspeed Spectrum per statistics)				
Average Fuel Flow (lb/hr)	140	144	146	149
Average mission time (hr)	2.57	2.5	1.57	2.42

APPENDIX C
PRELIMINARY STABILITY AND CONTROL ANALYSIS
FAN-IN-FUSELAGE DIRECTIONAL CONTROL CONCEPT

Introduction

The prime objectives of the stability and control analysis are to establish the capability of the directional control concept to meet pertinent requirements of applicable military specifications and to anticipate any substantial deviations from the characteristics of the unmodified aircraft soon enough that they may be taken into account in subsequent planning. While the major changes in handling characteristics can be expected to occur in the lateral/directional sense, particularly in hover and at low airspeeds, the fan-in-fuselage installation results in aerodynamic and inertial changes which also require a brief consideration of longitudinal dynamics.

Major Physical Changes

Externally, the principal changes in the basic OH-6A required to accommodate the fan-in-fuselage design are:

1) removal of the fuselage from Station 137.5 aft of the tail boom, and of the tail rotor assemblies and replacement by the buried fan, air inlets, ducting, and control hardware

2) redesign and relocation of the horizontal stabilizer to the top of the vertical fin

3) replacement of the tail rotor pylon with a fin/rudder assembly with an area of 9.56 ft^2

These changes, and others which do not alter the outward appearance of the aircraft, result in an incremental increase in gross weight of 14.6 pounds and a c.g. travel 4.95 inches aft of its location at the takeoff gross weight of 2407 pounds.

Replacement of the tail rotor by the fan-in-fuselage will have its most noticeable impact on handling qualities in hover and low speed flight, for it is here that directional control is most essential. Not only must turning power be available for maneuver, but additional force to react main rotor torque, which is near its maximum, is also required. Handling qualities most likely to be changed by the fan-in-fuselage installation are the response to directional control input and changes in control-fixed lateral directional stability. At high speed the vertical empennage is actually an integral and active part of the total directional control and stability package and it can be designed to take over all or part of these functions. The significant design problem here is establishment of an acceptable fin/rudder geometry with a suitable scheduling of the two directional control methods.

Fan-in-Fuselage Characteristics

The low-speed and hover flying qualities will be impacted most noticeably by two properties which may be affected by the fan-in-fuselage concept. The first of these might be called pneumatic lag. Essentially, directional control is obtained in hover and at low airspeeds by increasing or decreasing the fan blade pitch, thus altering the mass flow of air in the duct by changing its velocity. Since this cannot occur instantaneously, because of air inertia and the duct losses, a delay in the development of the yaw control moment may be expected.

A very simplistic approach to the determination of the order of magnitude of this control lag, which assumes a first-order response of duct velocity to a step pedal input, indicates that the pneumatic time constant is less than 0.6 second, based on a mass flow of 1.9 slugs/sec and total duct losses of 3.5 lb/ft^2. A second change in aircraft characteristics stemming from the fan-in-fuselage installation is the yaw damping which is generated in two ways. First, the ducting and the vertical empennage present a large flat plate area to the rotational velocity; second, the rotational motion alters the fan slipstream velocity in somewhat the same manner that tail rotor inflow is affected. The resultant yawing moment is in a direction which opposes the yawing velocity, hence it can be considered a damping moment. That portion due to the duct and vertical empennage can be visualized as the total effect of the incremental drag forces generated by the yaw rate of the helicopter about its center of gravity. In this context it can be shown that the moment derivative varies linearly with the yaw rate. It is, therefore, not a true viscous damping. A precise evaluation of this contribution to yaw damping can best be obtained from wind tunnel testing, but in any case it can be concluded that it will be substantially greater than that provided by the standard tail boom, owing to the much larger flat plate area exposed to the rotational velocity. The second damping effect, that due to momentum change in the fan slipstream, might be considered similar to that provided by the standard tail rotor as a first approximation.

Handling Characteristics in Hover and Low-Speed Flight

Pneumatic Lag and Yaw Damping

One might suspect that, from the pilot's point of view, the most noticeable change in the OH-6 handling characteristics due to the fan-in-fuselage installation will arise because of the anticipated lag in the directional control input. This point has been recognized in Reference 4, which states that "the first (problem) will be a delay in response time due to losses in the ducting, creating serious problems in meeting

MIL-H-8501A requirements". Although no indication of the magnitude of the control lag that might be expected was given in the reference, if we accept 0.6 second as a reasonable figure, we can assess its effect quite readily, for a detailed study of the effect of directional control lag on pilot rating of VTOL handling qualities has been reported[21]. This work, which was accomplished by pilots "flying" both fixed and moving base simulators whose characteristics could be widely varied, covered control time constants of up to 0.6 second at various levels of yaw damping, with and without time delays. The significant conclusion reached with regard to directional characteristics confirms the intuitive thought that pilot rating should deteriorate with increasing first order lag, but it was also found that pilot opinion was also strongly influenced by yaw damping. These points are illustrated in the figure appearing below, excerpted from Reference 21.

From this figure, the unmodified OH-6, which has a yaw damping ratio, N_r, of about 1.0, would qualify for a "satisfactory" pilot rating with a control time constant as high as 0.3 second. With a time constant of 0.6 second a Cooper rating higher than 4.5 would make the airplane unacceptable for normal operations although a successful landing could be accomplished. It becomes clear, therefore, that in redesign of the OH-6 to accommodate the fan-in-fuselage concept, provisions must be made to assure that the yaw damping level be maintained and control lags be minimized. As [illegible] pointed out previously, the airframe damping of the fan-in-fuselage configuration will be substantially increased over the standard OH-6; [illegible] rotor damping will remain essentially the same for either configuration, so the problem centers around the comparative behavior of the conventional tail rotor and the fan-in-fuselage. Calculation of tail rotor damping in hover is a well-known technique; if it is assumed that the same damping is achieved by the fan-in-fuselage concept, directional control time constants of 0.6 second lead to handling characteristics which would prejudice evaluation of the fan-in-fuselage concept. In view of this, it appears advisable to introduce the "quickening" technique, which effectively reduces the control time constant, into the fan-in-fuselage control system design. In its simplest implementation the concept works substantially as outlined below.

The pilot adds control (as a step pedal input), but the control moment, because of the pneumatic lag, builds up along the solid line with a trim constant, τ_0. If an increment of control is added only while the pilot's pedal is actually moving, the control buildup appears to occur along the dashed line with the same time constant τ_0. Relative to the pilots' input, however, the time constant is reduced to τ_1. The mechanism for incorporating the qhickening scheme into the control system is available, but precise definition of the necessary transfer functions depends upon

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[21]Vinje, Edward W. and Miller, David, P., FLIGHT SIMULATOR EXPERIMENTS AND ANALYSES IN SUPPORT OF FURTHER DEVELOPMENT OF MIL-F-83300-V/STOL FLYING QUALITIES SPECIFICATION, United Aircraft Research Laboratory AFFDL-TR-73-34, Air Force Flight Dynamics Laboratory, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, June 1973.

acceptable definition of the control system lag and overall helicopter yaw damping. Since both these factors are influenced significantly by the configuration, a final version of the quickener should be delayed until they can be measured.

Directional Control Power

Control sensitivity of the OH-6 helicopter has been measured at 46 deg/sec^2 in. pedal at a gross weight of 2100 pounds[3]. This converts to 478 ft-lb/in. pedal. The fan-in-fuselage maximum directional control moment in hover is 3090 ft-lb which, with the standard OH-6 pedal displacement of 7.6 inches, results in a control sensitivity of 406 ft-lb/in. pedal. This, combined with increased yaw inertia, reduces control sensitivity of the fan-in-fuselage configuration to about 24 deg/sec^2. For the standard OH-6, however, the yaw displacement at the end of 1 second was 20 degrees, substantially above the MIL-H-8501A requirement of 8.5 degrees. For the fan-in-fuselage configuration, with the yaw acceleration about half the level measured on the OH-6 we might expect a yaw displacement of 10 degrees assuming constant angular acceleration. Since this is only about 2 degrees higher than the minimum required by MIL-H-8501A, we should expect that the modified helicopter may be marginal in this respect assuming the same lag in the directional control moment as occurs with the tail rotor type control. If the lag is greater, it may be necessary to increase control thrust by a control ratio change.

Lateral/Directional Dynamic Stability

The uncoupled control-fixed lateral/directional stability of the OH-6 in hover is calculated to be positive in all modes. The modified fan-in-fuselage configuration, because, as previously discussed, yaw damping should be the same or better than the standard configuration, is expected to show no significant changes, but in order to evaluate the effect of yaw damping, we can consider the worst case, where the only yaw damping is provided by the main rotor. Here the spiral mode for the fan-in-fuselage configuration remains lightly stable (6.1 seconds to half amplitude) and the Dutch roll mode goes from lighly stable (23 seconds to half amplitude) to lightly unstable (27 seconds to double amplitude). Thus, if we completely ignore any damping due to the fan-in-fuselage installation, all the motions are docile and can be handled easily by the pilot without stability augmentation.

Handling Characteristics With Airspeed

Directional Control

At speeds above 60 knots the vertical empennage can take over both the control and stability functions. Figure 9, which shows the control requirements with rudder control capability superimposed, illustrates the feasibility of rudder/fin control and stability. From this it can be seen that, for level flight at 50 knots, a 30-degree rudder deflection is adequate for trim but provides no margin for maneuver; at 62 knots, both trim and maneuver requirements can be satisfied with a maximum rudder deflection of 30 degrees. Thus, the transition region where directional control is transferred from the fan to the aerodynamic surfaces lies between these two speeds where the blade pitch of the fan is reduced, duct flow is reduced to zero, and rudder deflection is introduced.

Lateral/Directional Stability

The table below compares the pertinent stability derivatives for assessment of the impact of the fan-in-fuselage on OH-6 lateral/directional stability at speeds near 60 knots and above. Note that the static directional stability derivative and the yaw damping have been divided by the respective yaw inertia terms so that direct comparison can be made.

AIRSPEED (KT)	OH-6 N_v	OH-6 N_r	OH-6 MODIFIED N_v	OH-6 MODIFIED N_r	ROOTS OF LATERAL/DIRECTIONAL CHARACTERISTICS EQUATION - OH-6 HELICOPTER		
57.5	.077	-1.155	.052	- .720	-3.181	-1.732	$-.0854 \pm .248$
76.7	.086	-1.275	.067	- .960	-3.654	-2.018	$-.0830 \pm .267$
75.7	.092	-1.361	.086	-1.201	-4.338	-2.112	$-.0732 \pm .299$
115.1	.096	-1.429	.104	-1.441	-4.901	-2.498	$-.0534 \pm .360$

This tabulation shows that the OH-6 is stable throughout the flight speed range. We can expect that the sideslip derivatives and the yaw damping will have minimal impact on the highly stable roll subsidence since the major contribution to roll moments is from the main rotor. At speeds between 60 and 80 knots we should expect some deterioration in both the spiral and the Dutch roll modes, but the spiral mode is so heavily damped that it will present no problem in any case. The Dutch roll is only lightly damped, but its period is so long(23 sec) that, even if it were to go unstable, it could be easily controlled by the pilot. It should be noted, also, that the preceding tabulation is conservative in that the derivatives for the modified configuration are calculated neglecting the contribution of the ducting. Were this accounted for, the control-fixed lateral/directional stability characteristics of the fan-in-fuselage would be indistinguishable from those exhibited by the standard OH-6.

Longitudinal Dynamic Stability

The longitudinal control-fixed stability of the modified OH-6 was examined very briefly. The significant derivatives would not be expected to change markedly for the fan-in-fuselage configuration being a function primarily of changes in horizontal stabilizer design. The longitudinal characteristic equation for hovering flight has a complex root with a positive real part indicating a mild unstable oscillation; at an airspeed slightly above 38 knots the complex root becomes two real roots, one of which is positive with a time to double amplitude of nearly 6 seconds. This characteristic continues as airspeed builds up, as shown in the tabulation below.

AIRSPEED (KT)	ROOTS OF LONGITUDINAL CHARACTERISTIC STABILITY EQUATION		
0	-1.470	0.027 ± .402	-0.413
38	-1.311 ± 1.52	0.004 ± .112	
77	-1.752 ± 2.77	0.120	-0.141
115	-2.335 + 3.56	0.126	-0.171

The experimental results pertaining to longitudinal stability as reported[22] are in general agreement with the above calculated longitudinal stability characteristics. A short period pitch oscillation in hover was demonstrated experimentally, which is not predicted analytically; probably because pitch damping is overestimated. With airspeed, however, the heavily damped pitch oscillation is evident along with a gently divergent oscillation which ultimately becomes a pure divergence which qualitatively agrees with results noted in Reference 22. Here it is reported that at airspeeds above 105 knots, the OH-6 helicopter has a pitch-up tendency which requires substantial pilot effort to counteract. The fan-in-fuselage configuration may improve this situation somewhat if it can be considered to provide added angle-of-attack stability. Again, the final decision is but delayed until reliable test data is available. In any event the longitudinal stability is at least as good as, and possibly better than, it is for the standard OH-6 helicopter.

[22] Statum, Connie M., and Anderson, William A., PART ONE OF TWO PARTS REPORT OF THE ENGINEERING FLIGHT TEST - STABILITY AND CONTROL PHASE OF THE OH-6A HELICOPTER, UNARMED (CLEAN) AND ARMED WITH THE XM-7 OR XM-8 WEAPON SUBSYSTEM, USATECOM Project No. 4-3-0250/51/52153 DA Project No. IR141803D168, U. S. Army Aviation Test Activity, Edwards AFB, California, 1964.

Conclusions

This preliminary stability and control analysis has considered briefly only those features of the fan-in-fuselage directional control concept as it is currently conceived that deviate obviously from the standard tail rotor control installation on the OH-6 helicopter. Nonetheless, several conclusions which are believed to cover the most significant changes in aircraft handling qualities that may be expected are listed.

1) The modified aircraft can be flown through a technically productive flight test program without a stability augmentation system.

2) Some means of compensating for a directional control lag will be required to avoid deterioration of pilot rating of handling qualities to an unacceptable level. Further analysis of the pneumatic lag and yaw damping is required to establish reliable data for a first cut at a directional control quickener.

3) The flight test vehicle, as currently designed, is expected to be marginal in terms of directional control power available to meet requirements of MIL-H-8501A in hover even with a control quickener installed; but control power can easily be increased.

4) The proposed vertical empennage appears to be suitable for both control and stability functions at speeds above 60 knots.

5) Longitudinal dynamic stability may be somewhat improved by the fan-in-fuselage installation.

LIST OF SYMBOLS

A_d	Area, equivalent frontal, ft^2
A_e	Area, exit, ft^2
A_f	Area (net), fan, ft^2
A_n	Area at point n in duct, ft^2
C_l	Integrated lift coefficient, fan
d	Duct diameter, ft
D_f	Diameter, fan, ft
D_f	Diameter, fan hub, ft
f	Friction loss factor
I_y	Moment of inertia, yawing slug-ft^3
K	Duct loss factor, total referred to exit velocity
K_n	Duct loss factor at point n
n_f	Fan efficiency
N	Rotor speed, rpm
N_r	Rate of change, yaw moment/yaw
N_v	Rate of change, yaw moment/sideslip velocity
N_z	Tail rotor or fan speed, rpm
p_d	Pressure loss, psf
P	Power required, total, hp
P_d	Power, drive system losses, hp
P_f	Power required antitorque, fan, hp
P_r	Power required, main rotor, hp
P_x	Power, auxiliary, hp

Q	Rotor torque, ft-lb
r_f	Fan radius, ft
R	Distance from cg to exit, ft
T	Exit thrust, total, lb
T_c	Exit thrust, yaw control, lb
T_t	Exit thrust, antitorque, lb
V_e	Exit velocity, fps
w	Airflow rate, lb/sec
W	Gross weight, lb
ρ	Air density, slugs/ft^3
τ	Density ratio
τ_o	Trim constant
θ	Yaw displacement, deg

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