JET THRUST AS A COUNTER-TORQUE

FORCE FOR SINGLE-ROTOR, SHAFT-

POWERED HELICOPTERS

A THESIS

Presented to

the Faculty of the Graduate Division

by

Robert L. McDaniel

In Partial Fulfillment

of the Requirements for the Degree

Master of Science in Aeronautical Engineering

Georgia Institute of Technology

June, 1961

JET THRUST AS A COUNTER-TORQUE

FORCE FOR SINGLE-ROTOR, SHAFT-

POWERED HELICOPTERS

Approved:

Date Approved by Chairman: *May 25, 1961*

ACKNOWLEDGMENTS

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF SYMBOLS

A   Cross-sectional area of engine exhaust (A_E) or counter-torque jet (A_j), square feet

F   Jet thrust of counter-torque jet (F_j) or engine exhaust (F_E), pounds

ℓ Tail boom length measured from rotor shaft to counter-torque thrust line, feet

m   Mass flow of engine exhaust (m_E), secondary air (m_s), or counter-torque jet (m_j), slugs per second

P   Power of engine exhaust (P_E), of jet (P_j), delivered to rotor (P_R), foot-pounds per second

r   Radius of a nozzle or rotor wake, feet

R   Radius of a rotor, feet

T   Thrust of rotor, pounds

η Augmentation factor defined with Equation (4.2)

ρ Mass density of flow in engine exhaust (ρ_E), secondary air (ρ_s), or free air (ρ), slugs per cubic foot

τ Torque of rotor, foot-pounds

Ω Rotor angular velocity, radians per second

SUMMARY

The disadvantages of the counter-torque rotor system used on current single-rotor helicopters are discussed. It is shown that performance, reliability, and safety of single-rotor helicopters are degraded by the mechanical and aerodynamic complexity, the power requirement, and the exposed position of counter-torque rotors. Specific areas of possible improvement in counter-torque systems are listed.

A counter-torque jet system is considered using residual jet thrust from turbine engines to replace standard counter-torque rotor systems. The magnitude of residual jet thrust for helicopter turbine engines is compared with the thrust required for counter-torque balancing and yaw control. It is shown that the amount of available thrust is less than required and various methods of thrust augmentation are investigated.

It is shown that thrust augmentation by tail pipe constriction and by fan acceleration of exhaust efflux require a prohibitive loss of available shaft power. Static ejector augmentation is investigated and shown to present insufficient augmentation of the available thrust. After-burning is shown to present mechanical and operational complications as well as presenting insufficient augmentation.

The power requirements of a fan augmenter are investigated and it is shown that sufficient thrust can be provided at a power expenditure comparable to that of present counter-torque rotors.

The use of a vertical tail-rudder combination to provide a

counter-torque force during cruise conditions is investigated. It appears that the vertical tail can provide counter-torque balancing with greater reliability and with less power than the counter-torque rotor system. It is shown that the rudder can provide full yaw control at flight speeds from the maximum flight speed to flight speeds below one-half the design cruising speed. The use of the vertical tail permits use of the counter-torque jet thrust for propulsion during cruising flight.

The use of the jet thrust for propulsion reduces the parasite power requirements of the main rotor. Further, fuselage and rotor disk tilt can be lessened resulting in reduced induced power and blade profile drag power required by the rotor. Reduced rotor disk tilt will result in less retreating blade stall for high speed flight conditions.

Control response of the counter-torque jet is considered and it appears that adequate response can be available if the thrust control is applied at the nozzle. Control cross-coupling will be increased if the jet thrust is turned for yaw control.

The U. S. Army HU-1B helicopter was selected as a standard of comparison and used to evaluate the relative merits of the counter-torque rotor system. The investigation indicates that an improvement in range, speed, and a reduction in mechanical complexity could result from the adaption of a counter-torque jet system.

CHAPTER I

INTRODUCTION

Single rotor helicopters have evolved into an almost standard configuration with a large lifting rotor and a small counter-torque rotor. The lifting rotor is normally shaft driven and provides propulsion as well as lift. The counter-torque rotor balances torque transmitted to the main rotor and provides yaw control. The counter-torque rotor is shaft driven and is located at the rear of the helicopter, with its axis of rotation normal to the longitudinal and vertical axis of the helicopter (1)[*].

Although much experimental work has been done on tip-driven rotors in order to eliminate the counter-torque problem, no fully successful, operational helicopter currently uses a tip-driven rotor. At this time, the only successful recourse to the use of a counter-torque rotor is the use of multiple lifting rotors.

Despite its wide acceptance on single rotor helicopters, the counter-torque rotor contributes neither to the helicopter's lift nor propulsion and degrades performance and reliability by its power requirements and mechanical complexity. A qualitative examination of some of the disadvantages of current counter-torque rotor systems justifies the investigation of alternate systems to indicate areas where performance or reliability might be improved.

---

[*]Refers to items in Bibliography

The power required by the counter-torque rotor system is a function primarily of the power transmitted to the main rotor, the angular velocity of the main rotor, the distance the counter-torque rotor is located from the main rotor shaft, the size of the counter-torque rotor, and the mechanical and aerodynamic power losses in the counter-torque rotor system. In the hover, the counter-torque rotor system requires 7-10 per cent of the power delivered by the engine (2). Due to cyclic flapping in the forward flight condition, the plane of rotation of the counter-torque rotor is normally tilted, as shown in Fig. 1.
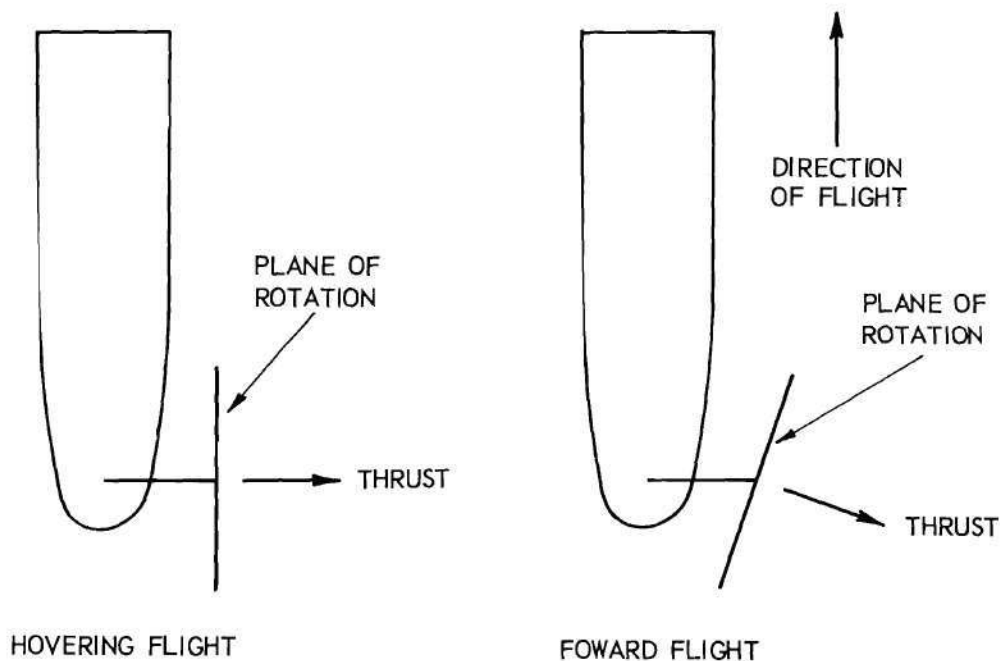

Fig. 1 Counter-Torque Rotor Incidence Due to Flapping

This tilt of the plane of rotation reduces power requirements of the counter-torque rotor, but it also tilts its thrust vector so as to con-

stitute a drag force, thereby reducing range. Some helicopters have inclined flapping hinges or other mechanical devices to reduce this rearward tilt of the counter-torque rotor's thrust vector and thus its drag. This results in a continued high-power expenditure. Most current helicopters accept the increased drag in exchange for the reduced power requirements (3).

The mechanical complexity of the counter-torque rotor system is perhaps its greatest disadvantage. The orientation of the counter-torque rotor, with respect to the flow in forward flight, prevents the use of a simple propeller and requires the use of a fully articulated rotor capable of free cyclic flapping. Variation in the counter-torque requirements and use of the counter-torque rotor for yaw control require the rotor to be capable of collective pitch change under control of the pilot. The location of the counter-torque rotor, with respect to the wake of the main rotor, subjects the counter-torque rotor to cyclic loading of significant magnitude, presenting vibration and fatigue problems. The distance of the counter-torque rotor from the main rotor transmission, from which counter-torque rotor power is obtained, requires a complex power train. Since this power train is supported by a relatively flexible tail boom, the power train must contain numerous supporting bearings, a flexible or segmented shaft, and a 90° transmission. In order to provide ground clearance during certain flight operations and clearance for the main rotor when flapping rearward, the tail boom usually contains at least one significant bend. This bend requires an additional coupling or transmission in the power train and adds to the weight of the tail boom. In order to keep counter-torque rotor power requirements reasonable, the counter-torque rotor must have a

diameter approximately 1/5 that of the main rotor[*]. All of the above requirements must be met at a minimum weight penalty, since additional weight at a large distance from the helicopter's mass center increases the second mass moment about the pitching and yawing axis. Any increase in these moments increases control force requirements about these two axes. This critical weight requirement penalizes the use of rugged counter-torque rotor blades and transmissions, leading to reduced service life.

Despite this mechanical complexity, any failure of the counter-torque rotor system in flight constitutes a more serious emergency than a complete power failure[**]. Failure of either the counter-torque rotor or the engine requires an immediate autorotative landing, but without the benefit of yaw control in the case of counter-torque rotor failure. No dual or satisfactory fail-safe system has been used in conjunction with the counter-torque rotor system because of weight and compatability problems.

Operationally, the counter-torque rotor is subject to damage by collision when operating in confined areas, presents a hazard to personnel in the area, and is subject to damage by objects disturbed and recirculated by the main rotor. Further, no successful method of ice prevention has yet been developed for counter-torque rotors. Anti-icing systems applied to other parts of the helicopter are of little value

---

[*] This figure is based on an average of five operational, single-rotor helicopters.

[**] The U. S. Army Board for Aviation Accident Research reports that 10 per cent of all single-rotor helicopter accidents in the calender years 1959 and 1960 involved failure or collision of the counter-torque rotor.

without an adequate method of preventing ice accumulation on the counter-torque rotor; therefore, operation under icing conditions is not currently practicable. Cross-coupling of cyclic pitch, collective pitch, and power controls with variation in counter-torque power adds to the complexities of helicopter operation (4). Although this control coupling is of small consequence to experienced helicopter pilots, it adds substantially to the initial training requirement of helicopter pilots.

All of the disadvantages discussed above have been long recognized but no successful single rotor helicopter is currently in operation which does not use a counter-torque rotor. The success of shaft-driven, single rotor helicopters is confirmed by their wide operational application despite the problems involved.

From the above discussion, it is apparent that any system being considered to replace the present counter-torque rotor system might be advantageous if any of the following goals are realized:

1.   Reduce power penalty in the hover or power/drag penalty in cruise,

2.   Reduce mechanical complexity,

3.   Be fail-safe with reliability no less than that of main rotor,

4.   Reduce susceptibility to damage,

5.   Simplify operation through reduced control cross-coupling,

6.   Be compatible with an acceptable anti-icing system.

The relative freedom from mechanical complexity of jet flows appears to offer an area of possible improvement over counter-torque

rotor systems. The residual jet thrust inherent in turbine engines pro-
vides a potential source of jet thrust which will be investigated for
torque balancing.

CHAPTER II

A COUNTER-TORQUE JET SYSTEM

The use of jet thrust for a counter-torque force is basically no different in principle than the use of a counter-torque rotor. Both systems employ the reaction principle. The adaption of turbine engines, with a significant residual jet thrust, leads to the consideration of an effective use of this thrust and the use as a counter-torque jet follows.

Counter-torque jets were considered by Cierva (5) and Hiller (6) during and immediately after World War II. These investigators designed helicopters which were flown using a reactive jet instead of counter-torque rotors. However, in these designs the only source of jet flow was provided by fans powered by the helicopter's primary engine. Even though the fans also provided engine cooling and engine heat and exhaust were used to augment the thrust, the power requirements of the counter-torque jet were prohibitive for the reciprocating engines used at that time. The wide introduction of turbine engines, with their favorable power to weight ratios and residual thrust, justifies re-evaluation of the use of a counter-torque jet.

A qualitative consideration of the relative static thrust power requirements of the counter-torque rotor and counter-torque jet is informative. For a given thrust and induced power, the disk area of the ideal rotor is twice the area of a properly designed jet nozzle in the static thrust condition (7). Since the use of a jet nozzle with an exit area approaching half the disk area of current counter-torque rotors is

rather obviously impractical, it appears that the counter-torque rotor has an inherent power advantage.

When compared with an actual rotor, the jet nozzle becomes more competitive. Profile drag and tip losses account for a substantial part of total power required by any rotor. Variation of induced velocities along the rotor radius further reduces the effective area of the rotor. Real rotors designed for both forward flight and hovering require approximately 140 per cent more power than the ideal rotor, based on a typical Figure of Merit[*] of 0.70 (8). A well-designed nozzle, on the other hand, will have only negligible losses when operating at the design point (9).

Power required for a given thrust varies inversely as the square root of wake area for a rotor and the nozzle area for a jet as follows:

$$P = \sqrt{\frac{T^3}{4\rho\,(\pi r^2)}} \qquad (2.1)$$

T = thrust

ρ = mass density of air

r = radius of rotor wake or nozzle

It follows that for a given thrust and power and a Figure of Merit of 0.70, the rotor wake and jet nozzle have an area ratio of approximately 2 for the static thrust being considered.

---

[*]Figure of Merit, as used here, is the ratio of the power required by an ideal rotor or actuator disk to the power required by a real rotor in hovering.

$$P_{jet} = \frac{1}{.7} \sqrt{\frac{T^3}{4\rho(\pi r^2)}} = \sqrt{\frac{T^3}{4\rho(.49\pi r^2)}} \tag{2.2}$$

r = radius of rotor wake

Since the area of the rotor wake is one-half the rotor disk area, the required area of a well-designed nozzle will be approximately one-fourth that of a typical rotor for a given power and thrust. Similarly, it can be shown that a Figure of Merit of 0.60 will permit a nozzle area approximately 1/6 of the equivalent rotor area. The above analysis does not consider losses associated with the jet flow and any augmenting system. It will be shown later that substantial power losses will be present which must be considered when the nozzle area is selected.

An evaluation of the power required for counter-torque jets is instructive. The torque (τ) transmitted to the main rotor must be balanced by the jet thrust acting at the end of the tail boom. The equation for torque equilibrium is:

$$\tau = \frac{P_R}{\Omega} = \ell F_j = \ell m_j V_j = 2 \ell \frac{P_j}{V_j} \tag{2.3}$$

ℓ = length of tail boom

F_j = counter-torque force

which, with some manipulation, yields:

$$\frac{P_R}{P_j} = 2 \left(\frac{\ell}{R}\right) \frac{(\Omega R)}{V_j} \tag{2.4}$$

R = rotor radius

P_R , P_j , and P_E are power of the rotor, counter-torque jet, and engine exhaust. The tail boom length is ℓ and the angular velocity of the main rotor is Ω . If turbine efflux is used as the source of jet thrust, Equation (2.4) can be written:

$$\frac{P_R}{P_E} = 2(\frac{\ell}{R}) \frac{(\Omega R)}{V_E} \qquad (2.5)$$

Some of the important parameters can now be evaluated.

The ratio P_R/P_E is determined by the engine manufacturer who makes the ratio as large as practicable for turboshaft engines. This ratio is of the order of 25 for aircraft turbine engines currently in service (10). There appears to be no change in the ratio figure for the turboshaft or turboprop versions of the same engine. The ratio (ℓ/R) in present[*] helicopters using tail rotors is about 1.4. The term (ΩR) represents the rotor tip speed and has a practical limit of about 700 feet per second because of compressibility effects (11). Exhaust velocities are of the order of 350 feet per second. These representative figures yield:

$$\frac{P_R}{P_E} = 25 \text{ versus } 2(\frac{\ell}{R}) \frac{(\Omega R)}{V_E} = 5.5$$

From Equation (2.5) and the above values, it is obvious that the power in the exhaust must be augmented, the (ℓ/R) ratio increased, rotor tip speed, (ΩR) increased, efflux velocity (V_E) decreased, or a combination of these changes if we are to balance the torque using this residual

---

[*]An average based on current, operational, single-rotor helicopters.

thrust.

Another form of the equilibrium equation is instructive.

$$\frac{P_{HP}}{F_E} = \left(\frac{\ell}{R}\right) \frac{(\Omega R)}{550} \tag{2.6}$$

F_E = engine exhaust thrust

P_{HP}= horsepower delivered to rotor

A typical range of horsepower to pounds thrust for modern turboshaft engines is 5 to 9 (10). A representative value of the right side of Equation (2.6) is of the order of two. Although increasing rotor tip speeds above present values is a subject of wide research, it does not appear that any significant increase is currently possible. The ratio (ℓ/R) might be increased somewhat as the counter-torque jet will offer reduced vibration and mechanical complexity over the counter-torque rotor system and will be less subject to damage when operating in confined areas. Increases in (ℓ/R) of the magnitude necessary to balance Equation (2.6) are clearly impractical. The magnitude of thrust augmentation required is now evident. Even though the residual thrust of the turboshaft engines is significant, it is not sufficient to provide counter-torque thrust unless substantially augmented. An evaluation of the feasibility of providing the necessary augmentation will be an important part of this study.

Although this study is applicable to all single-rotor, turbine-powered helicopters, comparison of various systems will be made with a specific helicopter in order to make the evaluation meaningful. The U. S. Army HU-1B, shown in Fig. 2, has been selected for this purpose.

The HU-1B, manufactured by Bell Helicopter Corporation, is in wide serv-
ice with the U. S. Army, and will present a valid basis for evaluation
of a jet counter-torque system. This helicopter is powered by the Ly-
coming T53-L-5 gas turbine engine.

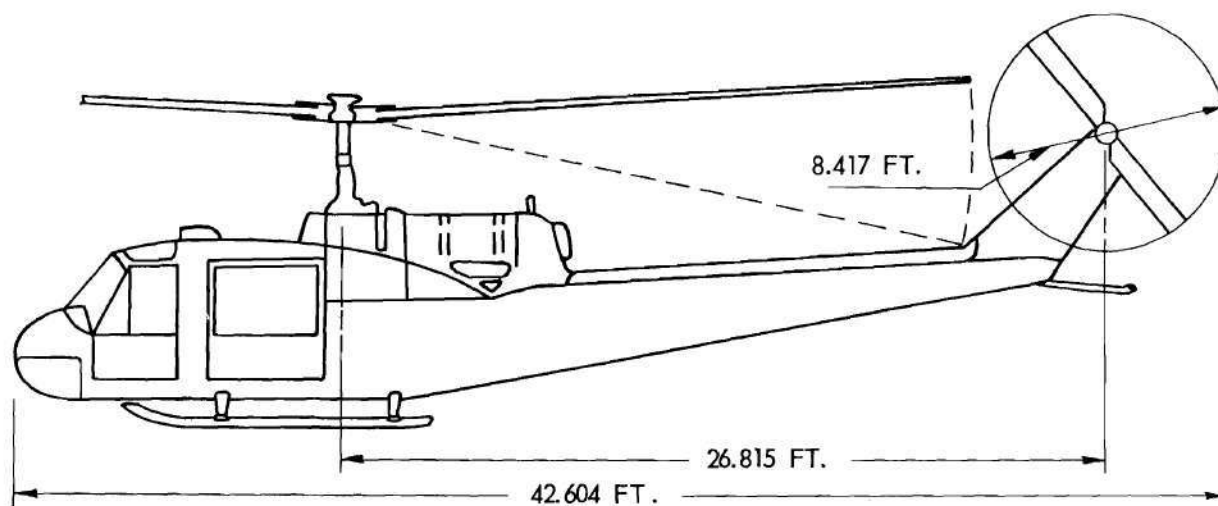8.417 FT.

26.815 FT.

42.604 FT.

Fig. 2 U. S. Army HU-1B Helicopter

The counter-torque requirements of this helicopter will be pre-
sented at this point for the purpose of later comparison. For standard
sea level conditions and a maximum gross weight of 8250 pounds, flight

test data shows that the HU-1B helicopter requires 926 horsepower to hover out of ground effect (12). Assuming that 90 per cent of this power is delivered to the rotor, 843 horsepower transmitted to the rotor shaft at an angular velocity of 314 revolutions per minute yield a torque as follows:

$$\tau = \frac{P_R}{\Omega} = \frac{843}{314} \left(\frac{550 \times 60}{2\pi} \right) = 14,100 \text{ ft. lb.}$$

The induced power required by an ideal rotor under these conditions is 506 horsepower. This indicates a hover efficiency or Figure of Merit equal to $506/843 = 0.6$. Since the counter-torque rotor is designed for the same overall flight conditions as the main rotor, the same Figure of Merit can be used to approximate the counter-torque rotor power requirements in the hover.

The counter-torque rotor has a moment arm of 26.815 feet and a radius of 4.208 feet. The counter-torque thrust required is $14,100/26.815 = 525$ pounds. Using the Figure of Merit estimated above, the power required by the counter-torque rotor can be estimated.

$$P_{CT} = \frac{T_{CT}}{.6 R_{CT}} \sqrt{\frac{T_{CT}}{2\rho\pi}} = \frac{525}{.6(4.208)} \sqrt{\frac{525}{2(.002378)}} = 70.8 \text{ HP}$$

This indicates that the power required by the tail rotor in this flight condition is approximately 7.6 per cent of total engine power. It will be shown later under the discussion of yaw control that under certain flight conditions, a greater counter-torque thrust than given above is required. It might be noted that at a sea level power setting of 926

horsepower, the engine being considered delivers 110 pounds of residual thrust at a kinetic energy equivalent to 32.1 horsepower. This residual thrust is only 21 per cent of that necessary to balance torque, but the kinetic energy of this exhaust is equivalent to 45.3 per cent of the power used by the counter-torque rotor.

CHAPTER III

TORQUE BALANCING WITH A VERTICAL TAIL

Since the torque transmitted to the fuselage of helicopters is
not unlike the yawing moment found in airplanes under certain flight
conditions, the use of a vertical tail is worthy of consideration for
forward flight conditions. The design criteria of vertical stabilizer-
rudder combinations has been well developed in the literature and
should present no major problems for use on helicopters. Adequate ver-
tical tails are not used on single rotor helicopters because they pre-
sent compatability problems when used with counter-torque rotor systems.
Compatability with a counter-torque jet system appears to present no
real problems.

The vertical tail of a multi-engine airplane is designed to over-
come the fuselage instability and to provide stability during the assym-
metric loading in an engine inoperative condition. These instabilities
can produce either right or left yawing moments and for this reason ver-
tical tails on airplanes are nominally symmetric. For the helicopter,
the unstable yawing moment of the fuselage will normally be small com-
pared to the reaction torque of the lifting rotor. Since this torque is
always in the same direction, a vertical tail for the helicopter should
be designed assymmetrically using a cambered airfoil section. A tail de-
signed to provide torque balancing at cruise conditions will provide sta-
bility at flight speeds considerably below the cruise. The magnitude of

the stabilizing force provided by the rudder varies with the square of
the flight velocity, hence is less at lower forward speeds. Since re-
quired power, hence torque, also decreases with velocity to a certain
point, stability is maintained over a wide range of powered flight. The
vertical tail will be self-stabilizing at flight speeds near the design
point. Exact matching to obtain zero yaw may be accomplished by appli-
cation of the rudder by the pilot. Fig. 3 shows a plot of torque trans-
mitted to the fuselage at various level flight speeds against the
counter-torque force provided by a vertical tail adapted to the HU-1B
helicopter. The performance of a typical tail is shown in Appendix I.
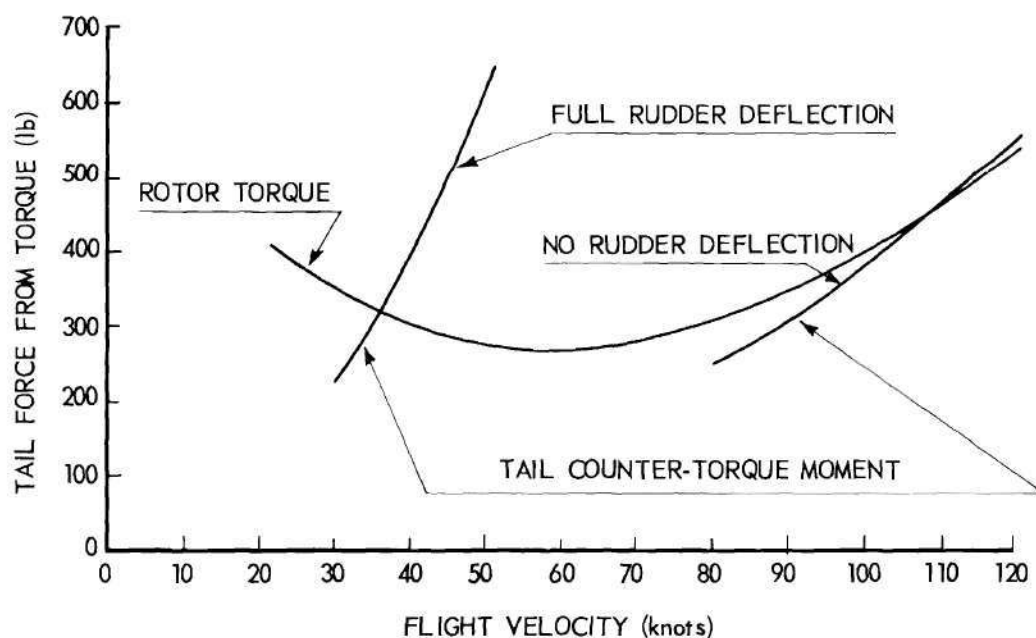

Fig. 3 Torque and Counter-Torque Forces
Versus Flight Velocity

It should be noted that a vertical tail system is extremely reliable in comparison with counter-torque rotor systems. Further, it is compatible with any of several successful anti-icing systems and is relatively immune to damage in confined area operations. The minimum speed for powered flight for tail moment balance will be reduced substantially from that shown in Fig. 3 during normal descents with reduced power. Adequate control should be provided throughout the normal autorotation range since there is no torque to overcome and some forward speed is normally maintained in autorotations. Autorotative landings in a crosswind will have a yawing angle with respect to the ground under some flight conditions and will require increased pilot proficiency. Since autorotative landings in crosswinds are avoided with all helicopters, this should be a small consideration. Further, if no power is being taken from the main rotor to power the counter-torque rotor, the autorotative sink rate will be somewhat reduced, allowing the pilot a greater margin for error.

The vertical tail still constitutes a drag force of consequence. For the example in Appendix I, the vertical tail has a drag force of 49 pounds at the cruise condition. This drag is equivalent to an expenditure of 16.4 horsepower. On the other hand, the counter-torque rotor at zero disk incidence has a negligible longitudinal force, but has a continuous expenditure of 50 horsepower[*]. The difference in horsepower is available for increased range or payload or can be used for higher cruise speeds.

---

[*] Estimate based on 70 per cent of static power requirement.

From the above considerations it is apparent that the vertical tail has significant advantages over the tail rotor system in the cruise condition. Since the major portion of most flight mission profiles are at cruise conditions, additional penalties, such as increased fuel consumption at the hover, might be warranted.

CHAPTER IV

JET THRUST AUGMENTATION

In the hover or at very low flight speeds, the vertical tail will
be ineffective and counter-torque thrust must be provided to balance
rotor torque. If the residual thrust from the turbine engine is to be
used for this purpose, it was shown in Chapter I that substantial aug-
mentation must be provided.

Various means of jet thrust augmentation have been investigated
and the potential of the conventional augmentation methods are well doc-
umented (13). The primary means used for thrust augmentation are:

(1)   Afterburning,

(2)   Ejector augmentation,

(3)   Rotary or fan augmentation.

In addition to the standard means of thrust augmentation listed
above, two other methods are available for the case being considered.
Unlike the case of pure jet engines, the available kinetic energy in the
gas flow of turbo-shaft engines is divided between shaft horsepower and
jet thrust. Turbine engines designed for helicopters allocate the maxi-
mum amount of kinetic energy to shaft power consistent with engine
weight and complexity. For any given engine, this ratio of shaft to
thrust power can be varied within reasonable limits by variation of the
exhaust nozzle area without materially changing the overall energy de-
veloped by the engine. Constriction of the nozzle will, within reason-
able limits, increase exhaust velocities without materially changing the

mass flow. This will increase the thrust produced by the exhaust but only at the expense of energy delivered to the power turbine (14). Increasing this thrust by increasing the velocity of the exhaust in this manner is mechanically simple but an inefficient utilization of available energy. Calculations in Appendix II with two typical turbo-shaft helicopter engines show that gross thrust can be doubled at a shaft horsepower penalty of approximately ten per cent. It was shown in Chapter I that thrust must be increased by a factor substantially greater than two. Since the power required by helicopter rotors is relatively high in the hover, any augmentation method which reduces shaft power significantly more than ten per cent will certainly be unacceptable.

A second method of augmentation not normally used in turbo-jet engines is the addition of energy to the primary exhaust flow with a fan or compressor. A fan placed in the exhaust duct can increase thrust by increasing momentum forces or pressure forces or both. For the application being considered, the exhaust duct must be designed so as not to create an unfavorable back pressure which will degrade engine performance. The power required to accelerate exhaust efflux sufficiently to provide counter-torque forces can be estimated by computing induced power requirements. To provide 525 pounds of thrust at the engine mass flow of .339 slugs per second requires a velocity of 1550 feet per second. Exhaust velocity from the engine is 250 feet per second. This is equivalent to an induced power of more than 700 horsepower and is clearly an unsatisfactory approach.

Afterburning is a standard method of jet thrust augmentation on high performance military airplanes and the theory of afterburning

thrust augmentation is simple and well established. For low velocities and negligible changes in the specific heat ratios, the augmentation ratio with afterburning increases as the square root of the absolute temperature ratio. The limiting factor is the maximum temperature which the exhaust ducting and nozzle can withstand. Since afterburning takes place after all rotating parts have been passed, no parts are subjected to high dynamic loading at afterburning exit temperatures. For this reason, afterburner temperatures over 4000°F may be acceptable. With normal jet exhaust temperatures, augmentation ratios of 1.5 are theoretically practicable; in practice, augmentation ratios of 1.4 are more normal (15).

Turboshaft engines normally have lower exhaust temperatures than turbojet engines so that higher augmentation ratios by afterburning are theoretically possible. Exhaust temperatures for the engine being considered are about 1500°R. With a temperature increase to 5000°R, the augmentation ratio is 1.8.

Using JP-4 as fuel, the afterburner fuel requirement for this augmentation ratio is nearly 1800 pounds per hour. This is a 300 per cent increase in fuel requirements over the engine fuel consumption of 600 pounds per hour at the power settings being considered. An increased fuel consumption of this magnitude clearly could not be tolerated for any significant part of the total flight time. Except for certain specialized missions, the time spent in the hover is normally a small part of the total mission profile. An increased fuel consumption of the magnitude shown above would be unacceptable even when it is considered that no engine shaft power is being diverted from the lifting rotor for

counter-torque purposes. The pressure ratio of expansion is so low that relatively little of the heat added by burning is removed during the expansion process and most of the fuel added serves merely to heat the atmosphere (16). This extremely hot flow will present a decided operational hazard and will probably prohibit final exhaust temperatures as high as those found on jet airplanes operating in a different operational environment.

Unlike airplane afterburners which can generally suffice with a two-position, on-off nozzle, the application being considered will require a fully variable area nozzle in order to balance torque and provide yaw control. The requirement for a variable area nozzle may also limit final exhaust temperatures. All afterburner installations require a great deal of empirical development to insure a successfully functioning system. Fuel metering, fuel nozzles, flame holders, and ignition devices are the subject of wide empirical investigation for afterburners and should present no special problems in the case being considered. The relatively low turbine exhaust temperatures will require a positive ignition device since self-ignition will not be possible. On the other hand, the relatively low exhaust velocities (about $M = .135$) will assist in the maintenance of a stable combustion process.

All afterburners generate a high noise level. The magnitude of this noise is difficult to predict because it is dependent on flame stability, resonance, and other factors associated with a particular afterburner. Certainly, the noise level with a high afterburning augmentation ratio will be much higher than with a counter-torque rotor system and almost assuredly greater than the noise of the turbine engine and rotor combined. Clearly, the penalties of afterburner augmentation will pro-

hibit full use in the application being considered.

The most successful method of thrust augmentation for jet airplanes is the use of fan augmenters. Both the ejector and fan augmenters work on the principle of using the available kinetic energy to increase the mass flow, thereby increasing effective thrust. Since this is accomplished without a proportionate increase in fuel consumption, the additional thrust results in better specific fuel consumption as well as increased performance.

In the ejector augmenter, the engine exhaust is discharged into another duct, as shown in Fig. 4. This relatively high velocity flow entrains additional air from the atmosphere by turbulent mixing on the flow boundaries and the resulting decrease of pressure causes further flow of air from the atmosphere.
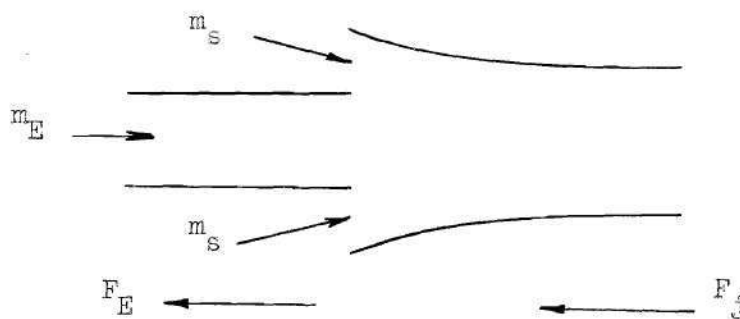
m_E = mass flow of exhaust F_E = thrust of exhaust

m_S = mass flow of secondary air F_j = augmented thrust

Fig. 4   Ejector Augmenter

As the entrained atmospheric air is accelerated, the exhaust flow is decelerated and, if mixing is complete, the entrained air and exhaust discharge at a common velocity. The total effect of the system is to discharge a greater mass of air at a lower velocity than that of the engine exhaust alone. Whether the product of mass flow and velocity is greater for the augmenter system or the engine exhaust alone depends on the design of the system. It has been shown (17) that the ratio of augmented thrust to primary thrust for augmenters with cylindrical mixing ducts and no diffuser operating in the static thrust condition is given by:

$$\frac{F_j}{F_E} = (1 + \frac{m_s}{m_E})(1 + \frac{m_s}{m_E} \frac{\rho_E}{\rho_s})(\frac{A_E}{A_j})\ 1.2 \tag{4.1}$$

ρ_E, ρ_s = mass density of exhaust and secondary flow

A_E, A_j = cross-sectional area of exhaust and nozzle ducts

The constant 1.2 is a semi-empirical factor which accounts for non-uniform velocities at the final exit plane. These investigators confirmed their ejector theory by experiment and found that the greatest thrust augmentation was obtained with a small pressure drop through the augmenter system. An examination of the expression above clearly shows the benefit of increased mass flow ratios in obtaining significant thrust augmentation. The maximum mass flow ratios obtainable by ejector augmenters are not clearly predicted from the theory but static thrust augmentation ratios of 1.5 are not uncommon (18).

The length of the duct, where the exhaust gases and secondary air mix, also effects the augmentation ratio and the duct should be long

enough for complete mixing for maximum augmentation. This optimum length varies from 4-8 times the exhaust diameter. Lengths substantially in excess of this optimum length sustain additional losses due to friction.

The fan or rotary augmenter is very similar to the ejector augmenter in principle. The main difference is that a low pressure fan is used to pump the secondary air so that a substantially larger ratio of mass flows is realized. This fan is shaft or gas driven using the available kinetic energy of the system. The mass flow ratio can be increased to any reasonable figure depending largely on what power is allocated to drive the fan. Since the fan produces a relatively low pressure ratio, operates in an axial flow, and is ducted, a simple fan will produce high efficiencies compared to a free rotor.

The thrust augmentation ratio for this method of augmentation can be given by the simple expression:

$$\frac{F_j}{F_E} = \sqrt{\eta \, \frac{m_j}{m_E}} \qquad (4.2)$$

The factor η is a ratio of the kinetic energy of the final flow after augmentation to the kinetic energy in the initial exhaust flow. This factor is less than unity, because of viscous effects, if energy from the exhaust is used to drive the augmenting fan. If additional energy is shafted to power the augmenting fan, η can be made larger than unity by almost any reasonable amount. A turbo-prop installation is an extreme example of this augmentation if it is considered that the propeller augments the thrust of the engine exhaust. Whether η is larger or smaller than unity, the mass flow ratio must be larger than

unity to realize a thrust augmentation in any reasonable configuration.

Clearly, the fan augmenter is the only method of those discussed which has the potential of adequate augmentation. Since the final exhaust pressures must be approximately atmospheric, the mass flow ratio will determine the required nozzle size. Figure 5 is a plot of theoretical power requirements for an exhaust nozzle to produce the required counter-torque thrust (525 lbs.) at various mass flow ratios. No losses are included in this chart and areas are determined assuming discharge at atmospheric pressure.
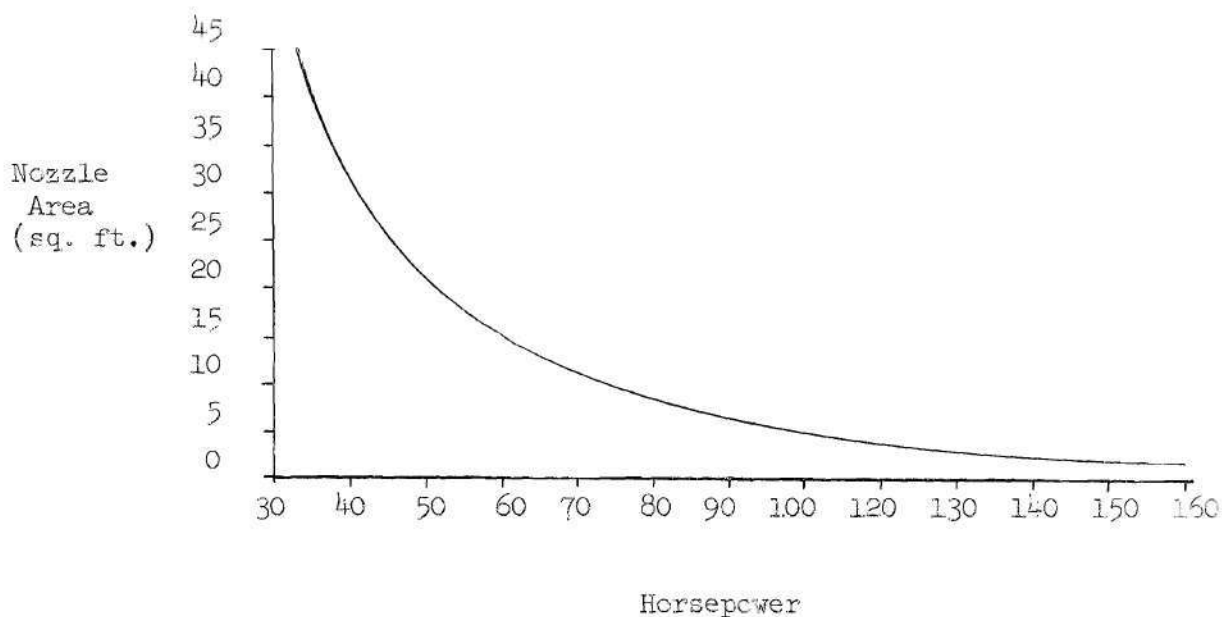

Fig. 5 Horsepower Variation with Nozzle Area

It is apparent that augmentation by increasing mass flow can reduce the counter-torque power requirements by any desired amount consistent with the airframe geometry. Since requirements are reduced as the exhaust nozzle area is increased, some maximum nozzle area consistent with the helicopter configuration can be selected and that will determine the power required. Certain known losses can be added to the power indicated in Fig. 5. The fan which provides the augmentation will be operating at a low pressure ratio, in an axial flow, and in a duct so that fan losses should not exceed 10 per cent (19). The turning losses are substantial and will equal 15 per cent of the kinetic energy making the turn for a reasonable elbow geometry (20). The propulsive efficiency of the nozzle will be high and losses at this point should amount to less than 5 per cent of the energy at the nozzle at the design operating condition (21). Losses in the duct to account for non-isentropic flow will be small since both temperature and velocity of the augmented flow are low. Thermal and frictional duct losses are examined in Appendix III. Exhaust temperatures will be low at high mass flow ratios yielding reduced heat losses in the duct. At the resulting low temperatures, no hazards should result from the exhaust flow[*].

Wide latitude in design is afforded by the fan augmentation. For simplicity the fan should be located as close to its power source as possible. The duct design must also insure that the engine is not exhausting into a region of unfavorable pressure to prevent needless engine power losses. Although the design of duct elbows necessary to turn

---

[*]At a typical augmentation ratio shown later in Fig. 6, the duct temperature is within 120° of ambient temperatures.

the flow is well established, some empirical design will be required to obtain high efficiency of the elbow in this application.

A possible configuration for an HU-1B helicopter modified to use a vertical tail and counter-torque is shown in Fig. 6. This configuration retains the same blade and ground clearances as the basic HU-1B helicopter.
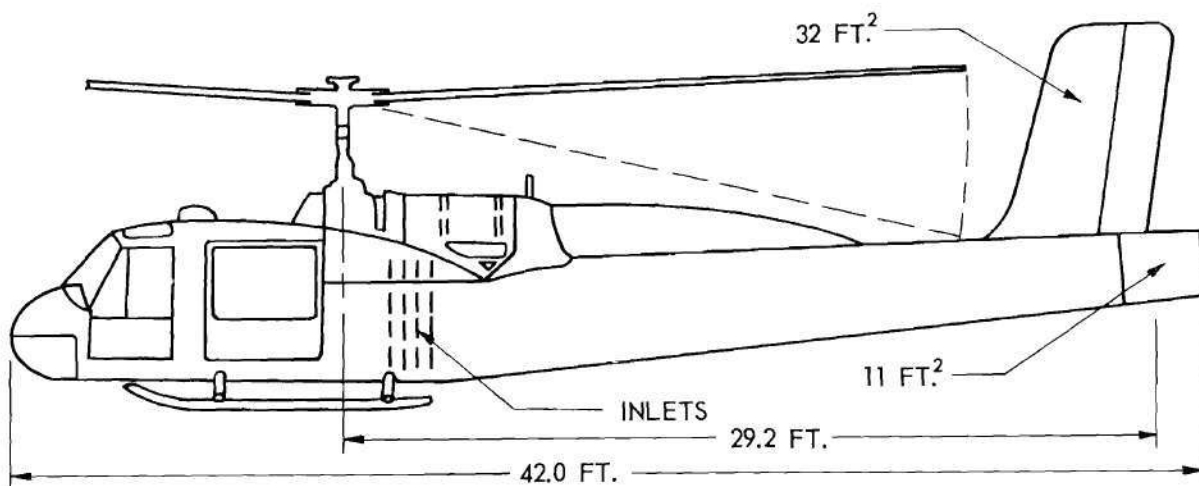
Fig. 6 Modified HU-1B Helicopter

The configuration shown will take 128 horsepower at the hover based on Fig. 5 and allowing 25 per cent for losses. Since 32.1 of the required horsepower is available from the exhaust, this will require

some 25 shaft horsepower more than the original counter-torque rotor. Fig. 5 shows the reduction in counter-torque power which can be obtained using a larger nozzle area. This selection would appear to be at the option of the designer. A larger duct-nozzle choice will require more weight than a smaller selection.

For any reasonable configuration, the weight saving over the counter-torque rotor system should be favorable. The two transmissions in the counter-torque rotor drive system weigh 43.5 pounds, and the drive shafting weighs 30.7 pounds (22). The weight of a vertical tail and rudder will depend on specific design but should not exceed the weight of the transmissions and shafting eliminated. The nozzle should not weigh more than the additional weight of the upturned tail boom on the original helicopter. Modification to the tail boom will require monocoque construction to facilitate flow. This construction will result in a tail boom slightly more expensive to fabricate but should not require any appreciable weight increase. At the mass flow ratios necessary to keep power requirements reasonable, the flow temperatures will be low enough to retain aluminum construction. The counter-torque rotor weighs 33 pounds (22) which is substantially more than the augmenter fan should weigh. Both fabrication and operating costs of the counter-torque jet system should be less than the comparable counter-torque rotor due to reduced mechanical complexity and maintenance.

The weight of the fan drive system will depend on the specific system chosen. Since counter-torque balancing can be accomplished more efficiently by the vertical tail at normal cruise speeds, the augmentation fan power is not required in cruising flight. For this reason, the

drive system selected should be able to transmit variable power to the augmenting fan. If shafted directly from the transmission, pitch change on the fan can change the power used from the maximum required to a minimum required dependent on the minimum drag of the fan blades. This minimum will be a small percentage of the maximum power for which the fan is designed. Any power transmission system which can use the nearly constant angular velocity at the transmission and transmit variable angular velocity to a constant pitch fan will suffice. Both electrical and hydraulic systems of standard design are available for this purpose at reasonable efficiencies. If the helicopter has an operational requirement for an auxilliary power unit for starting or ground operation of electrical equipment, light-weight turbines are available which could fulfill this requirement as well as providing the counter-torque power in flight. The use of an auxilliary turbine in this manner would enable the full use of the primary engine by the lifting rotor during all flight conditions.

It should be noted that the increase in the counter-torque force necessary at low airspeeds need not be an additional pilot control. The force can be automatically controlled as a variable of forward speed based on pitot-static differential pressures. Activation by the pilot's rudder pedals is possible but will give wide variation from the pedal neutral point at the cruise and at the hover and will probably not be acceptable.

It has been shown that the vertical tail provides a counter-torque force in cruise more efficiently than either a counter-torque rotor or counter-torque jet. For this reason, some arrangement must be made to

uncouple this power during the cruise condition. If this thrust has a useful purpose during cruise, the requirement to uncouple the power would not be necessary. System reliability would thus be increased. Although not a particular part of this study which is concerned with counter-torque jets, the counter-torque thrust has been augmented to a magnitude such that it can provide a useful propulsive force.

In the normal configuration for the HU-1B and other helicopters, the fuselage pitches downward with increasing forward speed. This pitch downward causes the fuselage to operate at an unfavorable angle of attack which has a larger parasite drag and a very significant download. For the HU-1B, the parasite drag area increases from about 17 to over 20 square feet and the download gives an effective increase in gross weight of nearly 800 pounds (23). Efforts to keep the fuselage level by the use of a larger horizontal tail have resulted in unacceptable stability prob-lems in the hovering and autorotation flight condition. Jet thrust being directed rearward from the end of the tail boom will greatly reduce or even eliminate any downward pitch. Further, the jet thrust can be vec-tored to trim the helicopter fuselage to any desired attitude without imposing any additional stability problems. Any downward component of this thrust would be insignificant when compared with the download being eliminated.

The result of being able to maintain the fuselage at its most favorable attitude would be to reduce the induced and parasite power re-quirements of the main rotor. Properly designed duct inlets can reduce the power requirements of the fan in cruising flight. Proper placement of the duct inlets relative to the area of flow separation behind the

fuselage might result in a reduction of fuselage drag. The reduction in total required power by the main rotor will result in increased range for any helicopter or increased forward speed on helicopters whose forward speed is retreating-blade, tip-stall, or power-limited. Longitudinal trim control by vectored thrust could be adjusted by the pilot for each center of gravity position and flight speed as with conventional trim controls or could be adjusted for an average center of gravity location. If the jet thrust equals the parasite drag of the fuselage, both rotor and fuselage will operate essentially at zero angle of attack.

The reduction in power required can be estimated by considering the parasite power requirement. At a cruising speed of 110 knots, the fuselage has a parasite drag area of 20 square feet. This is equivalent to a drag force of 820 pounds at standard sea level conditions. Trimming the fuselage to a more favorable angle of attack by thrust vectoring can reduce the drag force to 697 pounds and eliminate an 800 pound download.

If the fan is designed to provide the maximum static augmentation required in hovering, it is shown in Chapter V that 678 pounds of thrust will be required. With a nozzle area of 11 square feet, the exhaust flow will require 100 horsepower (including 33 horsepower delivered from the engine exhaust) and will have an exit velocity of 162 feet per second. If the fan inlet is designed so that no dynamic pressure from forward flight is recovered, no net propulsive thrust will remain at 110 knots flight speed. Use of a 130 pound vertical component for trim will leave a propulsive drag of 110 pounds which is less than the drag eliminated by trim. An overall power reduction may be realized from elimination of fuselage download and reduction of parasite drag by boundary layer suc-

tion at the fan inlet. Assuming no favorable boundary layer effect, the saving in rotor power is mainly induced power as discussed below. If parasite drag can be reduced 30 per cent by boundary layer suction at the fan inlet, the parasite power saved is 115 horsepower. If the fan inlet is designed to recover 80 per cent of the dynamic pressure from forward flight and a 30 per cent reduction in parasite drag is realized from boundary layer acceleration, the augmenting fan requirements increase to approximately 230 horsepower and the saving in parasite power is 277 horsepower. In any case, the power requirement of the augmenting fan must be compared to the power used by the counter-torque rotor in the same flight condition. Accurate prediction of drag reduction by boundary layer suction will require experimentation with a specific design. In addition to the parasite power reduction estimated above, flight tests of the HU-1B indicate that the rotor power which can be reduced by elimination of download is 62 horsepower at 110 knots. Additionally, reduction of rotor disk incidence will result in less blade profile drag power when operating at high forward speeds. It appears that a potential saving of 10-15 per cent of total power might be gained at 110 knots by drag reduction using boundary layer control and fan propulsion and by use of the thrust for trimming.

CHAPTER V

YAW CONTROL

In almost all standard helicopter configurations, yaw control is
provided by the same mechanism that provides the counter-torque force.
Therefore, any new counter-torque system being considered must provide
yaw control or other provisions must be considered. The vertical tail
discussed in Chapter III will provide sufficient stability and yaw con-
trol at cruise speeds with only nominal rudder application by the pilot.
If jet thrust is to provide the counter-torque force in the hover, the
moment of this force must exactly balance the shaft torque for equili-
brium and be capable of modulation by the pilot to provide a responsive
yaw control.

To be fully effective, the yaw control must have a response time
less than the pilot's reaction time. Otherwise, the pilot will make a
desired control application and, sensing no response, will increase the
control application leading to overcontrol. Current single rotor heli-
copters provide nearly instant response in yaw by a mechanical linkage
from the rudder pedal to change the pitch on the counter-torque rotor.
If a similar change in pitch or angular velocity were made to a fan, as
considered in Chapter IV, the control force response will be slow because
of the effective mass of the fan flow. Cierva, in his investigations,
confirmed this sluggish response when thrust modulation is applied at
the intake end of the duct (24). This control can be made as responsive
as the counter-torque rotor system by placing the thrust control at the

nozzle. If the counter-torque force is to be modulated at the nozzle end of the duct, the augmenting fan will use power continuously at a rate equal to the maximum force ever required. Rudder pedal deflection by the pilot will cause a responsive change in thrust if the pedals turn the counter-torque jet.

Using a turning elbow (20), as shown in Fig. 7, good efficiency can be obtained for counter-torque balancing and the nozzle walls can be retracted as shown to turn the flow as required. Other turning vane configurations may provide equal or better efficiencies and control.
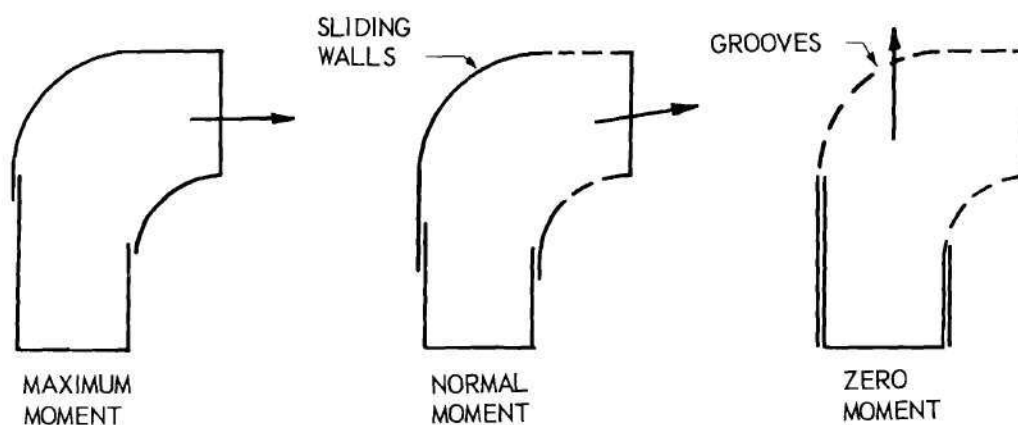

Fig. 7 Controllable Thrust Turning Elbow

The maximum yawing force will be required when making hovering turns against the rotor torque at near the hover ceiling or when hovering in the maximum acceptable crosswinds from the right. Military specifications require that helicopters in the weight class of the HU-1B be

capable of hovering turns of 3° in the first second after full control deflection in the most critical azimuth relative to the wind (25). In a 30 knot crosswind, the force against a 32 square foot vertical tail is approximately 125 pounds. The moment of inertia of the HU-1B about its yawing axis is 7,211 slug-square feet. At the end of a 26.8 foot moment arm, a force of only 28 pounds additional is required to obtain the required turning accelerations. Considering the 525 needed for torque balancing, a maximum total of 678 pounds of thrust may be required.

The additional 153 pound thrust requirement shown above constitutes a power loss for most hovering conditions. Since the augmentation fan must continuously provide the major part of the counter-torque force required under the most critical conditions, power is wasted during other than critical conditions. In a vertical take-off, 960 horsepower is available from the engine. This maximum power will require a counter-torque force of 537 pounds, which is not greater than the requirement stated above.

Since this power is being wasted for only a small part of the overall average mission profile (hovering), the penalty is small. A smaller vertical tail and a larger mass flow ratio (nozzle area) can reduce this excess power requirement, if necessary. Since any helicopter must have the reserve power to turn in the critical conditions stated above, the continuous generation of this power does not degrade performance except for the slightly higher fuel consumption in the hover. As was shown in the last chapter, this power can be put to efficient use at cruise conditions.

Since continuous thrust from the augmenting fan flow and engine exhaust are available, the nozzle should be designed for maximum thrust

at full left pedal deflection. At all other pedal settings, thrust can
be reduced as required by turning of the nozzle. High nozzle efficiency
will only be required for the maximum thrust so that nozzle losses due to
operation off the design condition are of no consequence. The greatest
losses will be while hovering at minimum weight and minimum density alti-
tudes. The maximum excess power exists under these flight conditions so
that the dissipation of more power is of little consequence.

The vertical tail will provide an unstable yawing moment in rear-
ward flight or tail-into-wind hovering. This instability is found in all
helicopters but rudder deflections will increase the unstable condition.
This instability will probably not be greater than on other helicopters
and, in any case, is subject to reduction if necessary.

Since yaw control is desirable at zero forward speed during auto-
rotative landings, the augmenting fan must be geared to the rotor during
the autorotation. If the nozzle has been turned rearward during cruise,
some fail-safe method of operation, such as spring loading, must assure
that the nozzle is in the counter-torque configuration for the landing.
Vertical autorotations are seldom performed except for the actual touch-
down. If complete yaw control is required during vertical autorotative
flight, complications will arise in providing a right turning moment.
Since this requirement does not appear to be a valid one to the author,
it will not be discussed further. During power-off flight, the inertia
of the augmenting fan will tend to drive the main rotor during rotor de-
celeration causing a right turning moment. The right turning moment can
be balanced by the jet or used to make a rightturn at the pilot's option.

The HU-1B and most other single rotor helicopters have an upward bend in the tail boom which permits the use of a relatively large counter-torque rotor and still assures ground clearance. This raised counter-torque rotor has an added advantage in that the center of thrust of the counter-torque rotor is raised nearly to the level of the plane of rotation of the main rotor which partially eliminates a control cross-coupling. The configuration shown in Fig. 8 has the same ground clearance as the HU-1B, but the thrust line is lower than the plane of rotation so that a control cross-coupling develops when the counter-torque force is increased or decreased. Fig. 8 shows a front view of the configuration shown in Fig. 6.
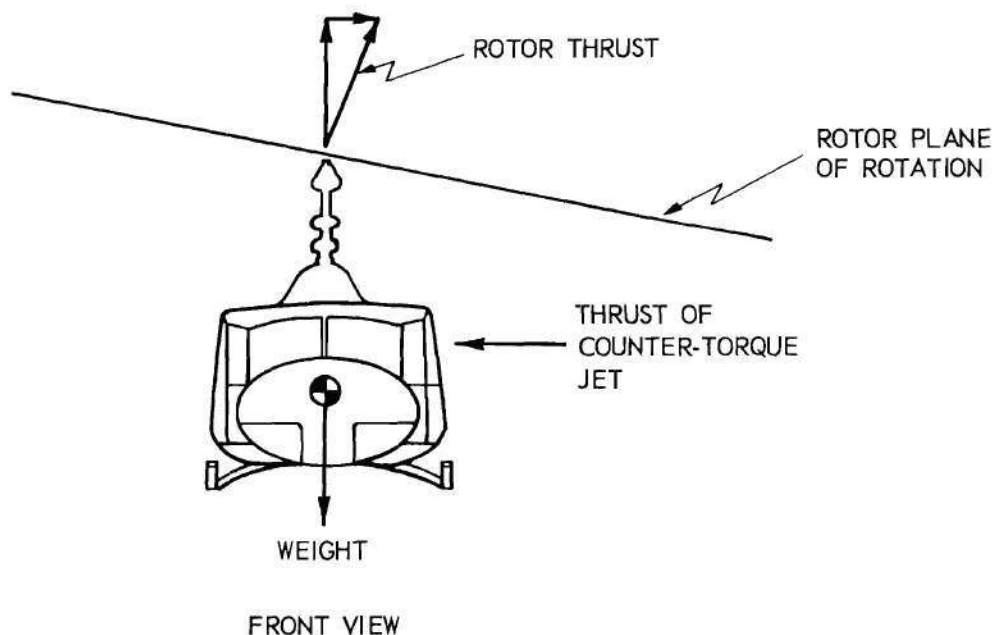

FRONT VIEW

Fig. 8   Resultant Control Cross-Coupling

The counter-torque force must be balanced by a sideward tilt of the rotor thrust. The sideward component of the rotor thrust equals the counter-torque thrust in magnitude but acts at the plane of the rotor and therefore forms a rolling couple which can only be balanced by another couple. This balancing couple is formed by a rolling displacement which offsets the helicopter center of gravity from the vertical thrust axis. A similar unbalance about the pitching axis will cause a pitching displacement as well. These pitching and rolling displacements are probably not large enough to justify bending the tail boom up into the rotor plane of rotation. They will occur only when the magnitude of the counter-torque force is changed. Since these changes are small compared with the basic force, the pitching and rolling movements will also be small. A cross-coupling of the controls remains in either case. A certain amount of fuselage motion will never be entirely absent during significant variations of counter-torque thrust. Control of the counter-torque jet by turning it rearward will cause another small control cross-coupling which can probably be ignored. The additional cross-couplings shown above are not inherent in the counter-torque jet system itself but only in the simple configuration proposed in Fig. 6.

CHAPTER VI

CONCLUSIONS

The standard counter-torque rotor used on single rotor, shaft-powered helicopters presents mechanical, aerodynamic, and operational problems which degrade the reliability of the helicopter.

The use of a vertical tail and rudder combination to provide counter-torque balancing and yaw control at forward flight speeds was investigated. The investigation indicated that counter-torque balancing and control can be provided by the vertical tail at flight speeds above 35 knots for the model selected at a substantial saving in counter-torque power expended. It was shown that the mechanical complexity of the vertical tail and rudder is less than the comparable counter-torque rotor system.

The requirements of a counter-torque jet system to provide the counter-torque force and yaw control during the hover were investigated. The magnitude of the residual jet thrust from helicopter turbine engines was shown to be less than needed for counter-torque balancing. Ejector, nozzle constriction, and after-burning jet augmentation were considered to increase the residual engine thrust and found to be unsatisfactory methods. Fan augmentation was considered and it appeared that the power requirements of augmenting the residual thrust by the use of a fan were comparable to the power requirements of the counter-torque rotor. The relative mechanical simplicity of the fan augmenter over the counter-

torque rotor system indicated a gain in reliability for the counter-torque jet system.

Control response of the counter-torque jet was considered. Response time appeared to be satisfactory if controlled at the nozzle end.

The reduction of fuselage parasite drag by boundary layer control using the inlet flow of the augmenting fan was considered.

It was shown that fuselage attitude trim by thrust vectoring can reduce both parasite drag and download. Rearward turning of the jet during cruise will provide a propulsive force reducing rotor tilt by further reducing the amount of parasite drag which the rotor must overcome. A possible total power reduction of 10-15 per cent at high cruise speeds was indicated. Higher cruise speeds and greater range would result.

APPENDIX I

VERTICAL TAIL FORCES

C_L = lift coefficient of vertical tail

C_D = drag coefficient of vertical tail

C_{d_o} = profile drag coefficient

S = surface area of vertical tail, square feet

q = dynamic pressure at vertical tail, pounds per square foot

D = total drag of vertical tail, pounds

L = total lift of vertical tail, pounds

A = aspect ratio of vertical tail

α^o = angle of attack of vertical tail, degrees

A representative vertical tail-rudder combination was selected with the following typical characteristics:

$S = 32$ ft^2 $\qquad\qquad$ $C_{d_o} = .008 + .008C_L^2$

$A = 1.5$ $\qquad\qquad$ $\alpha^o = -2^o + 23.3C_L$

Cruising speed = 110 knots = 185.7 feet per second

$q = 41$ pounds per square foot

The drag was estimated as follows:

$$D^* = C_D\, S\, q = (.008 + .008C_L^2 + \frac{C_L^2}{\pi A})\,(\frac{L}{C_L q})\, q = (\frac{.008}{C_L} + .008C_L + \frac{C_L}{\pi A})\, L$$

---

*Simplified expression for drag will give a representative, conservative estimate.

At 110 knots, the HU-1B helicopter rotor uses 765 horsepower which requires a 427 pound torque balancing force at the end of the tail boom. At $C_L = .363$ and $L = 477$, the drag is 48.6 pounds. At the design cruise speed, 48.6 pounds drag is equivalent to a power penalty of 16.4 horsepower.

At a design $C_L = .363$, the tail incidence angle will be 6.5°. A maximum $C_L = 2.4$ was assumed with full rudder deflection. The available counter-torque force based on the above considerations is compared with rotor torque in Fig. 2.

Velocity (Knots)	Tail $C_L = .363$	Forces $C_L = 2.4$	Rotor Power	Required Counter-Torque Force
110	477	----	765	477
100	392	----	650	406
90	319	----	567	354
80	352	1040	500	312
70	193	796	472	295
60	142	585	458	286
50	99	408	450	282
40	63	259	485	303
30	36	147	570	356

APPENDIX II

JET THRUST AUGMENTATION BY

TAIL PIPE CONSTRICTION

In order to evaluate the power penalty of increasing exhaust thrust by tail pipe constriction, two typical helicopter turbine engines were selected. Sufficient tail pipe restriction was assumed to cause a ten per cent drop in exhaust to inlet pressure ratios. The engines selected and the respective performance manuals are given below:

Engine	Performance Manual
Lycoming, Model T53-L-5	Specification No. 104.16-B dated December 10, 1959
General Electric, Model YT58-GE-8	Model specification No. E-1028, dated May 14, 1958

The T53-L-5 engine has the following characteristics at the take-off power rating:

Shaft horsepower (SHP) $= 960$ horsepower

Air mass flow (W_a) $= .322$ slugs per second

Exhaust temperature $(T_{10}) = 1590°R$

Exhaust area $= 203$ square inches

Exhaust thrust (F_n) $= 113$ pounds

Table IV on page 40 of the performance manual gives the change in shaft horsepower and exhaust thrust as a variable of exhaust pressures $(P_{t\ 10})$ as follows:

$$F_n = F_n(C_d)(\frac{P_{t\ 10}}{P_{t\ 10}}) = 113(.75)(1.11) = 94.0 \text{ pounds}$$

where $C_d = .75$ (from Fig. 4.3 of performance manual).

$$SHP = -SHP(C_d'')(\frac{P_{t\ 10}}{P_{t\ 10}}) = -960(1.07)(1.11) = -94 \text{ SHP}$$

where $C_d'' = 1.07$ (from Fig. 4.1 of performance manual).

This indicates a horsepower penalty of nearly 10 per cent to increase the exhaust thrust from 113 pounds to 207 pounds.

The YT58-GE-8 engine has a shaft horsepower (SHP) of 910 SHP when operating at an inlet to outlet pressure ratio (P_{t2}/P_{s6}) equal 0.90 at a shaft output speed 1950 revolutions per minute. Air flow (W_a), thrust (F_g), and exhaust temperature (T_6) at a $P_{t2}/P_{s6} = 1$ are as follows:

$W_a = 11.6$ pounds per second

$F_g = 114$ pounds

$T_6 = 1423°R$

The exhaust velocity (V_L) is then approximately

$$V_L = \frac{114(32.2)}{11.6} = 317 \text{ feet per second}$$

Sonic velocity (a_6) at the exhaust is

$$a_6 = \sqrt{(1.33)(32.2)(53.3)(1423)} = 1802 \text{ feet per second}$$

Which yields an exhaust Mach number of

$$M_6 = \frac{317}{1802} = .176$$

The ratio of stagnation (P^o_6) to static (P_6) pressures at the exhaust is

$$P^o_6/P_6 = (1 + \frac{.33}{2} M^2_6)^{\frac{1.33}{.33}} = 1.00205$$

The ratio of P^o_6 to the inlet static pressure $(P\infty)$ is then

$$P^o_6/P_\infty = (1.00205)(\frac{1}{.90}) = 1.113$$

The Mach number (M_{ex}) of the constricted exhaust can be found as follows

$$1 + \frac{.33}{2} M_{ex}^2 = (1.113)^{\frac{1}{4.02}} = (1.113)^{.249} = 1.027$$

$$M_{ex} = \frac{.027}{.165} = .404$$

The velocity (V_{ex}) in the constricted exhaust follows.

$$V_{ex} = (.404)(1802) = 730 \text{ feet per second}$$

The thrust (F_{ex}) produced with the constricted exhaust is

$$F_{ex} = \frac{11.6}{32.2} (730) = 263 \text{ pounds}$$

The normal power rating is 1050 SHP so that a thrust increase from 114 pounds to 263 pounds results in a shaft horsepower penalty of approximately 10 per cent.

APPENDIX III

ESTIMATION OF DUCT LOSSES

Energy losses in the duct due to non-isentropic flow will depend on specific design and will have wide variation with different configurations and duct material. The order of magnitude of these losses can be shown to be small if it is assumed the entire mass flow is completely mixed and flows in a constant area, straight duct with length equal to the rotor radius and cross-sectional area equal to the nozzle area. Fan, turning elbow, and nozzle losses are accounted for separately.

The change in kinetic energy of the flow caused by losses or gains of heat will be very small because of the low pressures in the duct. It can be shown that the net heat flow is negligible for normal augmentation ratios where the heat of vaporization from the fuel flow is recovered. For the example shown in Fig. 6, the mass flow ratio of free air to exhaust gases will be in excess of 12. This will result in a final flow temperature within 120° of free air temperatures. Heat flow through the duct walls at this temperature gradient will be dependent on the external flow conditions, but can be approximated by assuming a transfer coefficient equal to 4.0 BTU per degree-square foot hour. The duct has a diameter of 3.74 feet and the heat flow out is given by

$$Q_o = (4.0)(22)(3.74\pi)(120) = 124,000 \text{ BTU/hr.}$$

The engine is consuming 600 pounds per hour of JP-4 fuel which

has a heat of vaporization of 154 BTU per second (NACA RM E53 A21, page 21). This will be recovered in the duct when mixing with the augmenting air occurs. The heat gained by this cooling is

$$Q_i = 600 \times 154 = 924,000 \text{ BTU/hr.}$$

The net value is a negligible loss of 100,000 BTU/hr. which will change the flow temperature less than one degree.

Frictional losses can be estimated by assuming a typical friction coefficient of .005 (page 186, The Dynamics and Thermodynamics of Compressible Fluid Flow, Vol. 1, Shapiro). The duct wetted area is 258 square feet. The density can be found by assuming flow temperatures 120 degrees above standard and is

$$= \frac{2116}{(1715)(580)} = .00212 \text{ slugs per cubic foot}$$

With a duct flow of 162 feet per second, the duct drag is

$$D = (.005)(\tfrac{1}{2})(.00212)(162)^2(258) = 35.9 \text{ pounds}$$

This frictional loss is a small part of the required thrust and will be partly recovered from the energy of the vibrational mode gained in cooling the exhaust gases. The ratio of specific heats (γ) will change from approximately 1.33 at engine exhaust temperatures to 1.4 at duct flow temperatures. The thrust of the flow can be given by the expression

$$F = p A (1 + \gamma M^2)$$

where p , A , and M are the pressure, area, and Mach number of the nozzle. The change in momentum due to the change of ratio of specific heats alone will then be a measure of the kinetic energy recovered in the duct by cooling. Exhaust flow is Mach = .181 at atmospheric pressure and exhaust area equal 1.41 feet.

$$\Delta F = 2116 \ (1.41)(1.4 - 1.33)(.181) = 37.7 \text{ pounds}$$

The total losses of kinetic and thermal energy in the duct alone are then seen to be negligible.

BIBLIOGRAPHY

1. Shapiro, Jacob, Principles of Helicopter Engineering, New York: McGraw-Hill Book Co., Inc., Chapter 5, Section 5.3.

2. Ibid, p. 102.

3. Ibid, p. 94.

4. Gessow, Alfred and Myers, Garry C., Jr., Aerodynamics of the Helicopter, New York: The Macmillan Company, p. 45.

5. Klemin, Alexander, "Helicopter Engineering", Aero Digest Vol. 53, December, 1946, p. 91.

6. Aeronautical Engineering Review, Vol. 6:4, April, 1947, p. 66.

7. Shapiro, Jacob, The Journal of the Helicopter Association of Great Britain, p. 26.

8. Gessow and Myers, op. cit., p. 52.

9. Zucrow, M. J., Aircraft and Missile Propulsion, Vol. I, New York: John Wiley and Sons, Inc., p. 375.

10. Zucrow, M. J., Aircraft and Missile Propulsion, Vol. II, New York: John Wiley and Sons, Inc., p. 42.

11. Gessow and Myers, op. cit., p. 40.

12. Bell Helicopter Corporation, Report 204-099-763, undated, p. 37.

13. Smith, C. W., Aircraft Gas Turbines, New York: John Wiley and Sons, Inc., Chapter 8, p. 167-178.

14. Zucrow, M. J., Vol. I, p. 80-122.

15. Zucrow, M. J., Vol. II, p. 327-342.

16. Smith, C. W., Aircraft Gas Turbines, p. 289.

17. McClintock, Frank A. and Hood, Hall J., "Aircraft Ejector Performance", Journal of the Aeronautical Sciences, Vol. 13, November, 1946, p. 559.

18. Lockheed Aircraft Corporation Final Report for Contract Nonr-3067(00), December, 1960, p. 29.

19. Stepanoff, A. J., Turboblowers, John Wiley and Sons, Inc., p. 271.

20.   Harper, John J., "Tests on Elbows of a Special Design", Journal of the Aeronautical Sciences, Vol. 13, p. 591, November, 1946.

21.   Smith, C. W., Aircraft Gas Turbines, p. 134.

22.   Bell Helicopter Corp., Group Weight Statement, HU-1B, dated April 5, 1960.

23.   Bell Helicopter Corp., Report No. D 250-099-020, January 9, 1961, p. 23.

24.   Shapiro, Jacob S., "Some Technical Aspects of the W.9 Development", The Journal of the Helicopter Association of Great Britain, p. 27.

25.   Departments of the Army, the Navy, and the Air Force, Military Specifications, Helicopter Flying Qualities, Requirements for, MIL-H-850, November, 1952, p. 7.