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RECOMMENDATIONS FOR CORRECTING THE FLYING QUALITIES DEFICIENCIES OF THE GAZELLE HT Mk-3 HELICOPTER

A Thesis
Presented for the
Master of Science
Degree
The University of Tennessee, Knoxville

Keith W. Danel
May 1997

DISCLAIMER

The flight test results contained within this thesis were obtained during a United States Naval Test Pilot School sponsored academic project. The discussion of the data, conclusions, and recommendations presented are the opinion of the author and should not be construed as an official position of the United States Navy or the United States Naval Test Pilot School, Patuxent River, MD.

DEDICATION

This thesis is dedicated to my wife, Kimberley,
whose devotion and support made it possible,
and to my son Cooper, daughter Mollie,
and the Bump on the way.
May all three of you enjoy the educational opportunities
I have been afforded.
SEMPER FI!

ACKNOWLEDGMENTS

I would like to thank Mr. Robert Richards, Mr. Robert Miller, and Mr. J.J. McCue for their guidance and assistance in completing this thesis and for their commitment to the students of the United States Naval Test Pilot School.

ABSTRACT

In November of 1992, the author conducted an evaluation of the Gazelle HT Mk-3 helicopter as a final project for the United States Naval Test Pilot School. The purpose of that trial was to briefly, but comprehensively, evaluate the performance, flying qualities, and systems of the Gazelle to assess its suitability for the unarmed scout mission. The trial consisted of ground and flight tests that evaluated the cockpit, ground handling, flight control mechanical characteristics, low airspeed, and forward flight flying qualities of the Gazelle. Flight tests also evaluated the performance of the Gazelle in hover, vertical climb, level flight, and forward flight climb and descent profiles. Throughout the four hours of ground tests and six hours of flight tests, three serious deficiencies were uncovered: jack stall, fenestron stall, and negative, forward flight maneuvering stability. This thesis presents the test methods and research used to uncover these deficiencies, investigates their influencing factors, and analyzes and recommends engineering modifications to mitigate their effects. The overarching goal is to determine inexpensive and straight forward modifications to mitigate these three undesirable characteristics. It is also hoped that the analysis and recommendations in this thesis might serve as lessons learned for the future acquisition of similarly configured aircraft.

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SECTION 1: INTRODUCTION

1.1 BACKGROUND

In November of 1990, the Royal Air Force Aviation Standards Branch published a Flight Safety memorandum to Gazelle aircrew that cited fenestron stall as a possible contributing factor in at least 20 cases of aircraft accidents, although the manufacturer does not acknowlege the phenomenon. The memo emphasized that in all cases, mishandling by the pilot was the primary causal factor in that it instigated or exacerbated the stall. The Gazelle Aircrew Manual (Reference 1) states that the causes for fenestron stall are not known, but lists some flight conditions that possibly contribute to its occurrence, all conditions occur during low airspeed flight.

Fenestron stall is a flight phenomenon that resembles loss of tail rotor effectiveness for helicopters with conventional tail rotors. However, due to the unique design characteristics of the fenestron antitorque system, this condition is generally more sudden, more pronounced, and makes for a more difficult recovery.

In addition to fenestron stall, the Gazelle has two other highly undesirable forward flight deficiencies: jack stall and negative maneuvering stability. Jack stall is the term used for a flight condition where the Gazelle hydraulic system is overloaded by aerodynamic forces being fed backward through the flight control system. When those forces are strong enough to overload the hydraulic system they are also strong enough to prevent the pilot from making control inputs, resulting in temporary loss of aircraft control.

Negative forward flight maneuvering stability is a flight condition in which normal acceleration increases

rapidly and unpredictably with little corresponding aft longitudinal control input. Negative maneuvering stability is indicated by increasing forward longitudinal control position with increasing normal acceleration, and occurs in the Gazelle during moderate to aggressive forward flight maneuvers, specifically, steep angle of bank turns and symetrical pull ups. During these maneuvers, once normal acceleration begins to increase the Gazelle tends to pitch nose up without being further comanded to do so. The resulting large nose up pitch attitude excursions and the concomitant increase in load factor make it difficult for the Gazelle pilot to control airspeed and prevent airframe overstress.

1.2 PURPOSE

All three of the deficiencies addressed in this paper, fenestron stall, negative maneuvering stability, and jack stall, could lead to Gazelle flight mishaps. As such, correcting these deficiencies, or at least mitigating their effects, could enhance the safety of the Gazelle's operating envelope. This thesis presents the flight test methods and research used to uncover these deficiencies, analyzes their possible causes, and recommends potential engineering modifications to mitigate their effects.

There were other deficiencies uncovered through the course of these tests. However, the author considers fenestron stall, negative maneuvering stabiity, and jack stall to be potentially dangerous. Therefore, this thesis focuses on how to mitigate these deficiencies. It is also hoped that the following analysis and recommendations might serve as lessons learned for the future acquisition of similarly configured aircraft.

1.3 DESCRIPTION OF TEST AIRCRAFT

The Gazelle HT Mk-3 tested in this trial was a light, multi-purpose, single-piloted helicopter designed to operate at a maximum gross weight of 1800 kg (3970 lbs) while carrying a pilot, a co-pilot/observer, and up to three passengers. The test aircraft was co-produced by Westland and Aerospatiale. An aircraft 3-view diagram of the Gazelle is presented in figure D-1. The main rotor was driven clockwise when viewed from above, at a nominal 380 rpm. The main rotor head was a fully articulated type in that it had three flapping hinges on which the sleeves were designed to pivot allowing blade incidence to vary. The main rotor blades were made of laminated glass fiber and were designed to fold manually. The fenestron (tail rotor) had 13 light alloy blades shrouded by the tail fin, mounted on the fenestron gear box. Fenestron pitch change was made by movement of a pitch control plate within the fenestron gearbox. The fenestron was driven clockwise when viewed from the right (i.e. bottom blade rotated aft) at a nominal 5774 rpm. The side by side cockpit incorporated dual, conventional, irreversible flight controls, which were mechanically linked and boosted by a single hydraulic system, that operated at a nominal pressure of 590 psi. The hydraulic system used three main rotor servos and one tail rotor servo. A stability augmentation system (SAS) provided rate damping and attitude hold in pitch and roll at all airspeeds and in yaw below 48 KIAS. A more detailed flight control system description is presented in appendix E. A more detailed description of the test aircraft can be found in the Gazelle Aircrew Manual (Reference 1). The test

aircraft, SerNo XZ939, was representative of production aircraft, except for the addition of a sensitive airspeed boom and yaw vane and the absence of a tactical periscopic scout sight.

SECTION 2: METHOD OF TESTS

2.1 GENERAL

The entire Gazelle evaluation was conducted during one 4.0 hour daylight ground evaluation and four flights totaling 6.0 hours in daylight, visual meteorological conditions (VMC), at the Empire Test Pilot School (ETPS) Boscombe Down, UK. Flight tests were conducted from 350 to 10,000 feet MSL over an OAT range of -8.0 to +8.3 degrees Celsius. Gross weight ranged from 3955-3288 lbs with an associated center of gravity (CG) range of 116.9-115.8 inches.

Flying qualities evaluations were conducted in accordance with the procedures of United States Naval Test Pilot School (USNTPS), Rotary Wing Stability and Control, Flight Test Manual-107 (FTM-107), Reference 2. The FTM-107 primarily serves the USNTPS as a text for the aviators and engineers attending the USNTPS. It also serves as a reference document for anyone engaged in rotary wing flying qualities testing. The FTM-107 summarizes applicable theory, shows the relationships of flight test variables to the theory, describes the flight test techniques, and aids in the data management required for each test flight. Specific flight tests relevant to the topics under discussion in this paper are presented in table 1 below. Detailed tests and test conditions for all tests conducted during this trial are presented in table F-1.

Table 1

TESTS AND TEST CONDITIONS

TEST/TASK CATEGORY	TEST/TASK	AVE OAT °C	AVE AIR SPEED KCAS	AVE ALT	AVE WEIGHT/CG-lbs/inches	REMARKS
MANEUVERING STABILITY	COORDINATED CONSTANT AIRSPEED TURNS	3.1	100	2000'	3655/116.5	COLLECTIVE FIXED, BALL CENTERED, 15°,30°,45° TURNS LEFT AND RIGHT
	SYMMETRIC PULLUPS AND PUSHOVERS	3.1	100	2000'	3625/116.5	PULLUPS TO 2.0g PUSHOVERS TO 0.3g 1/4" INCREMENTS, MAX INPUT:+1 1/2, -1 3/4
LOW AIRSPEED	TRIMMED FLIGHT CONTROL POSITIONS/ CRITICAL AZIMUTH	8.3	0-30	10'-20' (500' MSL)	3612/116.5	000, 045, 090, 135, 180, & 270 REL WIND, 10 knot INCR, ESTIMATED SPEED AND AZIMUTH

Notes:

1. Crew of two, 100 kg of ballast, full fuel (JP-8) at engine start. Hydraulics, SAS, and Stick Feel were on. Clean configuration, doors on, engine air bleed systems off.

2. Test gross weight ranged from 1793-1491kg (3970-3288 lbs). 100 kg ballast was added for all flights and was positioned on the cabin floor, centered laterally, behind the front crew seats. Test center of gravity (CG) range throughout the flights was (116.9"-115.8"); a mid-range CG for the Gazelle. A sample load statement is presented in table F-2.

3. Altitudes at/above 500' are pressure altitudes (MSL). Altitudes below 500' are above ground level altitudes (AGL), unless otherwise noted.

4. Low airspeed tests were conducted in steady 10 knot winds.

Several different rating scales were used to quantify the Gazelle flying qualities throughout the flight testing of this trial. Two of these scales are relevant to the results discussed in this paper, the Cooper-Harper Handling Qualities Rating (HQR) scale and the Vibration Analysis Rating Scale (VAR). Detailed descriptions and examples of these rating scales are presented in appendix C.

Instrumentation included sensitive indicators for control positions, airspeed, sideslip angle, outside air temperature (OAT), fuel counts used, and aircraft load factor (n_z). Data were recorded on hand held data cards and a portable voice recorder and were plotted and analyzed manually in accordance with reference 2.

2.2 FORWARD FLIGHT TESTS FOR MANEUVERING STABILITY

2.2.1 GENERAL TEST TECHNIQUES

The forward flight tests used in this trial to assess the Gazelle's maneuvering stability were forward flight symmetrical pull-ups and pushovers and fixed collective steady turns. Symmetrical pushovers are not addressed in this paper since the maneuvering stability deficiency uncovered pertained to higher, positive normal acceleration levels associated with pull-ups and steady turns. Symmetrical pull-up and fixed-collective steady turn tests were both conducted by first establishing coordinated, straight and level flight at 100 KCAS with control forces trimmed to zero.

2.2.2 STEADY TURNS

Once the aircraft was trimmed, the desired bank angle was established while maintaining constant collective pitch and airspeed without retrimming control forces; altitude was allowed to vary. These turns were conducted to the left and right using bank angles of 15°, 30°, and 45°. The primary data of interest were aircraft load factor (n_z) and longitudinal control position (δ_{long}) from the trim position. These data were collected once bank angle, airspeed, and load factor were stabilized.

2.2.3 SYMMETRICAL PULL-UPS

Once the aircraft was trimmed, a small aft cyclic control input was made to decelerate the aircraft below the trim airspeed, followed by a small forward cyclic input to accelerate back toward the trim airspeed. As the aircraft was accelerating back to trim airspeed the cyclic control was returned to the trim position. Just prior to the moment the aircraft reached the trim airspeed of 100 KCAS, a precise amount of aft cyclic control was input to initiate a pitch up moment. The aft cyclic inputs ranged from 1/4" to 1 1/2", in increments of 1/4". The primary data of interest collected for this test were longitudinal control displacement from the trim position and peak load factor.

2.3 LOW AIRSPEED TESTS FOR DIRECTIONAL STABILITY

2.3.1 GENERAL

The two salient low airspeed tests were trimmed flight control positions and critical azimuth. The purpose of the trimmed flight control position test was to determine the control margins and handling qualities in simulated crosswind, downwind, and low airspeed forward flight. The purpose of the critical azimuth test was to evaluate control margins and pilot workload with changes in airspeed and relative wind azimuth. The qualitative evaluation of trim changes was an integral part of both tests while the qualitative evaluation of pilot workload was integral to critical azimuth analysis (Reference 2, p.8.48-8.50).

2.3.2 TEST TECHNIQUES

The test technique for evaluating trimmed flight control positions and critical azimuths normally involves flying a formation position on a ground pace vehicle with a calibrated airspeed system. A pace vehicle was not available for this trial so the test was conducted using reported surface winds at the field and pilot estimated ground speed to determine airspeed.

The trimmed flight control and critical azimuth tests were accomplished by collecting quantitative trimmed flight control position and pitch attitude data at relative wind azimuths of 000°, 045°, 090°, 135°, 180°, and 270°. Test airspeed was varied in 10 knot increments. Handling qualities ratings (HQR) were assigned based on the pilot workload required to maintain heading, attitude, and altitude at each test condition. Aircraft vibration ratings (VAR) were also assigned.

Data were recorded after establishing a zero drift, unaccelerated, trimmed hover on the desired test heading, at the estimated test airspeed. The stick feel, stability augmentation system (SAS), and hydraulic system were engaged for these tests. Test altitude was maintained between 10 and 20 feet AGL at a pressure altitude was 500' MSL. Outside air temperature (OAT) was 8.3° C, average aircraft weight was 3612 pounds, while average CG was 116.5" aft of aircraft datum. The average CG of 116.5" was a mid-range CG for the Gazelle, but it was approximately 10" aft of the main rotor mast location and varied little with changes in weight.

Data for low airspeed trimmed flight control position tests and critical azimuth tests were gathered simultaneously on the same flight test point, but were

plotted differently to depict two distinct, but related results. The trimmed flight control position data were plotted with flight control position and pitch attitude versus forward, aft, left, and right airspeeds. The focus of the trimmed flight control position plots was on control margin remaining for various wind speeds at the four major relative bearings.

The critical azimuth data were plotted two ways. One depiction plotted flight control position versus wind azimuth for a constant 30 knot estimated airspeed. The other depiction plotted HQRs and VARs on a circular plot versus wind azimuth and estimated airspeed. The focus of these two plots was on pilot control workload at each test condition.

2.4 SCOPE OF TESTS

Tests were conducted within the limits of the Gazelle Aircrew Manual (Reference 1), except where modified by the ETPS Gazelle Flying Limitations (Reference 3) or the test plan for this trial. The following limits directly affected the scope of these tests:

A. Normal Acceleration: 0 to +2.0 g
B. Sideslip angle: ± 20° up to 108 KIAS
C. Maximum sideward/rearward flight: 30 KTAS
D. Maximum power was defined as the red index needle on the torque meter, the collective maximum pitch stop, or T4 of 580° C.
E. Control response test limits were as follows:
    1. Attitude limits
        a. Pitch and roll: ± 45°
        b. Recoveries begun by 30°
    2. Forward flight
        a. Longitudinal: ± 1 3/4"

(load factor limit)

b. Lateral: ± 1 3/4" (roll attitude limit of 45°)

3). Low airspeed

a. Longitudinal and lateral: ± 1" (test plan limit)

b. Pedal: ± 1/2" (test plan limit)

SECTION 3: MANEUVERING STABILITY

3.1 GENERAL

Maneuvering stability is influenced by the forces and moments acting on the helicopter due to a disturbance in normal acceleration. The forces and moments due to a disturbance in normal acceleration are balanced by longitudinal control forces and moments. Maneuvering stability is characterized and quantified by the magnitude and direction of change in longitudinal control position (δlong) relative to changes in normal acceleration, measured in flight test as the change in load factor (n_z) (Reference 2, p.6.37). A helicopter with positive maneuvering stability tends to return to trim conditions and a load factor of one, following a disturbance. Positive maneuvering stability is indicated by aft longitudinal control position change from trim for an increasing change in load factor. A helicopter with negative maneuvering stability tends not to return to trim or a load factor of one, following a disturbance (Reference 2, p.6.37). Therefore, negative maneuvering stability is indicated by neutral or forward change in longtudinal control position from trim with increasing changes in load factor.[1]

3.2 MANEUVERING STABILITY RESULTS

Gazelle maneuvering stability data are presented in figure G-1. The Gazelle maneuvering stability gradients (δ_{long} vs. n_z) were positive for symmetrical pull-ups and left steady turns, in that increasing aft

[1] A load factor of 1.0 will, in some instances, be referred to as 1.0g or one multiple of the acceleration of gravity, 32.174 lbm-ft/lbf-sec^2.

longitudinal control position displacement (δ_{long}) was required to generate increased load factor (n_z). However, the gradients for symmetrical pull-ups and left steady turns began to flatten out at an n_z of 1.6 and appeared as though they would reverse (becoming negative) beyond an n_z of 2.0; however, additional data beyond a 2.0 load factor would be required to confirm this. The Gazelle maneuvering stability gradient during right steady turns was only positive to an n_z of approximately 1.2. As load factor increased beyond 1.2, the maneuvering stability gradient quickly reversed, becoming negative at an n_z of about 1.5.

Average maneuvering stability gradients were shallow. The average δ_{long} vs. n_z gradient during symmetrical pull-ups and left steady turns was approximately 1 1/2" per each 1.0 increment of n_z; or 1 1/2" per "g." During right steady turns, average δ_{long} vs. n_z gradient was less than 1/2" per "g." During symmetrical pull-ups, load factor increased approximately linearly for the first 1 1/4" of aft longitudinal control displacement, generating a gradient of +0.1g for each 1/4" aft longitudinal control input. A symmetrical pull-up using an aft δ_{long} of 1 1/4" generated a load factor of 1.6. However, when symmetrical pullups were executed using an aft δ_{long} of 1 1/2", normal acceleration increased non-linearly, to an n_z of 2.0; an increase of 0.4 for an additional 1/4" of aft δ_{long}.

Additionally, longitudinal control force cues to increased load factor for all three maneuvers were qualitatively assessed in flight as, "very light." This qualitative assessment is reinforced by the control force per "g" (Force vs. n_z) cross plot presented in figure G-1 The Force vs. n_z plot was gained through a cross plot of force versus longitudinal control deflection data (Force vs. δ_{long}) gathered during

mechanical characteristics testing, and longitudinal control position vs. load factor data (δ_{long} vs. n_z) gathered during maneuvering stability testing.[2]

For symmetrical pullups, the overall Force vs. n_z gradient from a load factor of 1.0 to 2.0, was approximately 1.6 lbs. However, for the load factor range of 1.5 to 2.0 for symmetrical pullups, the Force vs. n_z gradient was negligible. In other words, during symmetrical pull-ups, the additional aft longitudinal control force was negligible for an increase in load factor from 1.5 to 2.0. According to Aeronautical Design Standard-33 (ADS-33)[3], the Force vs. n_z gradient for helicopter maneuvering stability should be no less than 3 lbs per g (Reference 4, p.33).

The maneuvering stability gradient reversals at low load factors, the shallow average gradients, and the light control forces make Gazelle maneuvering stability unsatisfactory. Maintaining airspeed within ± 10 KCAS at 45° angle of bank (AOB) or greater was not possible. Maintaining airspeed within ± 15-20 KCAS at 45° AOB required longitudinal control inputs of ± 1-2" every 3-4 seconds to correct for large, unpredictable nose up pitch attitude changes that tended to be divergent (HQR 7).[4]

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[2] Mechanical characteristics and forward flight control position results are included for reference in appendix H. Mechanical characteristics data are presented in figures H-1 through H-5. Forward flight trimmed flight flight control position data are presented in figures H-6 through H-8.

[3] ADS-33 is the military specification that contains the requirements for the flying and ground handling qualitites of rotorcraft.

[4] As presented in Figure (1), Enclosure (1), Appendix C; a handling qualities rating (HQR) of 7 equates to a major aircraft deficiency that requires maximum tolerable pilot compensation, to achieve adequate performance criteria for that specific maneuver. Desired performance criteria were not achievable.

The combined effect of these undesirable maneuvering stability characteristics degrade the Gazelle pilot's ability to aggressively maneuver the Gazelle to evade enemy air or ground fire while conducting the Scout mission. The consequence is a potential decrease in the Scout pilot's chance of survival on the battlefield. Using the deficiency definitions presented in appendix B, the author considers the Gazelle's maneuvering stability to be a Part I deficiency.

3.3 FACTORS INFLUENCING MANEUVERING STABILITY

Characterizing the maneuvering stability of the Gazelle involved a relatively straight forward flight test process. Determining how to improve the Gazelle's maneuvering stability is more involved, requiring analysis of the stability derivatives involved. There are four factors, or stability derivatives, that affect maneuvering stability: angle of attack stability (M_w), pitch rate damping (M_q), vertical height damping (Z_w), and longitudinal control sensitivity ($M_{\delta_{long}}$). The two primary influences of maneuvering stability are M_w and M_q. Z_w and $M_{\delta_{long}}$ are relatively minor in comparison. These four stability derivatives affect maneuvering stability through equation 1 below.

$$\frac{d\delta_{long}}{dn_z} = \frac{-g}{u_0 M_{\delta_{long}}} \left(\frac{M_q - \frac{u_0 M_w}{Z_w}}{1 + \frac{u_0 M_w}{M_{\delta_{long}}}} \right)$$

Equation 1: Maneuvering Stability Gradient
(Adapted: From Reference 5, p.58.[5])

Where:

$d\delta_{long}$	Change in longitudinal control position
dn_z	Change in load factor
g	Acceleration of gravity
u_0	Forward velocity; test trim airspeed in this case
$M_{\delta_{long}}$	Longitudinal control sensitivity derivative
M_q	Pitch rate damping derivative
M_w	Angle of attack stability derivative
Z_w	Vertical height damping derivative

Equation 1 applies specifically to symmetrical pull-up and pushover maneuvers. For steady turns, the right side of equation 1 is multiplied by a factor of $(1 + 1/n_z^2)$.

The three moment derivatives that affect maneuvering stability take the form of equation 2.

---

[5] Equation 1 is adapted by substituting $M_{\delta_{long}}$ for $M_{B_{1s}}$, which are related by the gearing ratio, $\partial\delta_{long}/\partial B_{1s}$. $M_{B_{1s}}$ is the pitch moment due to longitudinal cyclic pitch angle.

$$\Delta M = \left(\left(Th + \frac{ebMs\Omega^2}{2} \right) \Delta a_{1s} + \left(h' + ha_{1s} \right) \Delta T + h\Delta H + \Delta M_{CGf\&t} \right)$$

Terms: (1) (2) (3) (4)

Equation 2: Forward Flight Pitching Moment
(Source: Reference 2, Equation 6.4, p.6.7)

Where:

a_{1s}	Longitudinal flapping angle, shaft referenced
b	Number of main rotor blades
e	Flapping hinge offset
H	Rotor hub force, perpendicular to shaft
h	Height of hub above Center of gravity (CG)
h'	Longitudinal distance between rotor mast & CG
M	Net moment about y axis, Pitch moment
M_{CGf+t}	Pitch moment due to aerodynamic forces on fuselage/tail
M_S	Blade mass moment; $M_S = (m_{blade})(r_{CG})$
M_H	Moment due to rotor hub forces; $M_H = (ebM_S\Omega^2/2)(a_{1s})$
T	Thrust
Ω	Rotor angular velocity

According to reference 2, term (1) is proportional to the aft tilt of the tip path plane (TPP)(Δa_{1s}). The first factor of term (1), $Th\Delta a_{1s}$, is the moment caused by the aft thrust component due to the TPP tilt, and is a function of the height, h, of the rotor above the CG and the change in longitudinal flapping angle (a_{1s}). The second factor of term (1), $(ebM_S\Omega^2/2)\Delta a_{1s}$, is the moment due to rotor hub forces, or the flapping hinge offset moment corresponding to the aft tilt of the TPP.

Term (2) is the pitching moment proportional to the change in thrust (ΔT) as influenced by the height, h of

the rotor above the CG, the distance, h' between the rotor shaft and the CG, and finally the trim value of a_{1s}.

Term (3) is the pitching moment proportional to the change in rotor hub force (ΔH) and depends on the height of the rotor above the CG. Term (4) is the pitching moment due to the fuselage and tail. Term (4) is primarily a function of horizontal tail size and its distance from the CG. Each of the three maneuvering stability moment derivatives, M_w, M_q, and $M_{\delta_{long}}$ are analyzed with respect to the four terms of equation 2. This analysis provides insight into how these derivatives might change to improve maneuvering stability (Reference 2, p.6.7-6.8).

3.4 ANALYSIS OF MANEUVERING STABILITY DERIVATIVES

3.4.1 GENERAL

Knowledge of the probable sign (+/-) of each derivative is necessary to determine how they must change to improve maneuvering stability. Figure 1 below presents the axis reference system used in this analysis. The arrows indicate direction of positive moments and rates.

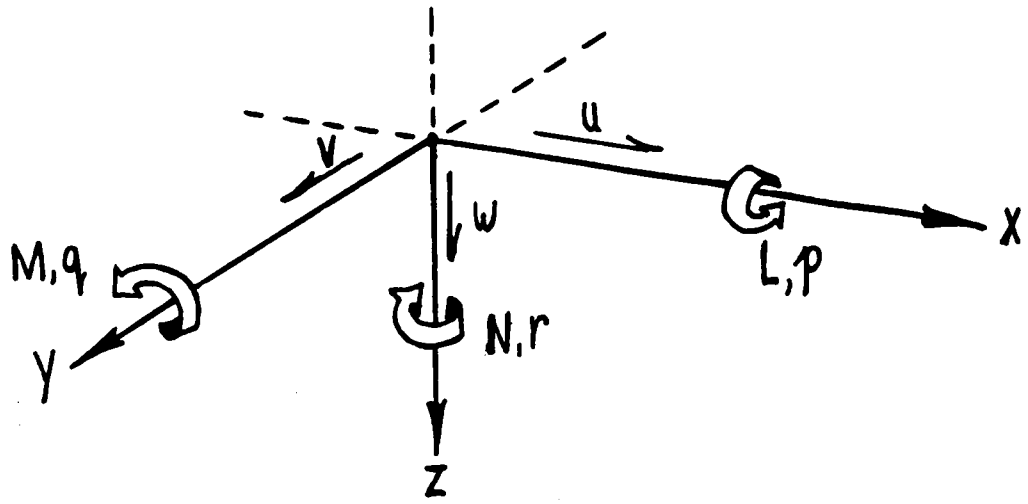


Figure 1: Axis Reference System

Where:

X Roll axis	Y Pitch axis	Z Yaw axis
p Roll rate	q Pitch rate	r Yaw rate
L Roll moment	M Pitch moment	N Yaw moment
u Long. velocity	v Lateral velocity	w Vert. vel.

3.4.2 ANGLE OF ATTACK STABILITY DERIVATIVE (M_w)

$$M_w = \frac{1}{I_{yy}}\left(\left(Th + \frac{ebM_s\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial w} + \left(h' + ha_{1s}\right)\frac{\partial T}{\partial w} + \frac{h\partial H}{\partial w} + \frac{\partial M_{CGf\&t}}{\partial w}\right)$$

Terms: (1) (2) (3) (4)

Equation 2a: Angle of Attack Stability Derivative
(Source: Reference 2, Equation 6.8, p.6.13)

First consider improvements to maneuvering stability through changes to angle of attack stability (M_w). M_w can be negligible, positive, or negative for a helicopter depending on the relative dominance of the

main rotor with respect to the tail. M_w is positive (+) when the main rotor dominates over the tail, negative (-) when the tail dominates over the main rotor, and neutral when neither dominates. Positive M_w is destabilizing in that it tends to cause the helicopter to depart from the trim condition with perturbations in w. Negative M_w is stabilizing in that it tends to cause the helicopter to return to trim condition after perturbations in w. M_w for the Gazelle is positive, or destabilizing at the test trim airspeed of 100 KCAS, as evidenced by the tendency of the Gazelle to depart from trim, to continue to pitch nose up, with small increases in angle of attack.

Analysis of the terms of equation 2a is integral to determining ways to mitigate the Gazelle's unsatisfactory maneuvering stability. The first three terms of equation 2a give the contributions of the main rotor to M_w, while the fourth term gives the contribution of the tail. There is little variation in rotor hub force (H) with changes in w, therefore the contributions of term (3) are small. Additionally, the other element of term (3), the height of the main rotor above the CG (h), would be neither practial nor cost effective to modify. Therefore, making changes to the elements of term (3) are not practical.

The contributions of terms (1) and (2) are primarily a function of CG location, flapping hinge offset (e), and trim moments due to the fuselage and tail. The Gazelle incorporates a flapping hinge offset and an aft CG.[6] Nose up, or positive, destabilizing pitching moments are generated by both of these characteristics at the trim airspeed when angle of attack is increased. Reducing

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[6] Average CG for maneuvering stability tests was 116.5" aft of aircraft datum. The mast was positioned about 106.6" aft of aircraft datum.

the flapping hinge offset of the Gazelle rotor system is not a feasible means of improving maneuvering stability either. On the other hand, changing CG location might have potential. Moving the CG forward reduces the moment arm, h', which will serve to make Mw less positive at 100 KCAS.

Finally, term (4) is the change in pitching moment about the CG due to changes in w at the tail. Term (4) is a major contributor to the sign and magnitude of M_w. A positive change in w causes an increase in angle of attack at the tail which yields a tail up load and a nose down, or stabilizing pitch moment. Modifying the horizontal tail can influence a change in M_w through term (4). For example, increasing the size of the horizontal tail or increasing the moment arm of the tail by moving the tail aft and/or moving the CG forward in this case can make M_w less positive.

3.4.3 PITCH RATE DAMPING DERIVATIVE (M_q)

$$M_q = \frac{1}{I_{yy}}\left(\left(Th + \frac{ebM_s\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial q} + \left(h' + ha_{1s}\right)\frac{\partial T}{\partial q} + \frac{h\partial H}{\partial q} + \frac{\partial M_{CGf\&t}}{\partial q}\right)$$

Terms: (1) (2) (3) (4)

Equation 2b: Pitch Rate Damping Derivative
(Source: Reference 2, equation 6.7, p.6.11)

The next major derivative influencing the maneuvering stability gradient is the pitch damping derivative (M_q). M_q is negative (-) in sign and has a stabilizing effect on maneuvering stability (Reference 2, p.4.56). Analysis of equation 1 reveals that a decrease in M_q (making it more negative), increases the δ_{long}/n_z gradient. Analyzing the terms of equation 2b with

respect to moment changes due to pitch rate presents some options for improving maneuvering stability.

The first factor of term (1), $Th(\partial a_{1S}/\partial q)$, provides us a relatively unfeasible option for increasing maneuvering stability, which is to increase the height, h of the main rotor above the CG. However, the second factor of term (1), $ebM_S\Omega^2/2(\partial a_{1S}/\partial q)$, may have potential. The $ebM_S\Omega^2/2$ element of the term was dismissed in earlier discussions because of the complexity of retrofitting these parts of the main rotor system. However, $\partial a_{1S}/\partial q$ presents an interesting characteristic worth pursuing.

The factor $\partial a_{1s}/\partial q = -16\Omega/\gamma$, where γ, the rotor system Lock number, equals $\rho acR^4/I_{blade}$; or $\partial a_{1s}/\partial q = -16\Omega I_{blade}/\rho acR^4$. The key here to potentially increasing M_q is through the I_{blade} term. Increasing the inertia of the rotor blades through the use of rotor blade tip weights for example, could make $\partial a_{1s}/\partial q$ and consequently M_q, more negative. Additionally, an Mq decrease is gained by increasing the blade mass moment (M_S); $M_S = (m_{blade})(r_{CG})$. Tip weights increase m_{blade} and r_{CG} of the blades which further decreases, or improves Mq.

Term (4), $\partial MCG_{f\&t}/\partial q$ is the other dominant term in determining the value of M_q. As the aircraft pitches nose up, the horizontal tail pitches down, inducing an upward velocity on the horizontal tail. The nose up pitch rate generates an angle of attack increase at the horizontal tail that creates a corresponding nose down pitching moment. Consequently, if the size of the Gazelle horizontal tail and/or the length of its moment arm from the CG are increased, the overall value of M_q becomes more negative, subsequently improving the maneuvering stability gradient.

With respect to term (2) of the M_q equation, changes in pitch rate have little impact on changes in thrust.

Therefore, the impact of term (2) on M_q is small. With respect to term (3), $\partial H/\partial q$ is positive, so the inclination is to reduce it. However, its magnitude is about a quarter that of term (1) and making changes to the height, h, of the rotor system is impractical. For similar reasons, changes to the factors of terms (2) and (3) are impractical.

3.4.4 VERTICAL HEIGHT DAMPING DERIVATIVE (Z_w)

The vertical height damping derivative (Z_w), is expressed by equation 3:

$$Z_w = -\left(\frac{1}{m}\right)\left(\frac{\partial T}{\partial w}\right)$$

Equation 3: Vertical Height Damping Derivative
(Source: Reference 5, p.29)

Vertical height damping (Z_w) is the vertical force due to changes in thrust resulting from changes in vertical velocity. Z_w is usually negative (-) due to the effects of vertical velocity (w) on Thrust (T) (Reference 2, p.4.55). A positive Δw induces an increase in angle of attack on the rotor system, resulting in a corresponding increase in thrust, which acts to counter the descent. The converse is true for a negative Δw. Analysis of equation 1 shows that decreasing Z_w (making it more negative), tends to increase the ratio of δ_{long}/n_z by increasing the ratio of M_w/Z_w, since M_w is positive for the Gazelle at the trim airspeed.

Nevertheless, $\frac{1}{Z_w}$ is merely a multiplier of M_w in equation 1 and is more of a fine tuning factor rather than a primary influence. Additionally, changing Z_w to improve maneuvering stability would be complicated and

impractical, as it would involve a change to any number of aerodynamic characteristics of the rotor system such as the number of blades and blade chord.

3.4.5 LONGITUDINAL CONTROL SENSITIVITY ($M_{\delta_{long}}$)

$$M_{\delta_{long}} = \frac{1}{I_{yy}}\left(\left(Th + \frac{ebM_S\Omega^2}{2}\right)\frac{\partial a_{1s}}{\partial \delta_{long}} + \left(h' + ha_{1s}\right)\frac{\partial T}{\partial \delta_{long}} + \frac{h\partial H}{\partial \delta_{long}} + \frac{\partial M_{CGf\&t}}{\partial \delta_{long}}\right)$$

Terms: (1) (2) (3) (4)

Equation 2c: Longitudinal Control Sensitivity Derivative (Adapted: Reference 2, equation 6.5, p.6.8)

Longitudinal control sensitivity, or pitch sensitivity ($M_{\delta_{long}}$) is usually positive (+) in sign and appears in two locations in equation 1 (Reference 2, p.4.56).[7] The effect on maneuvering stability from a change to $M_{\delta_{long}}$ in one location of equation 1 is offset somewhat by the corresponding and opposite effect to maneuvering stability due to the same change to $M_{\delta_{long}}$ in the other location of equation 1. Additionally, attempting to improve maneuvering stability through changes to longitudinal control sensitivity has ramifications for other flying quality characteristics such as longitudinal control response.

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[7] The table of Aerodynamic Stability Derivatives in Reference 2 discusses $M_{B_{1s}}$ vice $M_{B_{1s}}$. This table states that the usual sign of $M_{B_{1s}}$ is negative (-). $M_{B_{1s}}$ is the pitch moment due to longitudinal cyclic pitch angle, whereas $M_{\delta_{long}}$ is the pitch moment due to longitudinal cyclic control position. $M_{B_{1s}}$ and $M_{\delta_{long}}$ are related by the gearing ratio, $\partial\delta_{long}/\partial B_{1s}$ which is negative (-) in sign. $M_{\delta_{long}} = M_{B_{1s}}(\partial\delta_{long}/\partial B_{1s})$; therefore, $M_{\delta_{long}}$ is usually positive (+) in sign.

Through the course of this trial, the author found the Gazelle's longitudinal control response satisfactory in both the forward flight and low airspeed regimes. An increase in $M_{\delta_{long}}$ might increase longitudinal control power so much that an unsatisfactory increase in pitch rate results for a given longitudinal control input.

Additionally, the value of $M_{\delta_{long}}$ is influenced primarily by terms (1) and (2) of equation 2c. Consequently, attempts to change $M_{\delta_{long}}$ involve physical changes to the design of the main rotor system. Such changes are expensive and complicated. For all these reasons, changes to $M\delta_{long}$ are not practical for improving the Gazelle's maneuvering stability.

3.4.6 STABILITY AUGMENTATION SYSTEM

Another way to potentially improve the Gazelle's maneuvering stability gradient is by increasing the pitch rate damping (M_q) indirectly, through changes to the stability augmentation system (SAS). The Gazelle possesses a SAS which incorporates a pitch rate damper that is designed to make limited authority pitch control inputs to the rotor, to damp rotor perturbations due to wind gusts. The pitch rate damping of the Gazelle SAS could be increased by either increasing the gains within the pitch computer or by increasing the authority of the pitch actuator. The Gazelle SAS is described in greater detail in the flight control description presented in appendix E.

Increasing the gains within the pitch computer would seem relatively simple and inexpensive, but the amount of gain increase might be constrained by what is considered acceptable during a SAS failure known as a hardover. A SAS hardover is a SAS malfunction that causes the computer to command the pitch actuator to

deflect to the SAS limit. In this malfunction, if the pitch computer gains are too high, the pitch input made to the rotor system by the SAS computer may be too large and occur too fast for the pilot to counter with cyclic control inputs.

There is another condition that must first be met in order to make a gain change to the pitch computer. The pitch SAS actuator must have some authority remaining, (i.e. it must not be saturated, or deflected to a physical limit) at the higher levels of normal acceleration at which negative maneuvering stability occurs in the Gazelle. If the actuator is not saturated and the level of gain increase is acceptable, then this is a viable option for improving maneuvering stability.

If on the other hand, the SAS actuator is saturated during steady turns at n_z >1.25 and symmetrical pullups of n_z >2.0, then increasing the authority of the SAS pitch actuator might be a solution. This option requires hardware changes to the SAS actuator that begin to complicate the solution, but it still seems relatively straight forward and inexpensive compared to changing rotor system characteristics. In any case, a flight test program is required in order to determine first whether the SAS actuator is saturated during these maneuvers and second to determine the highest acceptable level of gain in the pitch computer during a SAS hardover.

3.4.7 CONTROL FORCE CUES

After modifications are made to improve the δ_{long}/n_z gradient, there still exists the problem with weak flight control force cues to the onset of normal acceleration. "The maneuvering stability or pitching moment change with change in load factor is determined

from the variation of longitudinal control position and control force with normal acceleration ..."(Reference 2, p.6.67). Increasing the force required to displace the longitudinal control enhances maneuvering stability in a flying qualities sense, because it increases the pilot's awareness to the onset of normal acceleration.

Control forces associated with longitudinal control displacement are light in the Gazelle. These forces are generated artificially for the lateral and longitudinal controls through a stick feel system, otherwise commonly known as a force trim system. The stick feel system incorporates electro-magnetic brakes that anchor the pitch and roll actuators to a datum. Springs provide the artificial control force. Incorporating stronger springs, ones that require greater force to displace the cyclic might improve the pilot cueing to the onset of acceleration during aggressive maneuvers. However, as discussed in Reference 2 (p.6.43), increasing the strength of stick feel springs may negatively alter the control force versus airspeed gradient. This effect has been countered in some helicopters through the use of proportional force cues that increase as a function of pitch rate. The obvious drawback here is that the solution begins to get complex and costly.

3.5 SUMMARY OF POTENTIAL MANEUVERING STABILITY IMPROVEMENTS

In review, the most straight forward options for improving the Gazelle's maneuvering stability are first to move the CG forward to decrease M_w. Moving the CG forward to improve maneuvering stability could be accomplished by adding dead weight to the nose of the aircraft or by remounting current aircraft systems more forward in the fuselage. Adding more weight, especially

dead weight, to an aircraft is never a desirable option. However, the number of systems and possible remount locations are limited in the Gazelle.

A second option is to increase M_q by adding rotor blade tip weights. Adding tip weights increases I_{blade}, thereby increasing $\partial a_{1s}/\partial q$. Tip weights also increase M_q by increasing M_s. This solution is straight forward, but it isn't trivial. Structural analysis and ground and flight test is required to validate this option. If adding tips weights proved valid, an autorotational evaluation is also prudent to validate that the autorotational characteristics of the rotor system are still satisfactory. It is possible the main rotor RPM (Nr) retention characteristics during autorotative flight would be enhanced by adding blade tips weights, but this would need to be tested to be confirmed.

A third option is to increase the size of the horizontal tail or its location in order to increase the moment arm between it and the CG. Installing a larger horizontal tail is potentially one of the more effective ways to decrease M_w, thereby improving the Gazelle's maneuvering stability.

A fourth option, and perhaps the best with which to begin, is to modify the pitch rate damper in the Gazelle stability augmentation system, either by increasing the gains of the pitch computer or, if necessary, by increasing the control authority of the pitch actuator itself. As already discussed, this would necessarily involve some ground and flight testing to determine the saturation point of the SAS actuator and the maximum acceptable pitch rate damping gain during a pitch SAS hardover.

A fifth option is a mechanical characteristics modification to increase the control force required to displace the longitudinal control. Increasing the

control force required to displace the longitudinal control enhances pilot cueing to the onset of normal acceleration, but may degrade the control force versus airspeed gradient and does nothing by itself to correct the aerodynamic deficiency.

The best means of improving the Gazelle's maneuvering stability is a stepwise approach that first involves modifications to the SAS and moving the CG forward, then installing a larger horiontal tail, and finally adding rotor blade tip weights.

SECTION 4: LOW AIRSPEED DIRECTIONAL STABILITY

4.1 GENERAL

The Gazelle suffers from two related directional control deficiencies in the low airspeed regime: yaw instability and fenestron stall. Yaw instability during right sideward flight, or a hover with a right relative wind, causes moderately large, frequent, and unpredictable excursions in yaw attitude. Yaw instability in right sideward flight is also characterized by moderately high pedal vibrations and severely limited directional control margins. All of these characteristics combine to cause high pilot workloads for the Gazelle pilot while attempting to control aircraft heading. Fenestron stall, which normally occurs during low airspeed maneuvers involving left pedal turns (tail to the right), is characterized by a total loss of yaw control that creates high left yaw rates. These sudden, dramatic left yaw rates are extremely difficult to counter once they occur. Gazelle yaw instability during right sideward flight and fenestron stall occur primarily due to the recirculation of vortices around the vertical fin back in the vicinity of, or into the intake of the fenestron. For a conventional tail rotor, this recirculation of vortices is referred to as tail rotor vortex ring state.

4.2 YAW INSTABILITY

Low airspeed directional stability was tested through trimmed flight control position and critical azimuth tests. Test data are presented in figures G-1 through G-5. The time allotted for the test precluded critical azimuth testing at 315° and 225° relative to the nose.

Gazelle control forces could be trimmed to zero, no control coupling was noted, and the field of view was excellent throughout the test envelope. Based on these trimmed flight control position and critical azimuth tests alone, the Gazelle critical azimuth region exists from 090° to 135° relative. Specific data summaries describing the nature of the handling qualities deficiencies that comprise this critical azimuth region are presented in paragraphs 4.2.1 through 4.2.4, below.

4.2.1 Results at 20 KTAS on Azimuth 090° Relative

- Maintaining heading ±5° was slightly difficult, requiring alternating left and right pedal inputs of 1/2" to 3/4", every 2-3 seconds (HQR 4).

4.2.2 Results at 30 KTAS on Azimuth 090° Relative

- Maintaining heading ±5° was difficult requiring alternating left and right pedal inputs of 1", every 2-3 seconds (HQR 5).

- A moderately distracting, high frequency vibration developed in the pedals (VAR 6).

- A slightly uncomfortable 5° right wing down attitude was encountered.

- Pedal activity required to maintain heading brought the left pedal margin remaining to within 5% of the control limit; 10% remaining is the acceptable minimum, as established in military specification MIL-H-8501A (Reference 6).

4.2.3 Results at 30 KTAS on Azimuth 135° Relative

- Full left pedal applied (0% margin remaining) would not stop a right yaw rate of approximately 10-15°/sec (HQR 10).

4.2.4 Yaw Instability Conclusions

The loss of heading control in 30 knot winds from the 135° relative azimuth in a hover close to terrain will cause the Gazelle Scout pilot to impact the terrain with the aircraft tail. Employing the deficicency definitions standards presented in appendix B, the author considers the loss of heading control in 30 knot winds from the 135° relative azimuth, a Part I deficiency. Additionally, the increased pedal activity, moderately distracting, high frequency pedal vibration, and limited directional control margin remaining at 30 KTAS and 090°, will distract the Scout pilot during right horizontal unmasking maneuvers, and is considered a Part III deficiency.

4.3 FENESTRON STALL

By November 1990, over 20 cases of fenestron stall were reported within the British Royal Air Force (RAF), which flies the Gazelle as a Scout platform. A memo published by the RAF Aviation Standards Branch stated that when encountered, fenestron stall was likely to lead to an aircraft mishap. The memo emphasized that in all reported cases of fenestron stall, mishandling by the pilot was the catalyst. Whether mishandling by the pilot creates the conditions for fenestron stall or exacerbates them is not as important as realizing that some form of pilot compensation is required to account

for an undesirable aircraft characteristic. That undesirable characteristic is a sudden and dramatic loss of yaw control resulting in high left yaw rates during low airspeed maneuvers involving left yawing turns, with an initial wind condition of 10 knots or greater from the 045° - 090° relative. The sudden left yaw rate is extremely difficult to counter with cockpit control inputs and also defines a critical azimuth region.

The flight condition under which fenestron stall usually occurs is during low airspeed maneuvers involving left yawing turns. However, fenestron stall could also occur during right sideward flight, as well. When encountered, left yaw rate increases suddenly and dramatically and cannot be arrested by application of right pedal. It may be the case that applying right pedal to stop the left yaw may actually worsen the yaw rate, at least initially (Reference 1).

4.4 SUMMARY: LOW AIRSPEED DIRECTIONAL STABILITY RESULTS

In summary, test results and research findings show that the Gazelle's low airspeed directional stability problems occur in conditions where the fenestron experiences moderate to high relative wind from the right. The overall critical azimuth region is from approximately 045° to 135° relative. Yaw stability problems begin at about 20 KTAS for sideward flight and at 10 KTAS for low airspeed maneuvers involving left yawing turns. The Gazelle's critical azimuth region is consistent with other helicopters, allowing for the Gazelle's main rotor direction of rotation being clockwise (when viewed from above).

4.5 FENESTRONS, TAIL ROTORS, AND VORTEX RECIRCULATION

Thrust produced by a conventional tail rotor is a function of the flow rate of the air mass through the tail rotor and the change in velocity of that air mass, as presented in equation 4 below.

$$T = \left(\frac{dm}{dt}\right)\Delta V$$

Equation 4: Thrust as a Function of Air Mass Flow Rate
(Source: Reference 7, Eq 5.22, p.5.15)

Figure 2 depicts the contraction and acceleration of the airflow through the tail rotor. The induced airflow through the tail rotor reaches its maximum velocity, or full maturity, at a downstream distance of approximately 7/10 of the diameter of the tail rotor. All of the flow acceleration and resultant antitorque thrust is due to work done on the flow by the tail rotor alone.

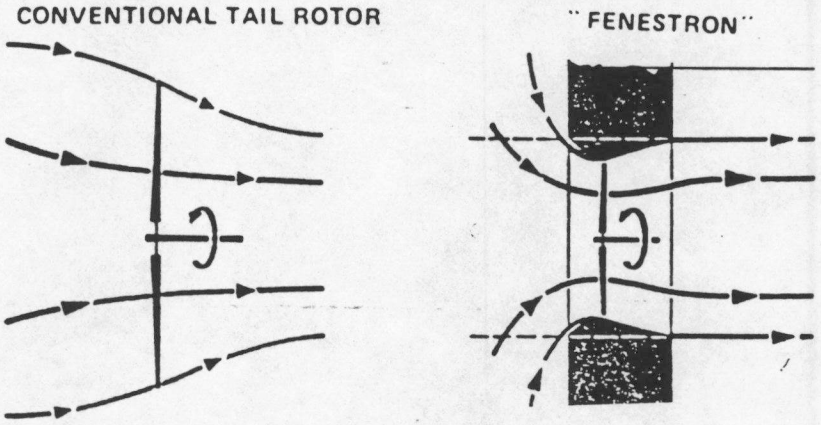


Figure 2: Air Flow Through Tail Rotor and Fenestron

Source: Rene Mouille, "Ten Years of Aerospatiale Experience With the Fenestron and Conventional Tail Rotor," Presented at the 35th Annual National Forum of the American Helicopter Society, Washington DC, May 1979, p.79-58-3.

The primary difference between the fenestron and the conventional tail rotor is the way in which the fenestron generates antitorque thrust. The fan of the fenestron and the airflow it induces comprises only about 50% of the fenestron's total antitorque thrust. The other 50% of the antitorque thrust is generated by a low pressure area at the inlet side of the fenestron duct. This is accomplished by incorporating the fenestron fan inside the fenestron duct. As the air mass is accelerated through the fenestron duct and fan by the fan, a low pressure region develops around the inlet lip of the duct. That low pressure region at the inlet is the basis for the other half of the fenestron's anti-torque thrust, as depicted in figures 3 and 4.

The unique design of the fenestron can also be more efficient than its conventional counterpart. The fenestron of a given size can produce more anti-torque thrust than a conventional tail rotor of the same size. Therefore, the fenestron designed to produce a certain thrust can be made smaller than its conventional equivalent. This fenestron could also be designed to produce lower induced velocities which reduces the induced power required to generate the desired anti-torque thrust. (Reference 8, p.54). Indeed the fenestron design is both efficient and unique. However, the fenestron's unique design characteristics are also what cause fenestron stall to be so sudden and dramatic.

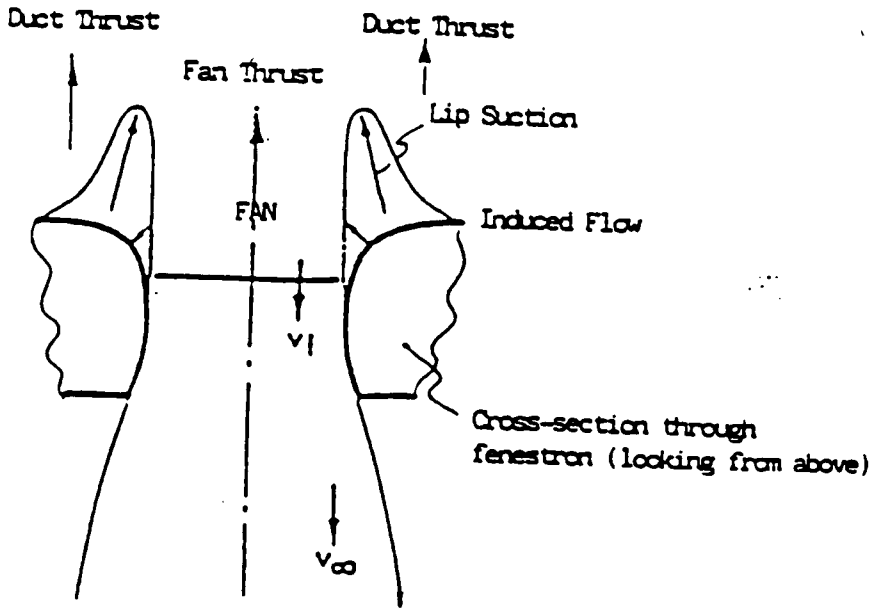


Figure 3: Thrust Mechanisms on Fenestron

Source: A. Vuillet, and F. Morelli. "New Aerodynamic Design of the Fenestron for Improved Performance." Presented at the 12th European Rotorcraft Forum. Garmisch-Partunkirchen, FRG September 1986, Annex 1.

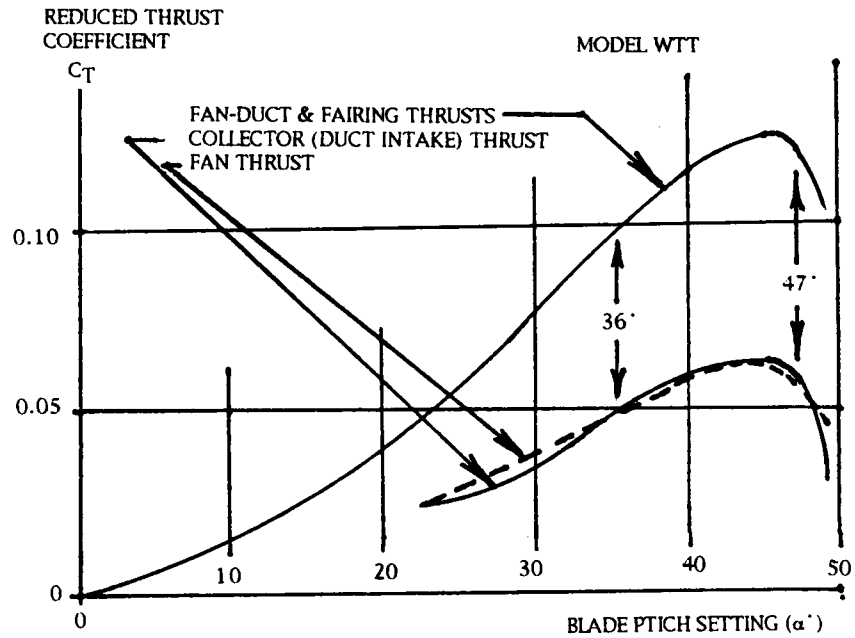


Figure 4: Thrust Sharing Between Fan and Duct Fairing

Adapted: A. Vuillet, and F. Morelli. "New Aerodynamic Design of the Fenestron for Improved Performance." Presented at the 12th European Rotorcraft Forum. Garmisch-Partunkirchen, FRG, September 1986, 16.5.

For a conventional tail rotor, vortex ring state occurs when the relative or ambient air flow around the tail rotor is approximately equal in magnitude and opposite in direction to the tail rotor induced velocity. In this condition the tail rotor tip vortices cannot move away downstream of the tail rotor disc. Consequently, the air mass being pumped, or induced through the tail rotor is trapped between the tail rotor disc and a high pressure gradient caused by the opposing flow, causing the reingestion of tail rotor tip vortices, as depicted in figures 5a through 5c. The disrupted flow associated with the vortices recirculating through the tail rotor causes local blade tip stalls and consequently great variations in the thrust of the tail rotor. As the tail rotor thrust varies so does yaw attitude and the pedal control activity required to maintain aircraft heading.

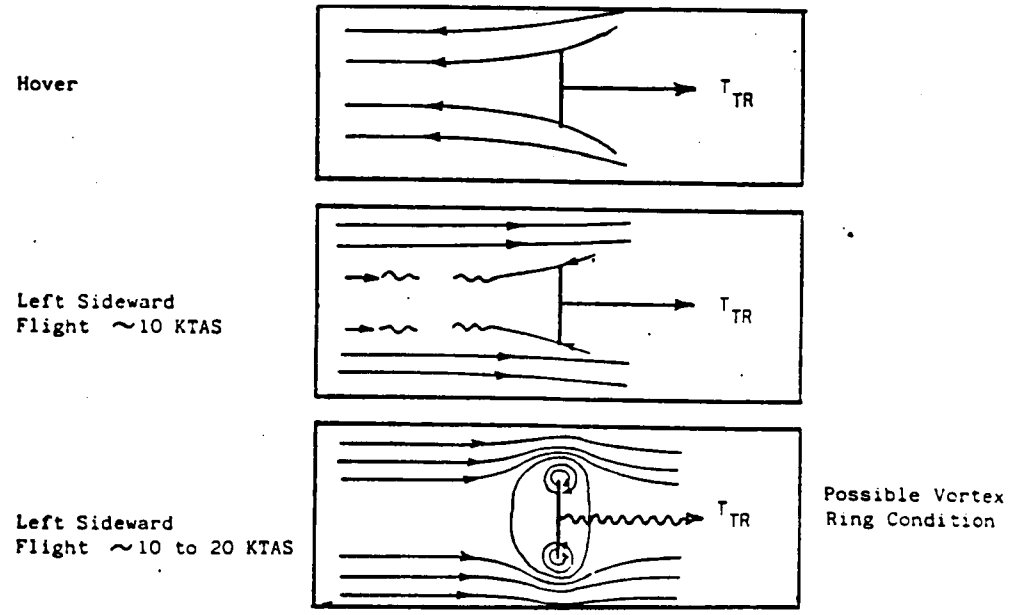


Figure 5: Air Flow - Tail Rotor Vortex Ring State

Adapted: Aircraft Research and Development Unit, *Bell 206B Directional Control in Low Airspeed Flight*. (Edinburgh, South Australia: Department of Defense Royal Australian Air Force, 1981), Appendix 1 to Annex E.

As mentioned above, conventional tail rotor thrust is a function of the mass flow rate of air moving through the tail rotor and the change in velocity of that air mass. Tail rotor thrust can also be expressed as a function of the change in pressure and the diameter of the tail rotor, as shown in the equation below:

$$T_{TR} = (\Delta P)(S) = (1/2V_\infty^2)(S)$$

Equation 5: Tail Rotor Thrust as a Function of Pressure & Rotor Diameter
(Adapted: Reference 7, Eq 5.19, p.5.13)

Where:

T_{TR}	Thrust of the tail rotor
ΔP	Change in air pressure $=(1/2\rho v_\infty^2)(S)$
ρ	Air density
S	Area of tail rotor disc
v_∞	Wake velocity; $v_\infty = 2v_i$
v_i	Induced velocity: the velocity imparted to the air mass by tail rotor disc.

When the magnitude of the opposing relative wind for a conventional tail rotor is about $\frac{1}{4}v_i$, unsteadiness in the flow and thrust begins; this is also where heading control starts to become more difficult. This flow and thrust unsteadiness peaks at approximately $\frac{3}{4}v_i$, (fully established vortex ring state), and disappears at about $1\frac{1}{4}v_i$ (Reference 9, p.12).

The recirculation situation due to vortex ring state for the Gazelle fenestron is much the same as for the

tail rotor. However, the resulting anti-torque thrust reductions are much greater because of how the fenestron produces that thrust. As discussed above, roughly half of the anti-torque thrust of the fenestron is generated by the fan, while the other half is generated by inlet lip suction. Once the fenestron fan is generating Vi, it draws air into the intake of the duct, where it undergoes an acceleration and produces a low pressure region, which in turn generates thrust. The total thrust of the fenestron is represented by the momentum theory equations below:

Equation 6a: $T_{Fenestron} = T_{Fan} + T_{Duct}$

Equation 6b: $T_{Fan} = (P_2 - P_1)(S) = (\frac{1}{2}\rho v_\infty^2)(S)$

Equation 6c: $T_{Duct} = (P_1 - P_0)(S) = (\frac{1}{2}\rho v_i^2)(S)$

Adapted: A. Vuillet, "Operational Advantages and Power Efficiency of the Fenestron as Compared to a Conventional Tail Rotor," VERTIFLITE, Jul/Aug, 1989, p.29. (Note: Vuillet's Momentum theory analysis is presented in appendix I.)

Where:

P_2	Pressure at vicinity of fenestron fan exit
P_1	Pressure at vicinity of fenestron fan face
P_0	Ambient air pressure
ρ	Air density
v_i	Induced velocity
v_∞	Wake velocity
S	Area of the fan disc
T_{Fen}	Total fenestron anti-torque thrust
T_{Fan}	Fenestron fan anti-torque thrust
T_{Duct}	Fenestron duct anti-torque thrust

As was the case with the tail rotor, when the fenestron encounters a flight condition that creates

sufficient opposing flow, recirculation of the induced air mass occurs. Again this is due to the air mass being trapped between the fan and the high pressure gradient generated by the opposing flow. There are also vortices generated by the vertical fin moving through the air mass to the right that recirculate to the inlet region of the duct.

The Gazelle's vertical tail fin, quite large for a helicopter of its size, serves well to off load the fenestron during forward flight, providing ample directional stability at airspeed. However, during right sideward flight and left yawing turns on the spot, the vertical fin generates large vortices that recirculate through the fenestron. Additionally, recirculation is likely occurring from the upwind vertical and horizontal stabilizers and shed vortices from the tips of the main rotor blades (Reference 1) Since the fenestron has two means of producing anti-torque thrust, the fan and the inlet lips, both means will be analyzed to determine how they are affected during recirculation of vortices.

To produce a left yawing turn in the Gazelle, the pilot increases left pedal input, which reduces fenestron blade pitch, which in turn reduces fenestron thrust, allowing the torque of the main rotor to turn the nose of the aircraft to the left. As left yaw rates increase the opposing relative flow from the right increases. When the opposing relative flow from the right is approximately equal and opposite to the induced velocity from the left, several things happen.

First, a strong high pressure region develops downstream (to the right) of the fenestron fan. This high pressure region eventually becomes too strong for the induced air flow of the fenestron to push through. When this occurs recirculation of the induced air flow

from the fan begins, as presented in figure 6. Of course recirculation of the other sources of vortices may already be occurring. When developed, the recirculation of vortices disrupts the angle of attack of the fan blade tips, eventually causing them to stall.

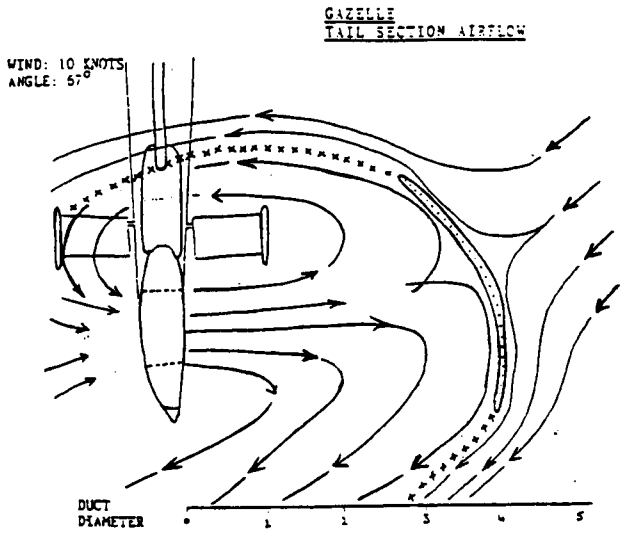


Figure 6: Airflow: Fenestron Vortex Ring State

Source: "Fenestron Stall," Helicopter Aerodynamics Lecture Notes, United States Naval Test Pilot School, p. 12.

As the blade tips approach stall, v_i and fan thrust continue to decrement causing faster left yaw rates, greater opposing flow velocities, and stronger and closer adverse pressure gradients, as depicted in figure 7. All of this results in a deeper state of recirculation, and fenestron stall. Ultimately, when the fan blade tips exceed $C_{l_{max}}$ they can no longer supply head pressure to activate the flow across the inlet tips. (Reference 10, p.16.6) Consequently, the velocity profile across the inlet lips is disrupted, as

is the inlet low pressure area. Consequently, much of the anti-torque thrust generated by this low pressure area (T_{Duct}) is lost. Additionally, about half of the fan thrust (T_{Fan}) is lost in this condition as well. This accounts for nearly 75% of the fenestron's total anti-torque thrust. In summary, fenestron stall is self-sustaining (divergent), develops rapidly, and results in nearly a 75% loss of the fenestron's total antitorque thrust. It is clear why fenestron stall is so sudden and dramatic.

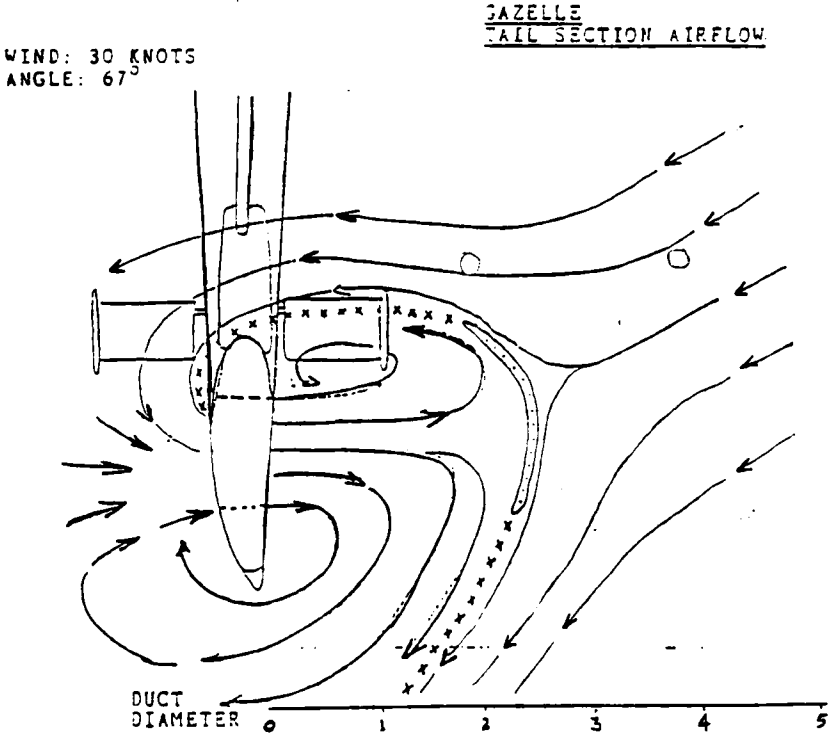


Figure 7: Airflow: Advanced Fenestron Votex Ring State

Source: "Fenestron Stall," Helicopter Aerodynamics Lecture Notes, United States Naval Test Pilot School, p.13.

However, another potential cause of fenestron stall is the loss of fenestron thrust due to disruption of the low pressure region at the inlet lips, without significant corresponding fan blade tip stall. This situation is characterized by the disruption of flow at

the inlet lip alone. In reality, disruption of the inlet flow in isolation from the stalling of the fan blade tips is unlikely. However, it is feasible that the fenestron thrust due to the low pressure gradient at the lips could be interrupted to a great degree before any significant fan stall occurred. In essence, the duct inlet region experiences its own local adverse pressure gradient which leads to a reduction in Δv, ΔP, and consequently thrust, across the inlet lips. As the vortices shed from the various tail control surfaces are erratic, the disruption of flow across the lips is also likely to be erratic, causing erratic and unpredictable heading perturbations that complicate the pilot's workload to maintain heading.

Eventually, the condition worsens to the point that inlet lip flow separates, the low pressure region is lost, and nearly all of the resultant anti-torque thrust is negated, which again is approximately 50% of the fenestron's total thrust. Considering the high disc loading on the fenestron, the consequence is that the fenestron could stall just from losing inlet thrust alone. In reality, all of these effects are probably occurring together and in rapid succession, quickly leading to a complete stalling of the fenestron.

4.6 IMPACT ON FLYING QUALITIES

With regard to handling qualities, it is easy to see why pilot workload associated with maintaining heading during right sideward flight increases so dramatically. As vortices build and shed from the fin and other tail surfaces at irregular intervals, the corresponding fenestron thrust levels fluctuate significantly and irregularly. Additionally, the weather cocking characteristics of the Gazelle are strong due to its

relatively large vertical fin. All else being equal, the stronger the weather cocking characteristics, the smaller the remaining pedal margins. The pedal margins are reduced further when coupled with dynamic pedal control inputs made to correct for moderately large, rapid, and unpredictable heading excursions. The combined affect of these two factors reduce the remaining pedal control margins below minimum acceptable levels.

For the Gazelle pilot experiencing fenestron stall, the problem is exacerbated when he begins to get into the loop to correct the stall. The pilot's natural reaction to the undesirable high left yaw rate is to add right pedal. However, adding right pedal further increases the pitch on the already stalled fenestron blade tips, driving them deeper into stall, thereby further reducing tail rotor thrust. This sequence increases left yaw rate, yielding an apparent directional control reversal, wherein adding right pedal actually increases the left yaw rate instead of damping it (Reference 1).

In summary, the yaw problems for the Gazelle result from the recirculation of vortices into the fenestron. Recirculation normally occurs for the Gazelle during right sideward flight and low airspeed maneuvers involving left yawing turns. Heading control difficulties due to yaw instability during right sideward flight within the critical azimuth region begin at approximately 20 KTAS. The excessively high left yaw rates associated with fenestron stall develop during low airspeed maneuvers involving left yawing turns, with as little as 10 knots of relative wind, from 045°-090° relative.

4.7 ANALYSIS OF POTENTIAL LOW AIRSPEED DIRECTIONAL STABILITY IMPROVEMENTS

4.7.1 GENERAL

Fenestron modifications to mitigate Gazelle directional control deficiencies are not likely to eliminate them entirely. Rather, the goal is to delay the onset of vortice recirculation that leads to fenestron stall and the handling qualities difficulties associated with right sideward flight, to a higher relative wind speed. In so doing, it is hoped that the Gazelle's current low airspeed envelope will be made safer for more aggressive handling, and that flying qualities and control margins in these conditions are improved as well.

4.7.2 FENESTRON FAN DIRECTION OF ROTATION

One engineering modification that would improve the Gazelle's fenestron authority or effectiveness would be to change the direction of rotation of the fan. It is now known that a bottom blade forward direction of rotation is optimum for tail rotors in general. One benefit of bottom blade forward rotation is that it takes advantage of the main rotor vortices flowing over the forward portion of the fan, which increases the relative wind velocity on the advancing blades of the fan, as depicted in figure 8. A similar advantage is garnered in rearward flight when the ground vortex flows over and influences the fenestron. Because the direction of rotation of the ground vortex opposes the fan, the relative velocity seen by the leading edge of the fan blades is higher.

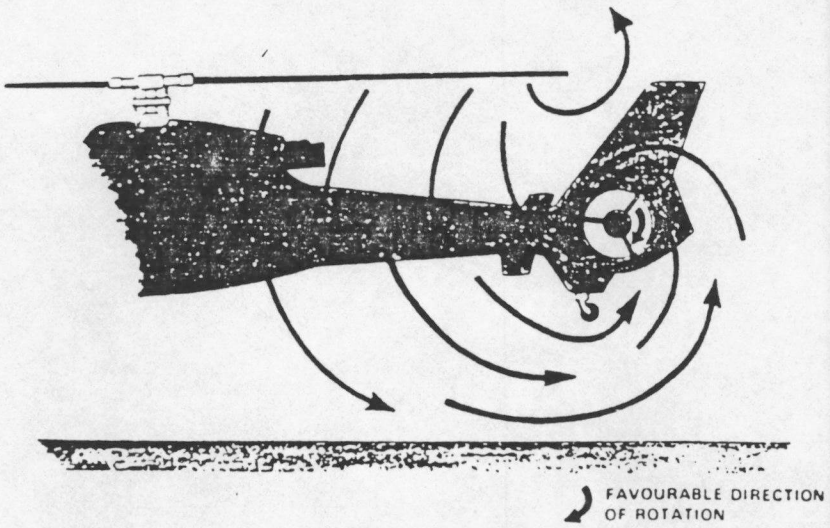


Figure 8: Interaction: Main Rotor Vortex and Fenestron

Source: Rene Mouille, "Ten Years of Aerospatiale Experience with the Fenestron and Conventional Tail Rotor," Presented at the 35th Annual National Forum of the American Helicopter Society, Washington, May 1979, p.79-58-4.

The Aerospatiale Dauphin provides evidence that changing the direction of rotation of the fenestron fan works. In changing the direction of rotation on the Dauphin's fenestron fan to bottom blade forward, the higher pedal displacements required in ground effect (as they were with the Gazelle) were neutralized (Reference 11, p.79-58-4).

Wind tunnel testing conducted by Boeing engineers also supports a bottom blade forward direction of rotation. Wind tunnel tests of a model helicopter rig with an 8' diameter main rotor and 1.57' diameter tail rotor were conducted by Boeing Vertol to investigate various characteristics of tail rotor performance in and out of the presence of the main rotor (Reference 12, p.2). It was demonstrated that main rotor vortices positively increase the tail rotor thrust output. Figure 9 shows the higher thrust generated in the presence of the main rotor.

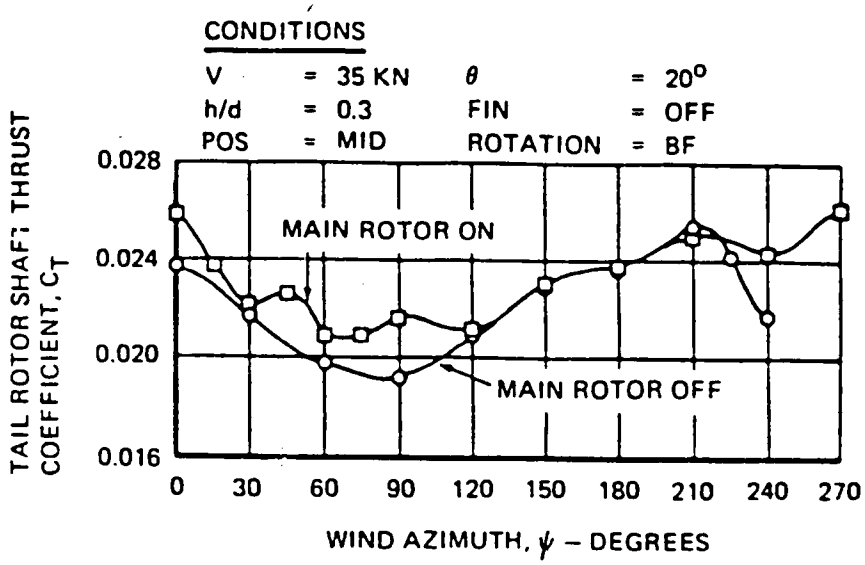


Figure 9: Effect of Main Rotor on Tail Rotor Thrust

Source: Wayne Wiesner and Gary Kohler "Tail Rotor Performance in Presence of Main Rotor, Ground, and Winds," Presented at the 29th Annual National Forum of the American Helicopter Society, May 1973, p. 6.

Additionally, figures 10a and 10b show the net tail rotor thrust coefficient for the bottom forward rotation was higher for all wind azimuths than it was for the bottom aft rotation, for both 20 and 35 knots (Reference 12, pp.5-6).

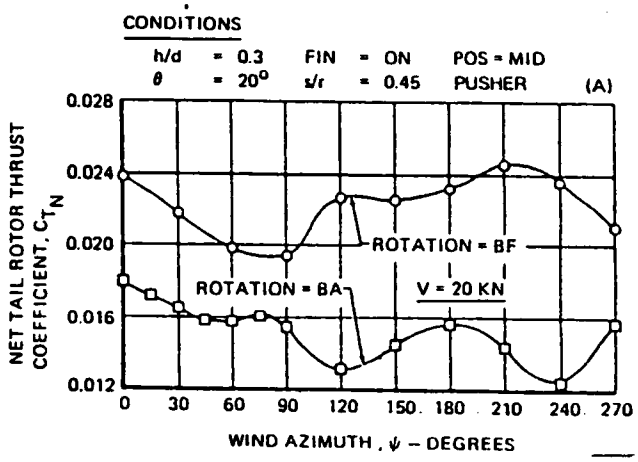


Figure 10a: 20 knots

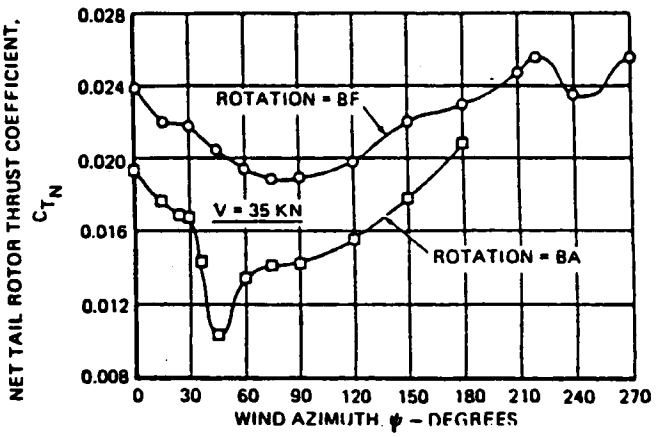


Figure 10b: 35 knots

Figure 10: Effect of Tail Rotor Direction of Rotation on Tail Rotor Thrust, 20 and 35 knots

Source: Wayne Wiesner and Gary Kohler "Tail Rotor Performance in Presence of Main Rotor, Ground, and Winds," Presented at the 29th Annual National Forum of the American Helicopter Society, May 1973, p.6.

4.7.3 FENESTRON FAN BLADES

Another modification worth pursuing is the incorporation of new fan blades. Aerospatiale explored different blade types for the Panther fenestron, investigating blades with different planforms, twist, and camber. The result was development of a family of blades with spanwise variable thickness, capable of taking increased load at the tip. The resulting benefit of these blades was increased $C_{l_{max}}$ at the tips which delayed blade tip stall and maximized the depressure on the inlet lip area of the duct (Reference 10, p.16.8).

4.7.4 FENESTRON DUCT GEOMETRY

Another fenestron modification that potentially improves directional control is a change in fenestron duct geometry. For starters, fenestron performance is a function of diameter. As the diameter increases, the power required to drive the fenestron decreases. Aerospatiale increased the Dauphin's hover gross weight by 260 pounds by increasing the diameter of its fenestron fan from 35" to 43" (Reference 13, p.16).

The fenestron is more efficient and requires less power to generate the same anti-torque thrust as a conventional tail rotor. The efficiency stems from the power required to generate the anti-torque thrust, the induced power. Induced power is a function of thrust and induced velocity. The fenestron can generate the same thrust as a conventional tail rotor with slower induced velocities because only half of the total anti-torque thrust comes from the acceleration of the air mass through the disc. The fenestron is estimated to produce the same thrust as a conventional tail rotor that is 30-50% larger (Reference 13, p.16). The problem with the Gazelle fenestron is that, at 27" in diameter, it is less than 50% of the size of its would-be conventional equivalent (Reference 13, p.16).

In fact both the Gazelle and the Dauphin fenestrons are small by comparison to other ducted fan anti-torque devices. For example, the Fantail tested on the Sikorsky H-76, (10,500lb takeoff weight) is 4 feet in diameter (Reference 14, p.3). The consequence of the fenestron being smaller than an equivalent tail rotor is that the Gazelle's fenestron requires more engine power to drive it for an equivalent thrust output.

Another geometric modification of the fenestron involves increasing the depth of the fenestron duct.

The Gazelle duct depth is approximately 12 1/2", or about 40% or so of its diameter of 27.2" (Reference 1). R.W Prouty asserts that the depth of the duct should be at least 20% of its diameter (Reference 13, p.16). Given the current high ratio of duct depth to duct diameter, the additional drag created by a wider duct in forward flight weighs against this option, at least until other options have first been investigated.

4.7.5 FENESTRON STATORS

Another potential modification to improve the yaw characteristics of the Gazelle is to add stators in the fenestron duct downstream of the fan to recover the rotational energy of the induced flow. Straightening the flow increases the induced velocity of the fenestron, as depicted in figures 11 and 12.

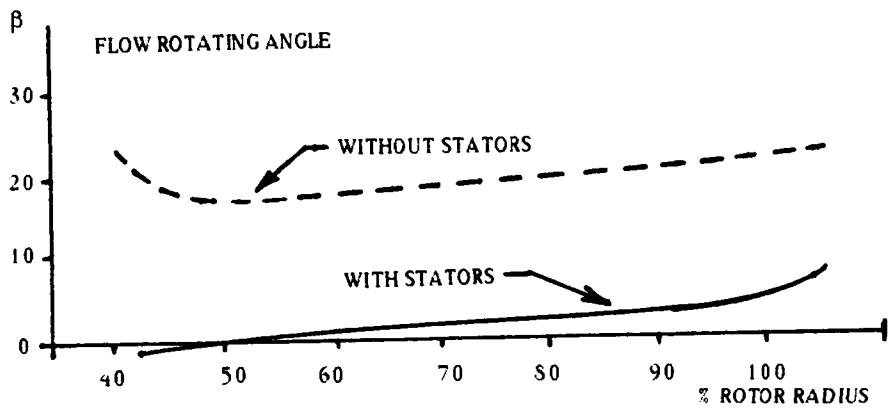


Figure 11: Influence of Stator Blades on Flow Rotating Angle at the Diffuser Exit

Adapted: A. Vuillet and F. Morelli, *New Aerodynamic Design of the Fenestron For Improved Performance*, Presented at the 12th European Rotorcraft Forum, Garmisch-Partunkirchen, FRC, September, 1986, p.16.8.

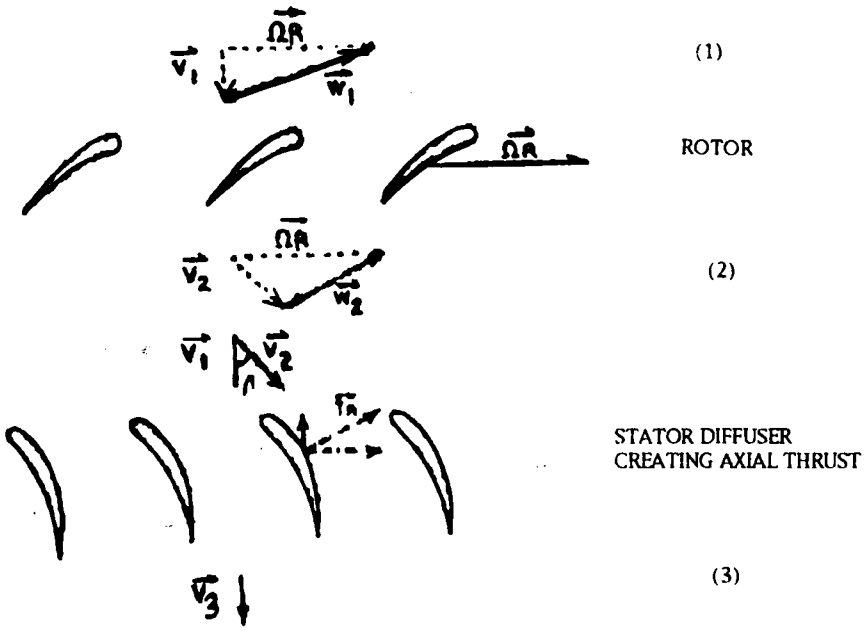


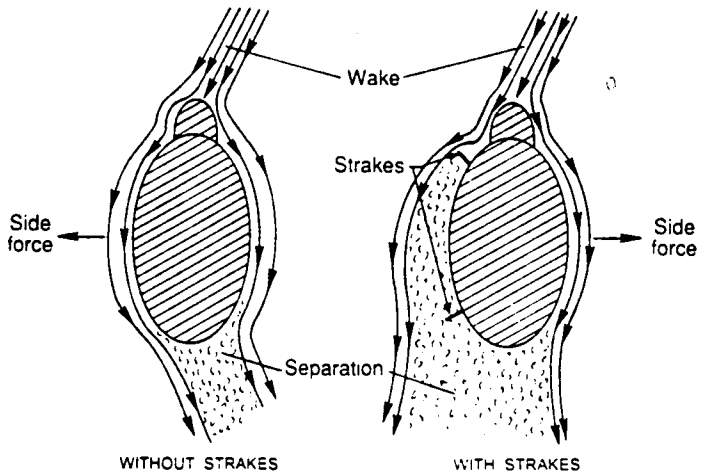
Figure 12: Influence of Stator Blades on Axial Thrust

Adapted: A. Vuillet and F. Morelli, *New Aerodynamic Design of the Fenestron For Improved Performance,* Presented at the 12th European Rotorcraft Forum, Garmisch-Partunkirchen, FRC, September, 1986, 16.8.

Stators were added to the Dauphin for just this reason, but had the added benefit of allowing a shallower depth duct, which in turn reduced parasitic drag in forward flight (Reference 9, p.26) Stators are also incorporated on Eurocopter's EC-135 to improve fenestron performance. Additionally, the EC-135's stators interact with asymmetrically spaced rotor blades to reduce noise signature through frequency modulation (Reference 15, p.29). However, adding stators to the duct likely requires an effort similar to changing duct geometry. Consequently, for reasons of cost and complexity, this is an unlikely place to start for improving the Gazelle.

4.7.6 TAILBOOM STRAKES

Yet another modification to improve yaw stability might be to add strakes to the tail boom of the Gazelle. This relatively simple concept takes advantage of the main rotor wake that flows over the tailboom in low airspeed flight. As seen in figure 13, the resultant flow over the tail boom induces a low pressure on both sides of the boom. In right sideward flight without tailboom strakes, the difference in low pressure regions on both sides of the boom works to worsen the fenestron effectiveness, by creating a side force to the left. Attaching strakes to the upper and lower left sides of the tail boom reverses the direction of the side force by reversing the side with the lower pressure region. The resultant effect is to increase the left pedal control margin remaining in right sideward flight.



(View Looking Forward)

Figure 13: Influence of Strakes on Flow Patterns Around Tail Boom in Right Sideward Flight

Source: John C. Wilson and Henry L. Kelley "The Strake: A Simple Means for Directional Control Improvement." VERTIFLITE, Volume 39, No.2, March/April 1993, 29.

In 1986, a low airspeed flight test of the Bell 204B was conducted at the NASA Wallops Flight Facility. At the critical azimuth of approximately 060° relative, with a relative wind speed of 20 knots, the mean left pedal margin remaining was 8%. Adding strakes increased mean left pedal margin remaining to 15% (Reference 16, pp.30-31). Other aircraft, such as the Bell OH-58A, only gained about a 5% improvement in pedal margin remaining at the critical azimuth. Although not as large an improvement, there was still a "modest" corresponding improvement in pilot workload to maintain heading (Reference 16, pp.30-31).

The side force created on the tail boom with strakes is a function of main rotor wake speed, the diameter of the tail boom, and the size of the strake. Main rotor induced velocity is a function of aircraft weight and rotor disc diameter, as seen from the following equation: $v_i = (\frac{W}{2\rho A_d})^{1/2}$ (Reference 17, p.5.VI). The Gazelle's maximum gross weight (3970 lbs), main rotor diameter, and tail boom diameter are very similar in scale to the OH-58A/C. Therefore, it is possible that any improvement to the Gazelle's yaw stability in right sideward flight due to addition of tail boom strakes would be of a similarly modest order of magnitude.

Nevertheless, a 5% gain in pedal margin and a slight improvement in pilot workload stills helps to mitigate the problem. Considering the relative simplicity and low cost associated with adding strakes to the Gazelle's tail boom, even a small improvement such as this is probably worth pursuing.

4.7.7 VORTEX GENERATORS

Another option, specifically targeted at mitigating fenestron stall by delaying the disruption of the low pressure region at the inlet lips, is to add vortex generators to the duct inlet. As discussed earlier, about 50% of the fenestron's total anti-torque thrust, that thrust due to the low pressure area generated at the inlet lip, is lost during fenestron stall due to the disruption of that low pressure region during the recirculation of vortices. In one scenario discussed, the vortices recirculating around the vertical fin and back into the inlet create an adverse local pressure gradient that causes the flow across the inlet lips to separate. In another scenario discussed, the recirculating vortices were causing the fan blade tips to exceed $C_{l_{max}}$, thereby causing a loss of head pressure which disrupted the flow across the inlet. In either case, cleaner, attached flow across the inlet lip surface and into the tip region of the fan blades at the inlet is required. The goal is to delay flow separation during conditions of vortex recirculation through use of low-profile, Wheeler vortex generators.

The Wheeler concept for boundary layer control through vortex generators is to incorporate submerged triangular channels into the flow-control surface. These channels have sharp upper edges to generate vortices, divergent side walls normal to the flow control surface, and floor contours arranged to conduct the streamwise flow of the vortices below the level of the local flow control surface. As depicted in figure 14, when the submerged channels are nested together in an overlapping manner, a streamwise cascade is generated that allows the formation of new vortices downstream to replace upstream vortices on the verge of bursting due to strong adverse

pressure gradients (Reference 18, p.4). In essence, the Wheeler vortex generators straighten the airflow by introducing precisely shaped, controlled vortex patterns (Reference 19, p.88).

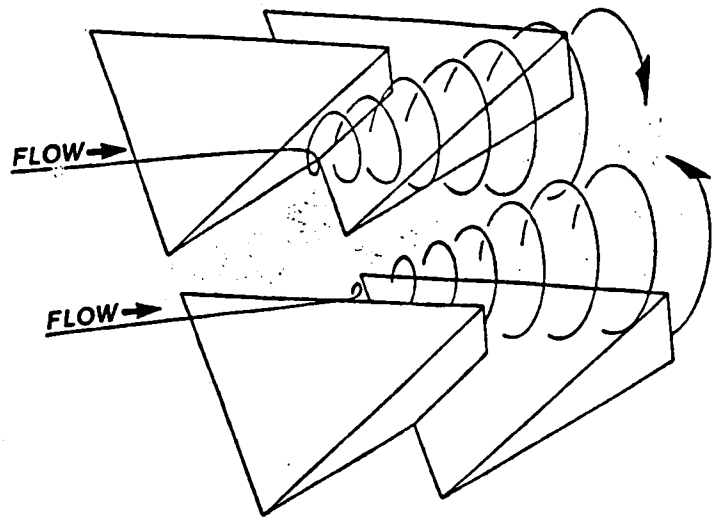


Figure 14: Wheeler Submerged Channel Vortex Generators

Source: "Low-Profile Vortex Generators Provide Aerodynamic Improvements Through Boundary Layer Control." Aviation Week and Space Technology, 1 June 1987, 90.

A unique advantage of Wheeler submerged vortex generators is that they maintain attached flow in adverse pressure gradient regions without generating large amounts of adverse drag. Resulting drag is minimized by thinning the boundary layer through "dynamic turbulent mixing between vortices, boundary layer, and the free-stream fluid constrained within the submerged channel." (Reference 18, p.3). Additionally, when incorporated in nested series, or cascades, these submerged channels can generate the same boundary layer control as a larger vane-type vortex generator (Reference 19, p.90).

Wheeler vortex generators are in use on the Cessna Caravan 208B. Thin panels of Wheeler vortex generators are installed on the outboard leading edge of the flaps of the 208B to prevent premature separation of the airflow just prior to wing stall with the flaps extended. The Wheeler panels were selected by Cessna because they were thin enough to fit into the small space between the flap and the wing when the flaps were retracted. The Wheeler vortex generators in part, enabled the Caravan's maximum gross weight to increase by 500 lbs with negligible increase in stall speed (Reference 19, p.90).

The application for the Gazelle fenestron would be to bond pliable panels of Wheeler vortex generators in overlapping, nested cascades on the surface of the inlet section of the duct. Testing is required to determine the optimum location for these panels of vortex generators and the percentage of inlet area to be covered. To reiterate, the goal is to delay the onset of fenestron stall by delaying the separation of flow across the inlet and also by delaying fan blade tip stall. Additionally, it is anticipated that fenestron thrust will be stronger and more consistent at higher relative windspeeds in right sideward flight, thereby reducing the magnitude and unpredictability of aircraft heading excursions.

4.7.8 PILOT COMPENSATION

Two final potential modifications lean toward improving the pilot's ability to compensate for fenestron stall rather than delaying its onset. The first is to review the pilot training program to heighten awareness and understanding of the flight conditions under which the Gazelle might experience yaw

instability and fenestron stall. Certainly this type of training already exists and has value as a tool of prevention. This is also a relatively inexpensive approach, but pilots may still be required to compensate for an aircraft deficiency in flight if the prevention methods come up short.

The other modification geared toward enhancing pilot compensation, or reaction to fenestron stall, is a stall warning device. Many fixed wing aircraft incorporate metal flaps in the wing leading edge that are blown back during normal angles of attack. When the angle of attack approaches a stall condition the metal flap flaps forward and makes electrical contact that activates an audio warning to the pilot. One or more of this type of stall warning device might be fitted to the inlet area of the fenestron duct to indicate the onset of recirculating vortices and subsequent fenestron stall (Reference 20, p.27).

4.8 SUMMARY OF POTENTIAL LOW AIRSPEED DIRECTIONAL STABILITY IMPROVEMENTS

Each of the proposed modifications is feasible, though some are intuitively more complicated and/or costly than others. Adding strakes to the tail boom is a relatively straight forward approach to improve pedal control margins in right sideward flight. Additionally, the cost of adding and testing tailboom strakes is probably relatively low, making tailboom strakes a preferred option for increasing directional control margins in right sideward flight.

Adding vortex generators is probably easy in terms of bonding the actual panels onto the duct. However, the cost and effort to determine the optimum configuration, location, and percentage of duct coverage required may

not be trivial. Nevertheless, this is a good option for delaying the onset of fenestron stall. As with tailboom strakes, an advantage of the Wheeler vortex generators is that they are relatively straightforward modifications to the aircraft.

Changing the fenestron duct geometry, either by increasing its diameter or depth, involves a detailed redesign, a significant retrofitting effort, and like all the other options, a significant test program to validate the change. For these reasons, changing diameter or depth are best pursued once other cheaper options are exhausted. For the same reasoning, retrofitting with stators is not recommended as an initial course of action.

Reversing the direction of rotation of the fenestron fan has demonstrated potential, although it too is not trivial. This modification likely requires a new fenestron gear box and fan blades. However, in view of the dramatic improvements gained by the Dauphin and other helicopters by applying the bottom blade forward direction of rotation and new fan blades with higher values of $C_{l_{max}}$, this option merits further investigation.

Finally, adding some sort of stall warning device seems simple enough at a glance, but is likely to be quite difficult in practice. Finding the optimum location or locations for the devices could prove elusive. Achieving accurate and reliable indications without false indications during all maneuvers and conditions might require an extensive and expensive test program.

In summary, the order in which modifications to the fenestron should be approached is to first pursue tailboom strakes and inlet vortex generators, and stall warning indicators. If these prove insufficient,

reversing the direction of fan rotation, and incorporating new, high $C_{l_{max}}$ blades is the next recommended option, followed by an increase in duct diameter and the addition of stators. In any event, the pilot training program should also be reassessed for potential enhancement.

SECTION 5: JACK STALL

5.1 GENERAL

Another undesirable deficiency discovered through the course of this trial was jack stall. Jack stall is the term used for a flight condition where the Gazelle hydraulic system is overloaded by aerodynamic forces being fed backward through the flight control system. When those forces are strong enough to overload the hydraulic system they are also strong enough to prevent the pilot from making control inputs, resulting in temporary loss of aircraft control.

5.2 ANALYSIS

Jack stall is most likely to occur in forward flight dives at high airspeeds involving an aggressive turning pull out, or during any other flight maneuver that generates high aerodynamic loads on the rotor system. These aerodynamic loads saturate the 590 psi hydraulic system. The result is that the pilot is unable to make control inputs to the rotor sytem. In other words, the aircraft is temporarily out of control. Recovery involves waiting until the aerodynamic forces on the rotor system subside then making whatever control inputs are necessary to regain controlled flight.

5.3 POTENTIAL IMPROVEMENTS FOR JACK STALL

The way to mitigate the effects of jack stall are to train pilots how to avoid getting into the condition in the first place, as well as how to get out of it once it is encountered. The way to correct the jack stall problem is to install a higher pressure hydraulic system

capable of handling the aerodynamic loads that can be generated by the Gazelle in aggressive pull outs from high speed dives.

While training is the less expensive approach, a greater capacity hydraulic system would not only eliminate the problem, it would expand the operational envelope of the aircraft, allowing it to be flown more aggressively. Though there is no doubt some associated aircraft weight gain involved with installing a larger hydraulic system the trade off is worth the investment.

SECTION 6: SUMMARY AND RECOMMENDATIONS

6.1 SUMMARY AND RECOMMENDATIONS

Regardless of the type and magnitude of modifications made to the Gazelle to improve maneuvering stability and low airspeed directional control, an extensive ground and flight test program is required to revalidate all the Gazelle's flying qualities. Any change to the CG to improve maneuvering stability will impact the longitudinal control margin available during low airspeed flight. Adding tip weights increases the rotor's rotational moment of inertia and will affect rotor speed during autorotative flight. Higher inertia rotor systems decelerate more slowly after an engine failure, but they also acelerate more slowly which could be a problem if rotor speed is allowed to get too low in the first place. Changing the horizontal tail will surely affect forward flight static longitudinal stability as well. Changes to the stability augmentation system could result in unsatisfactory pilot control during a SAS hardover. Finally, care should be taken when increasing the control forces of the force feel system to ensure that flying qualtities are not negatively impacted.

Improving the Gazelle's directional stability deficiency and mitigating and delaying the onset of fenestron stall by the various modifications discussed will also require test programs. Verifying that the modification worked is only part of the challenge. Ensuring nothing else was negatively impacted by the modification is perhaps more challenging.

Finally, correcting the jack stall problem seems to be a much better way to go than to merely train pilots to avoid the flight conditions that lead to it. A training

program alone is far less expensive until a flight mishap, however improbable, occurs.

It is important to remember that any modifications made to improve forward flight maneuvering stability and low airspeed directional stability will affect other flying qualities parameters, perhaps positively, perhaps negatively. For maneuvering stability, the goal is to increase the gradient of δ_{long} vs. n_z. When this gradient is increased effectively, the Gazelle pilot will be able to execute aggressive pullups and turns at and above 2.0g, without concern of over accelerating the aircraft and without excessive pilot workload to maintain the desired pitch attitude. With respect to low airspeed directional stability and fenestron stall, the goal is to delay the onset of recirculation of vortices into the inlet duct to higher relative wind speeds. Doing so increases the safe right sideward flight limit and increases the level of aggressiveness with which the Gazelle pilot can execute low airspeed maneuvers involving left pedal turns or right sideward flight. The courses of action recommended in this paper to modify the Gazelle should go a long way to accomplishing these goals.

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APPENDIXES

APPENDIX A: GLOSSARY OF ABBREVIATIONS & ACRONYMS

ABBREVIATIONS & ACRONYMS

AGL	-	Above Ground Level
a_{1s}	-	Longitudinal flapping angle
b	-	Number of main rotor blades
C	-	Celsius
CG	-	Center of Gravity
e	-	Offset flapping hinge
ETPS	-	Empire Test Pilot School
FTM	-	Flight Test Manual
g	-	Acceleration of gravity
h	-	Height of hub above CG
h'	-	Longitudinal distance between the rotor shaft and the CG
H	-	Rotor hub force, perpendicular to shaft
HQR	-	Handling Quality Rating
I_{yy}	-	Pitch moment of inertia
IPS	-	Intermediate Pitch Stop (collective)
KCAS	-	Knots Calibrated Airspeed
kg	-	Kilograms
KIAS	-	Knots Indicated Airspeed
KTAS	-	Knots True Airspeed
lbs	-	Pounds
$M_{CGf\&t}$	-	Pitch moment due to the aerodynamic forces on the fuselage and tail
$M_{\delta_{long}}$	-	Longitudinal Control Sensitivity
M_H	-	Pitch moment due to rotor hub force
MSL	-	Mean Sea Level
M_S	-	Blade mass moment
M_q	-	Pitch Rate Damping Derivative
M_w	-	Angle of Attack Stability Derivative
N_R	-	Main rotor speed
n_z	-	Load factor
OAT	-	Outside air temperature
P_0	-	Ambient air pressure
P_1	-	Air pressure at fenestron inlet
P_2	-	Air pressure at fenestron exit
ρ(Rho)	-	Density
RPM	-	Rotations Per Minute
SAS	-	Stability Augmentation System
S	-	Area of the rotor disc
T_{DUCT}	-	Anti-torque thrust due to the fenestron duct

T_{FAN} - Anti-torque thrust due to the fenestron fan
T_{FEN} - Total fenestron anti-torque thrust
USNTPS - United States Naval Test Pilot School
VAR - Vibration Assessment Rating
V_i - Induced velocity
V_∞ - Wake velocity
V_w - Wake velocity
VMC - Visual Meteorological Conditions
VNE - Never Exceed Airspeed
Vrange - Best Range Airspeed
WT - Weight
Ω - Main rotor speed
z_w - Vertical Height Damping Derivative

APPENDIX B: DEFINITIONS OF DEFICIENCIES

APPENDIX B: DEFINITIONS OF DEFICIENCIES

Part I indicates a deficiency, the correction of which is necessary because it adversely affects:

(a) Airworthiness of the aircraft.

(b) The ability of the aircraft to accomplish its primary or secondary mission.

(c) The effectiveness of the crew as an essential subsystem.

(d) The safety of the crew or an essential subsystem.

Part II indicates a deficiency of lesser severity than a Part I which does not substantially reduce the ability of the aircraft to accomplish its primary or secondary mission, but the correction of which will result in significant improvement in the operational cost, effectiveness, reliability, maintainability, or safety of the aircraft.

Part III indicates a deficiency which is minor or slightly unpleasant and appears too impractical or uneconomical to correct in this model, but should be avoided in future designs.

APPENDIX C: PILOT RATING SCALES

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APPENDIX C: FLYING QUALITIES RATING SCALES

C.1 COOPER-HARPER HANDLING QUALITIES RATING SCALE

Various mission-type tasks were also flown to assess flying qualities, or in flight handling qualities. Handling qualities ratings (HQRs) were assigned using the Cooper-Harper Rating Scale as presented in enclosure (1), figure-1. The Cooper-Harper rating scale is a condition-based flow chart used to determine the adequacy of an air vehicle's flying qualities for a specific task. The air vehicle is evaluated against specific performance criteria and the degree of pilot compensation required to achieve that criteria. While conducting a flight task, the test pilot attempts to achieve the desired performance level for the task using whatever compensation is necessary while continuing to safely operate the aircraft. An example of a set of desired criteria for a specified flight task might be to maintain a hover position within 1' of desired altitude and 1' of desired lateral and longitudinal displacement.

If the desired performance level is easily obtained then the pilot uses less compensation, reducing the numerical value of the handling qualities rating (HQR). If the desired performance level is not achievable with maximum pilot compensation, then the pilot must attempt to achieve adequate performance for the task. Maintaining a hover position within 3' of desired altitude and 3' of desired lateral and longitudinal displacement might represent adequate performance in this example.

The threshold of maximum pilot compensation varies based on the flight task and to some degree the individual evaluation pilot. The pilot must continue to operate the aircraft safely at all times. Additionally, some auxiliary tasks cannot be ignored to accomplish the flight task being evaluated. Whatever the performance level finally achieved there will be some degree of associated pilot compensation and total workload. It is a comparison of these factors that is the basis for evaluating an air vehicle's flying qualities. It is the Cooper-Harper HQR scale that standardizes this process. The bottom line is, the lower the HQR value the better the aircraft handled for that specific task; an HQR 1 is the best while an HQR 10 is the worst.

C.2 VIBRATION ANALYSIS RATING SCALE

Aircraft vibrations were assessed using the Vibration Analysis Rating (VAR) Scale presented in enclosure (1), figure-2.

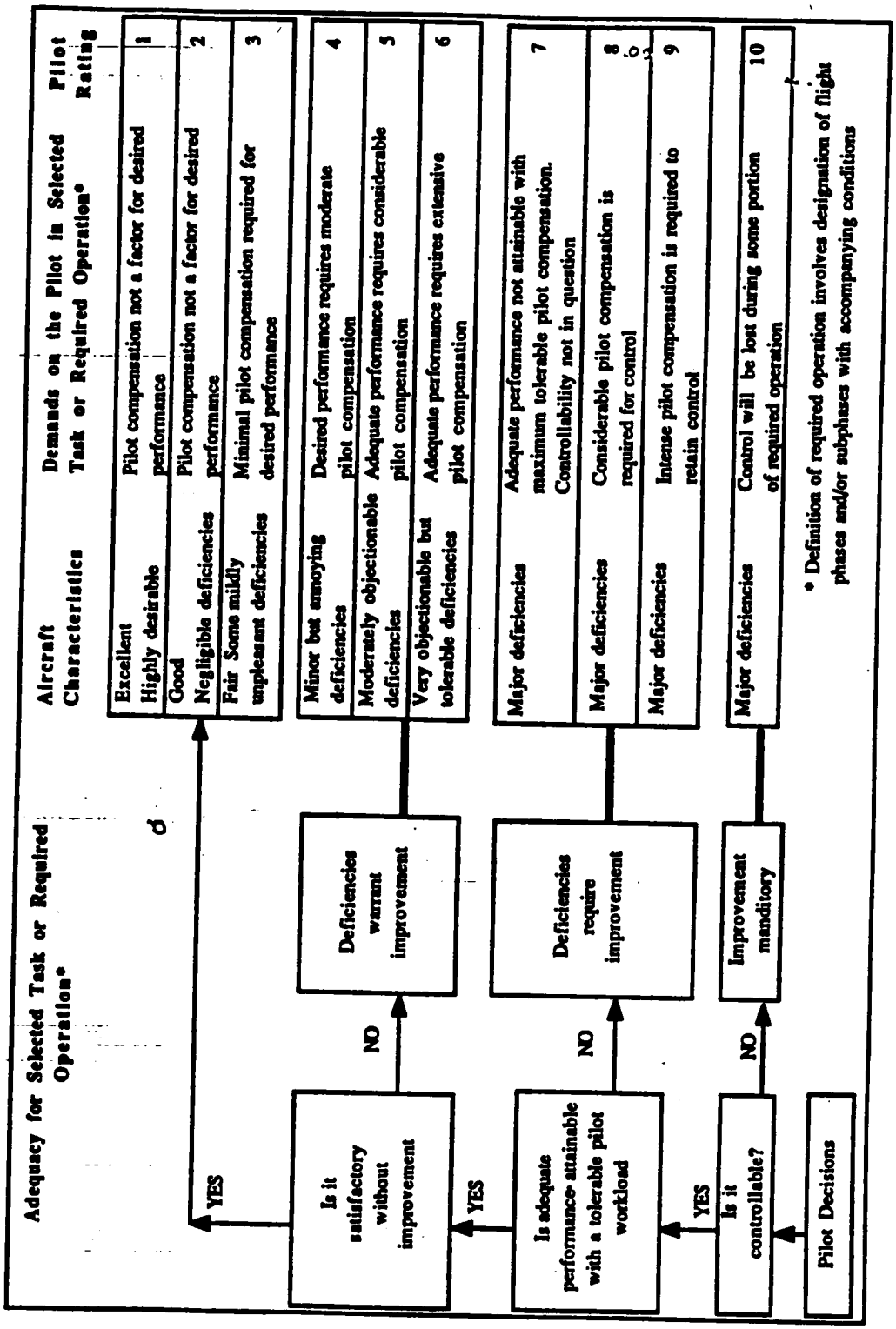


FIGURE 1

HANDLING QUALTIES RATING SCALE

APPENDIX C
ENCLOSURE (1)

Degree of Vibration	Description	Pilot Rating
No Vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1 2 3
Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.	4 5 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can be done only with difficulty.	7 8 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration.	10

FIGURE 2

VIBRATION ANALYSIS RATING SCALE

APPENDIX C
ENCLOSURE (1)

APPENDIX D: GAZELLE HT MK-3 THREE-WAY VIEW

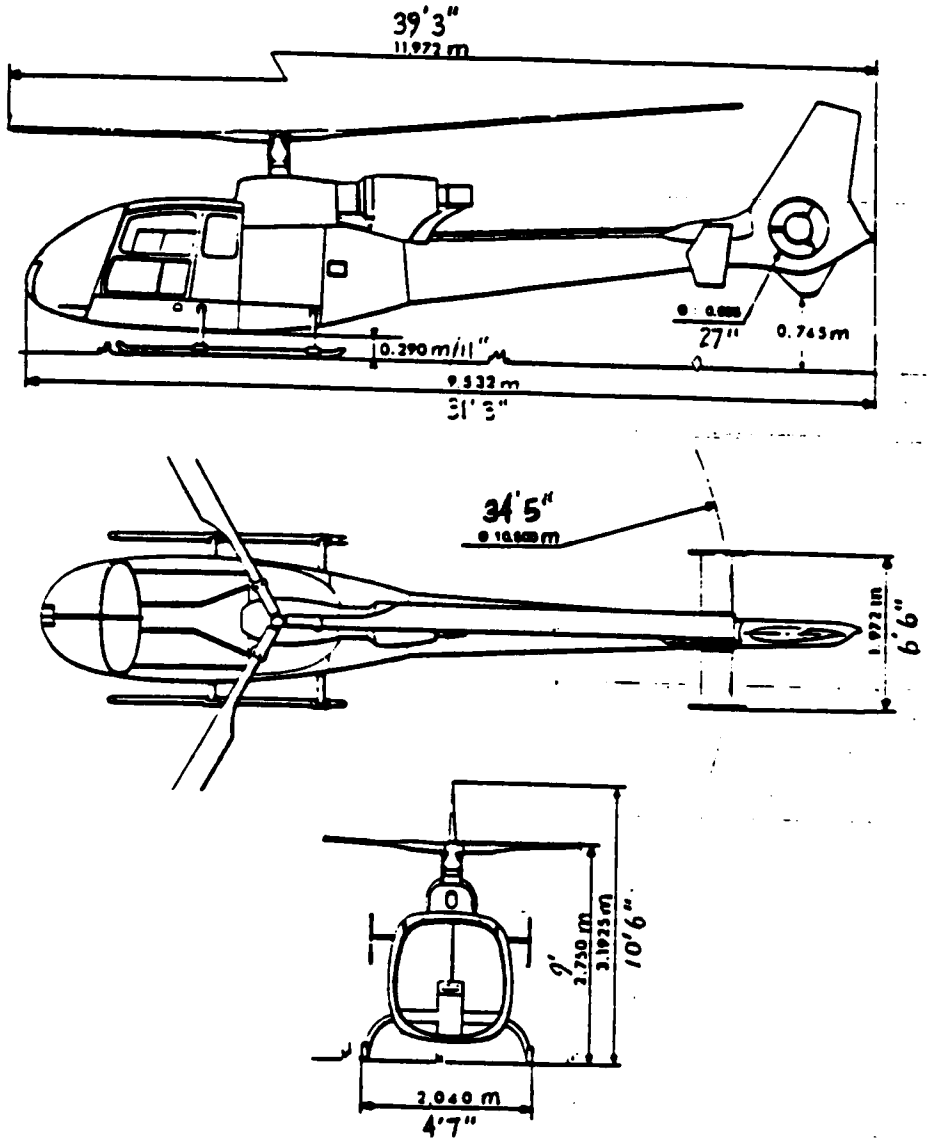


FIGURE 1
GAZELLE HT MK-3: THREE-WAY VIEW

APPENDIX D

APPENDIX E: FLIGHT CONTROL DESCRIPTION

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APPENDIX E: FLIGHT CONTROL DESCRIPTION

E.1 COCKPIT CONTROLS

E.1.1 GENERAL

The cockpit controls were comprised of dual, conventional cyclic sticks, collective levers, and yaw pedals to control lateral and longitudinal pitch, collective pitch, and directional pitch, respectively. Cyclic and collective control movements were transmitted to the main rotor swashplate via a series of control rods, a mixing unit, and servo jacks powered by a single hydraulic system. A stability augmentation system provided rate damping and attitude hold. Directional pedal control movements were transmitted to the tail rotor gear box spider via a series of control rods, bellcranks, cables, and a servo jack. A yaw damper was fitted to the pedals to restrict rate of yaw control. A layout of flight control mechanical linkages is presented in enclosure 1, figure 1, of this appendix.

E.1.2 CYCLIC

The pilot's and co-pilot's cyclic controls were interconnected and operated the swashplate by control rods, linkages, bellcranks, a mixer unit, and servo jacks. The fore-aft and lateral range of movement was limited by stops fitted to the cyclic linkage assembly, positioned under the cabin floor. A hand wheel friction device fitted to the base of the pilot's cyclic control stick allowed the pilot to "stiffen" the cyclic.

E.1.3 COLLECTIVE

The pilot's and co-pilot's collective controls were interconnected and operated the swashplate by control rods, linkages, bellcranks, a mixer unit, and servo jacks. The pilot's collective lever (pilot's only) controlled the collective pitch over the nominal range of 4-16 degrees. The collective pitch applied was shown over a range of 11-14 degrees by a pointer at the base of the collective lever quadrant. Additional markings on the scale showed the optimum pitch range for manual control, 8.5-9.5 degrees. A minimum pitch stop was set to give a nominal 4 degree collective pitch with the lever fully down. A spring-loaded maximum pitch stop was also provided; when the lever was raised the stop was contacted at about 14.5 degrees collective pitch indicating the maximum allowable powered flight collective pitch. The lever could be raised further against the spring during autorotations to 16 degrees pitch. An adjustable intermediate collective pitch stop was incorporated and was adjusted according to the OAT by a knob at the rear of the lever quadrant. The setting was shown by a pointer against the temperature scale on the quadrant. The scale was graduated from 20-50 degrees celsius. The pointer was set at 30°C for temperatures of 30°C, or less. A friction device, adjustable through the grip at the base of the lever, was used to increase/decrease the collective lever forces.

E.1.4 DIRECTIONAL PEDALS

The pilot's and co-pilot's directional pedals were interconnected and operated the servo control rod within

the tail rotor gear box via control rods, linkages, bellcranks, cables, and a tail rotor servo jack. The pedals were adjustable to two positions by pressing the pedal towards the pedal lever, against a spring force, and then rotating the footrest to the desired position. The spring returned the footrest to the locked position when pressure was released. The directional pedals also incorporated a yaw rate damper to limit the rate of tail rotor pedal displacement and provided the pilot with a degree of artificial "feel".

E.1.5 MIXING UNIT

A mixing unit was incorporated to synchronize the cyclic and collective control inputs to the main rotor. Movement of the collective caused its bellcrank to pivot on the X-X axis, which caused the three cyclic bellcranks to pivot simultaneously on the Y-Y axis and move the swashplate vertically through control rods 1, 2, and 3 and associated servos. When cyclic movements were made the cyclic bellcranks pivoted individually to move the associated control rod and hydraulic servo system jacks, thereby tilting the swashplate accordingly. A diagram of the mixing unit is presented in enclosure (1), figure 2.

E.2 HYDRAULIC SYSTEM

E.2.1 GENERAL

The single hydraulic system provided power to counteract rotor system feed back forces and assisted the pilot with control movements. The components for the supply and control of the hydraulics power were

integrated into a power pack attached to and driven from the main transmission gearbox. The pressure and return circuits consisted of rigid and flexible pipes situated between the power pack, distributor, and the main servo controls. System pressure was a nominal 590 psi. A hydraulic system schematic is depicted in enclosure 1, figure 3.

E.2.2 SERVO JACKS

Three main rotor servos, enclosure 1, figure 4, and one tail rotor servo, enclosure 1, figure 5, provided the power to counteract rotor system feed back forces and assisted the pilot with control movements. Each main rotor jack was comprised of a piston rod secured to aircraft structure, a jack body inserted into the control run, and the operating mechanism. The tail rotor servo jack body was secured to the aircraft structure and the input bellcrank lever was connected to the power piston. When flight controls were moved the input lever moved the pilot valve. The pilot valve directed hydraulic fluid to the appropriate side of the piston and the resulting movement of the jack body tilted the non-rotating swashplate of the main rotor. When control movement stopped, the jack body kept moving until the pilot valve was centralized, equalizing the pressure on both sides of the piston and arresting movement.

E.2.3 HYDRAULIC SYSTEM FAILURE

In the event of hydraulic system pressure loss, a lockpin locked the input lever and provided a fixed mechanical link in the control run. A drop in system

pressure below 440 PSI, illuminated the CWP caption "HYD", warning the pilot of impending reversion of flight controls to unboosted (No pressure indicator was provided). A small capacity accumulator was incorporated to maintain sufficient system pressure allowing time to level the wings prior to reversion to unboosted control.

E.3 STABILITY AUGMENTATION SYSTEM

E.3.1 GENERAL

The three-channel stability augmentation system (SAS) provided rate damping and attitude-hold. When selected, the system operated in pitch and roll at all speeds and in yaw at airspeeds below about 48 KIAS. Each channel consisted of a computer unit incorporating a rate gyro which controlled an electro-mechanical actuator in series with the associated flight control. The SAS system schematic is depicted in enclosure 1, figure 6.

E.3.2 SAS CONTROLS

The SAS control unit, enclosure 1, figure 7, housed the SAS engagement switch and the three servo power switches; pitch, roll, and yaw. When the engagement switch was selected ON, a 28V DC power solenoid held the switch in the ON position, and each computer received signals from the associated rate gyro. When a channel servo power switch was selected ON, computer output was connected to the associated actuator. The pilot could disengage the entire system by manually overriding the solenoid and turning the engagement switch OFF, or by

cutting electrical power to the solenoid by depressing the SAS release button on either cyclic stick hand grip. Individual SAS channels could be isolated ON/OFF by placing the respective servo power switch ON/OFF.

E.3.3 ATTITUDE HOLD

Attitude Hold operated in pitch and roll at all speeds and in yaw at airspeeds below about 48 KIAS. The attitude hold signals were disconnected by microswitches when control movements were made to maneuver the aircraft. The signals remained disconnected until the microswitches were re-made by removing the pilot input, and the 2 degrees per second rate sensing switches in each channel sense that the rate of change of control movement in the relevant axis is below 2 degrees/second. The rate sensing switches were not affected by rates of attitude change caused by gusts; the attitude hold remained in effect in these conditions. If stick trim was released with the SAS engaged, the attitude hold signals were disconnected but the rate damping remained in effect.

E.3.4 STICK FEEL

The SAS incorporated electro-magnetic brakes that anchored the pitch and roll actuators when the Stick Feel switch was on. Each brake incorporated a spring-feel system which held the cyclic stick in a datum position, while allowing it to move against the spring pressure. When the cyclic stick was moved, micro-switches in the pitch and roll channel brake units operated to cut out the attitude hold signals to the actuators; rate damping remained in operation. When the

cyclic was released the spring-feel system returned the cyclic to the datum position, then re-connected the attitude hold signals and returned the SAS to full operation about the original trim position. To retrim the brakes to a new trim datum, the brakes were released by pressing either trim release button; the datum was reset and spring-feel re-engaged when the button was released.

E.4 ROTOR SYSTEMS

E.4.1 MAIN ROTOR SYSTEM

The main rotor system was a three bladed, articulated system with a diameter of 34'5". The direction of rotation and was clockwise as viewed from above and operated at a nominal 380 RPM.

E.4.2 Head and Shaft Assembly

The articulated main rotor head and shaft assembly, enclosure 1, figure 8, formed the attachment point for the three main rotor blades and provided hinges for flapping, feathering, and damped lead-lag motion. The assembly consisted of a hollow steel main rotor shaft, rotating and non-rotating swashplates, pitch control rods, a hub yoke assembly, blade spindles, scissors assemblies, and drag dampers. Coning stops limited blade up-flapping to 18 degrees while the droop stop ring supported the blades while static. The rotor blades were attached to the spindle assemblies by pins which provided the lead-lag hinge and the hinge point for manual blade folding. Lead-lag motion was limited by dynamic adapters with integrated dampers. A torsion

rod within the spindle assemblies was incorporated to absorb centrifugal loads and permit pitch changes as transmitted up from the swashplates by the pitch control rods. A preset twist in the torsion rod was used to assist the pilot in making pitch changes in the event of hydraulic power failure.

E.4.3 Main Rotor Blades

The main rotor blades were of fiberglass construction primarily, and could be changed individually. Outboard of the attachment area, a constant cross section, symmetrical airfoil was used. Each blade used a one piece fiberglass leading edge spar with mass balance weights and a honeycomb plastic body. The blade skin consisted of layers of fiberglass cloth, a polyurethane anti-erosion strip on the leading edge, an aluminum alloy tip fairing, and an adjustable trim tab on the carbon fiber reinforced trailing edge.

E.4.4 FENESTRON SYSTEM (ANTI-TORQUE SYSTEM)

The fenestron (tail rotor) system, enclosure 1, figure 9, had 13 light alloy blades shrouded by the tail fin and was mounted on the tail rotor gear box. The cambered blades were hollowed out at the root end to accommodate the metal torsion bar which transmitted the pitch changes from the pitch control plate mounted within the tail rotor gearbox. Fenestron diameter was 27.25", rotated clockwise as viewed from the right (i.e. bottom blade rotated aft), and operated at a nominal 5774 RPM. Fenestron duct depth was approximately 12.5".

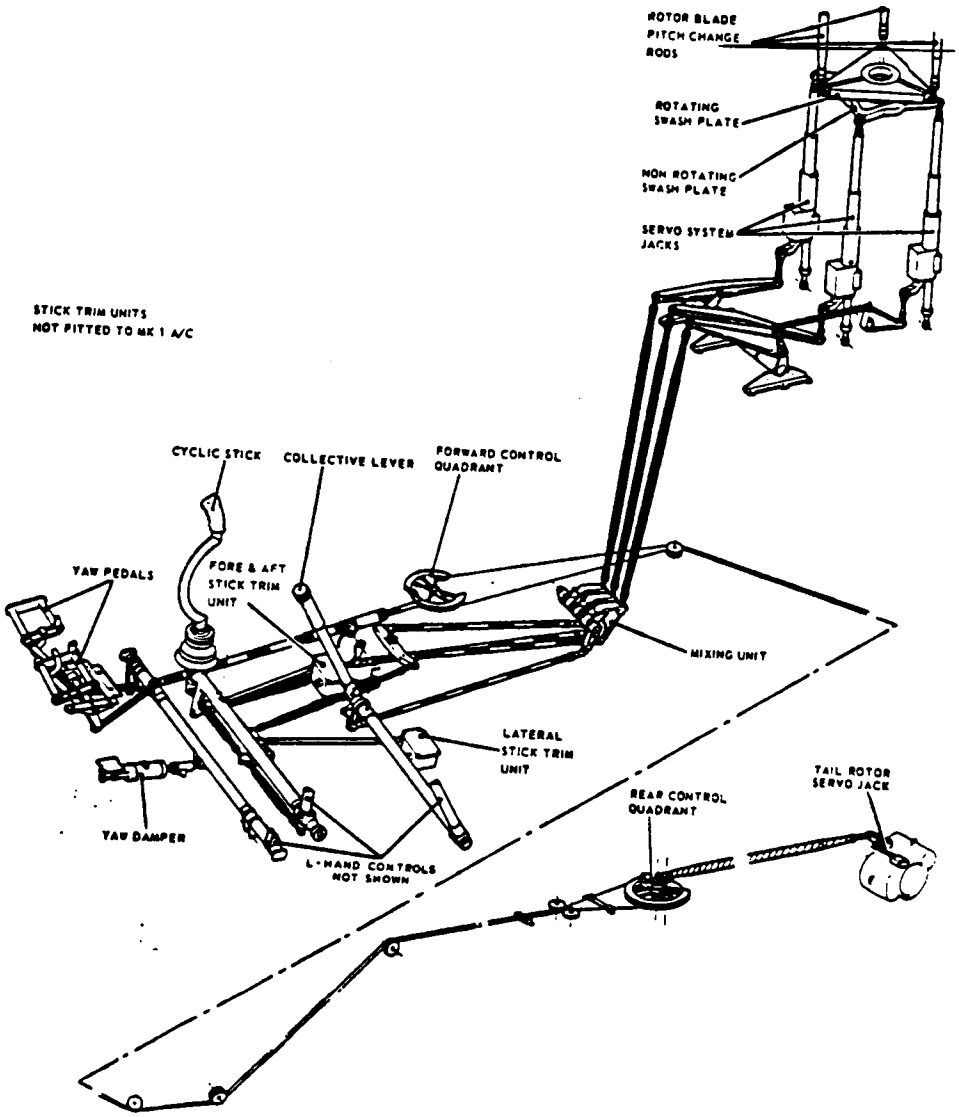


FIGURE 1

GAZELLE HT MK-3: FLIGHT CONTROL MECHANICAL LINKAGES

APPENDIX E
ENCLOSURE (1)

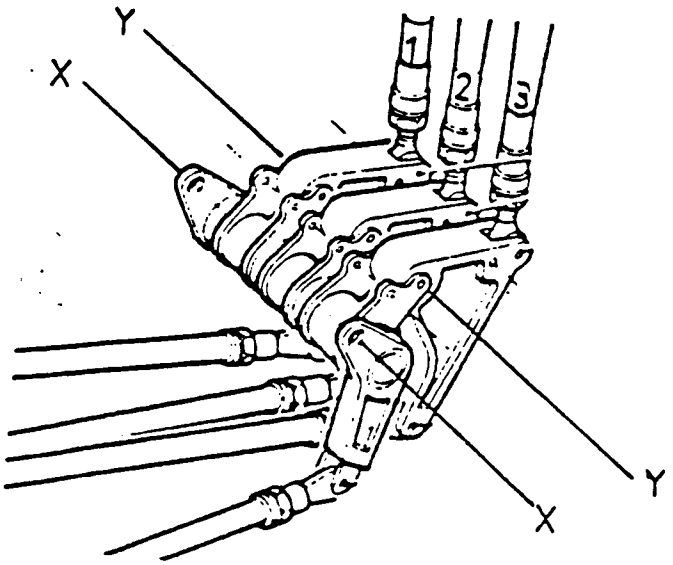


FIGURE 2

GAZELLE HT MK-3: MIXING UNIT

APPENDIX E
ENCLOSURE (1)

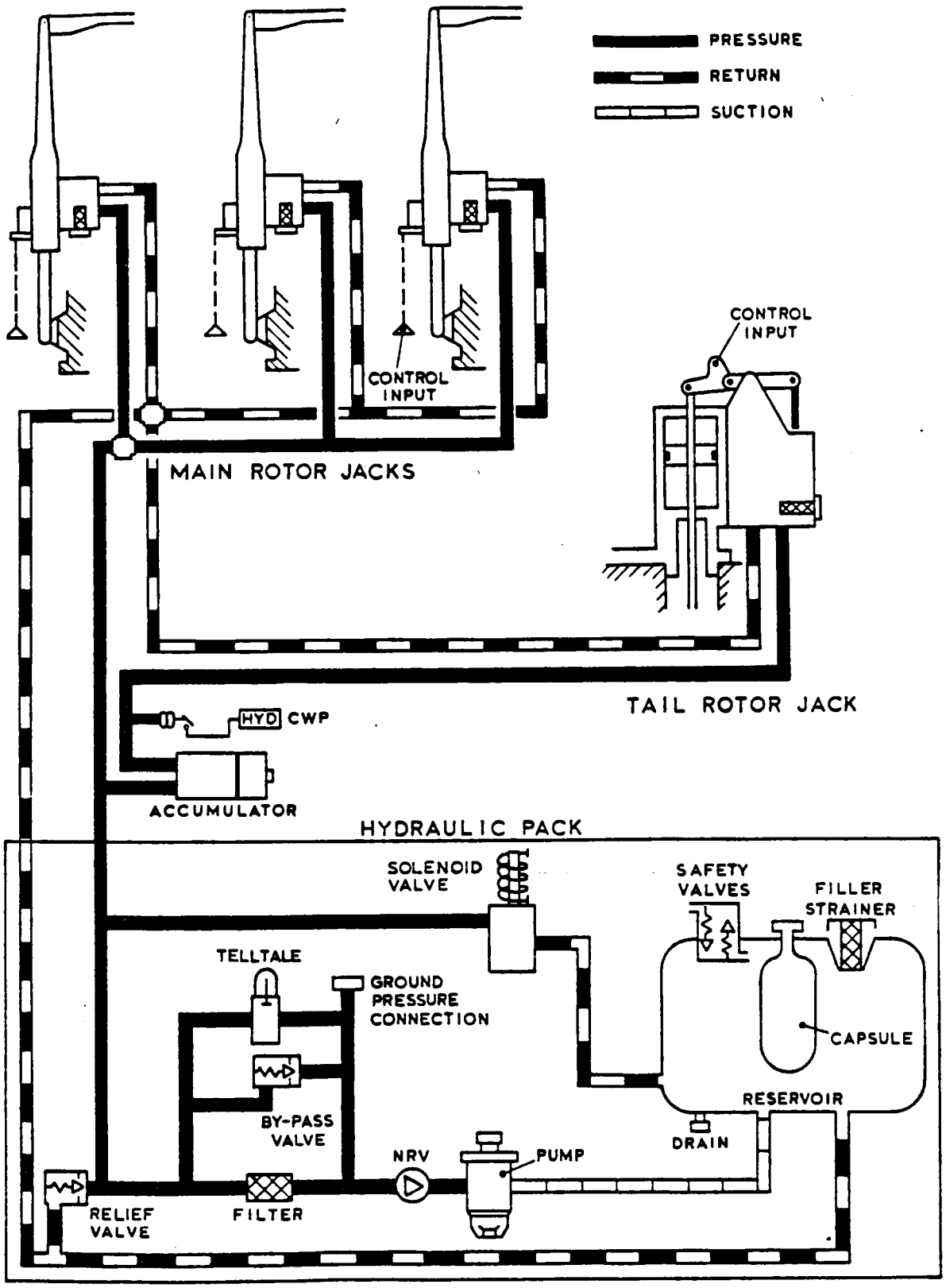


FIGURE 3

GAZELLE HT MK-3: HYDRAULIC SYSTEM SCHEMATIC

APPENDIX E
ENCLOSURE (1)

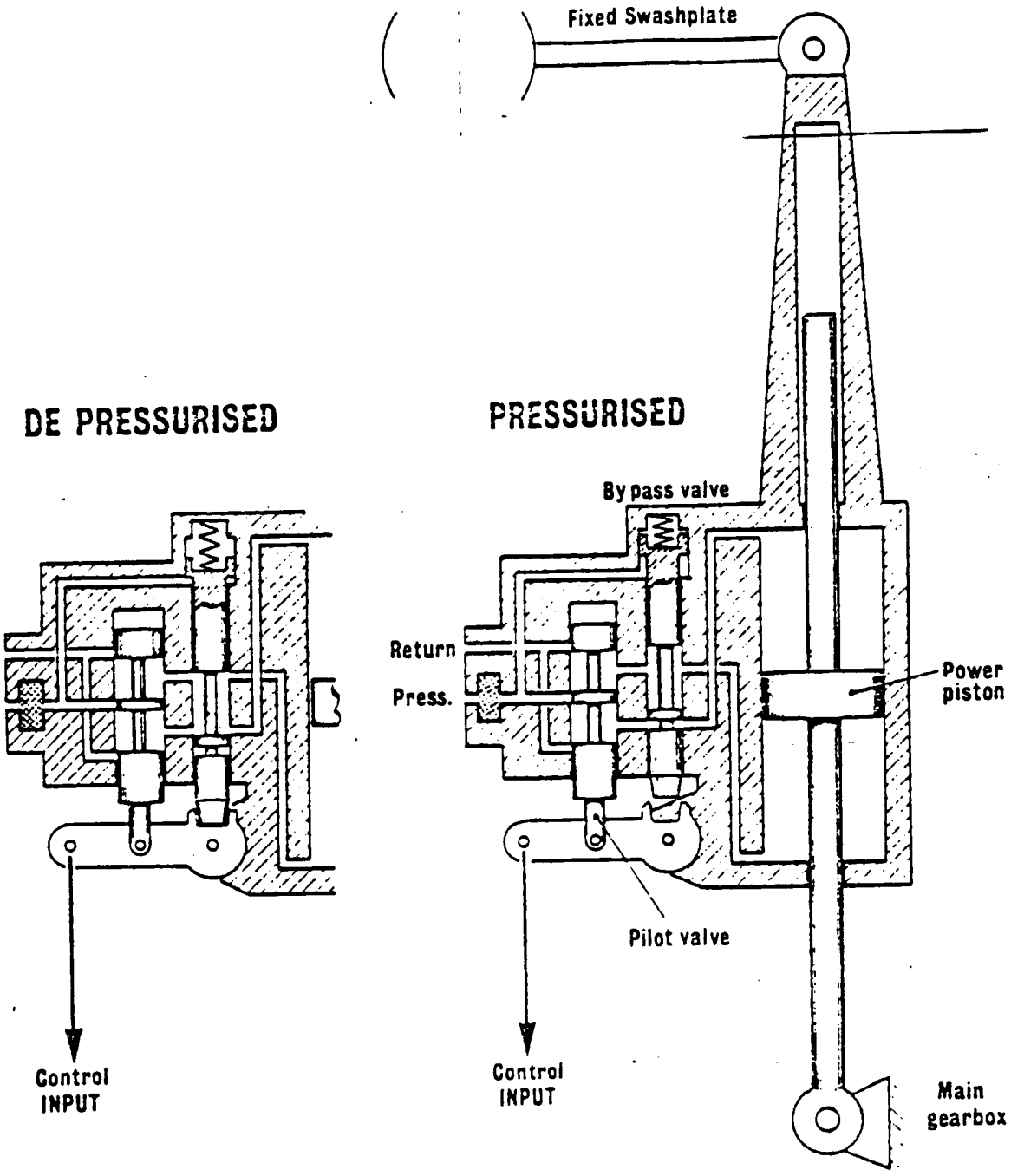


FIGURE 4

GAZELLE HT MK-3: MAIN ROTOR SERVO CUTAWAY

APPENDIX E
ENCLOSURE (1)

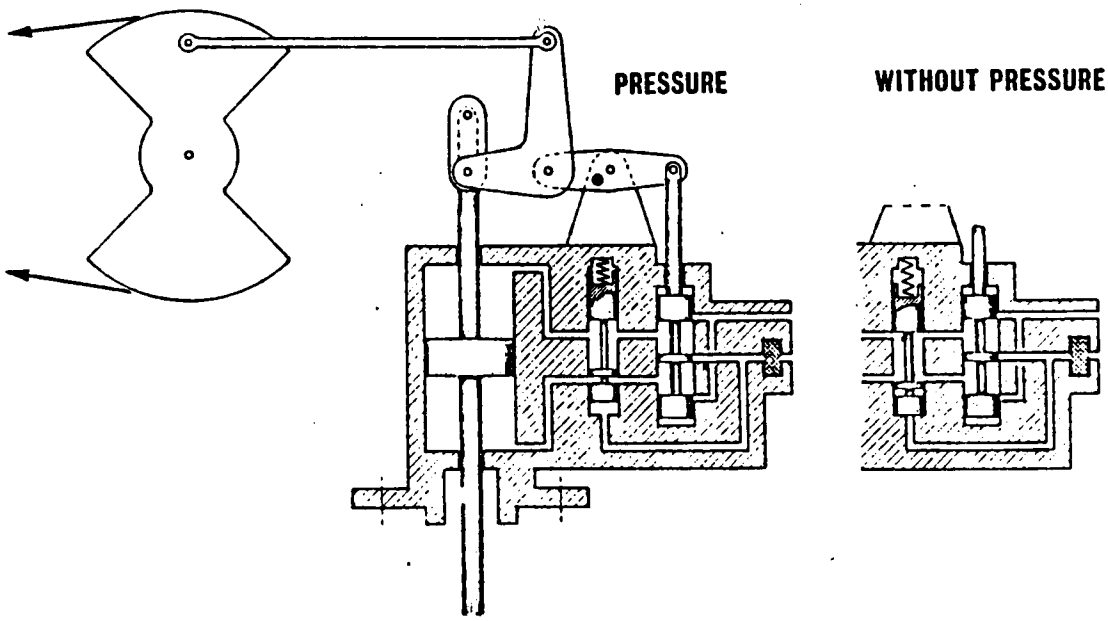


FIGURE 5

GAZELLE HT MK-3: TAIL ROTOR CUTAWAY

APPENDIX E
ENCLOSURE (1)

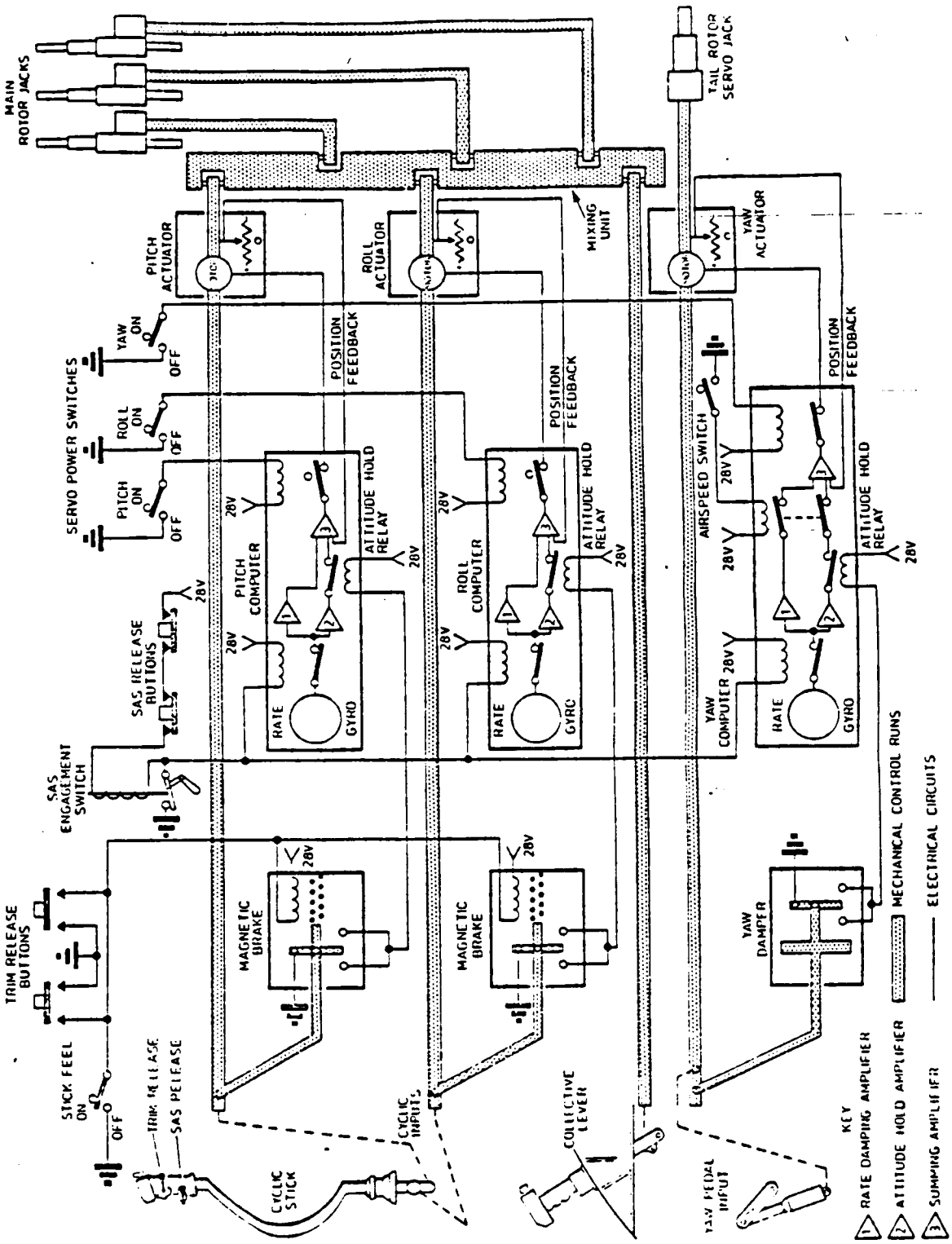


FIGURE 6

GAZELLE HT MK-3: STABILITY AUGMENTATION SYSTEM (SAS)

APPENDIX E
ENCLOSURE (1)

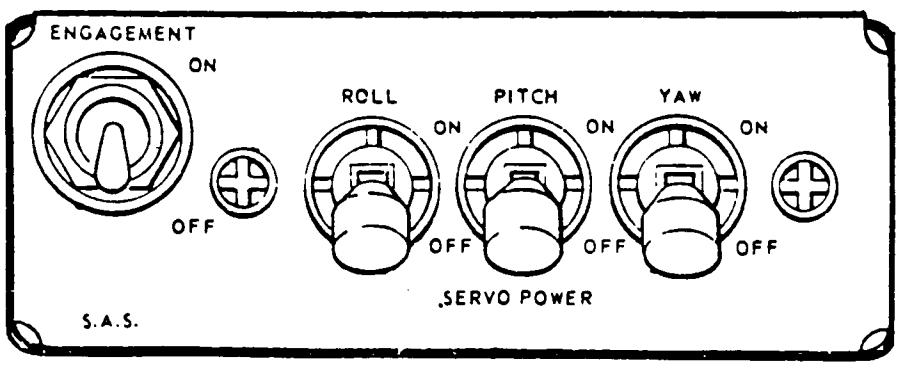


FIGURE 7

GAZELLE HT MK-3: SAS CONTROL UNIT

APPENDIX E
ENCLOSURE (1)

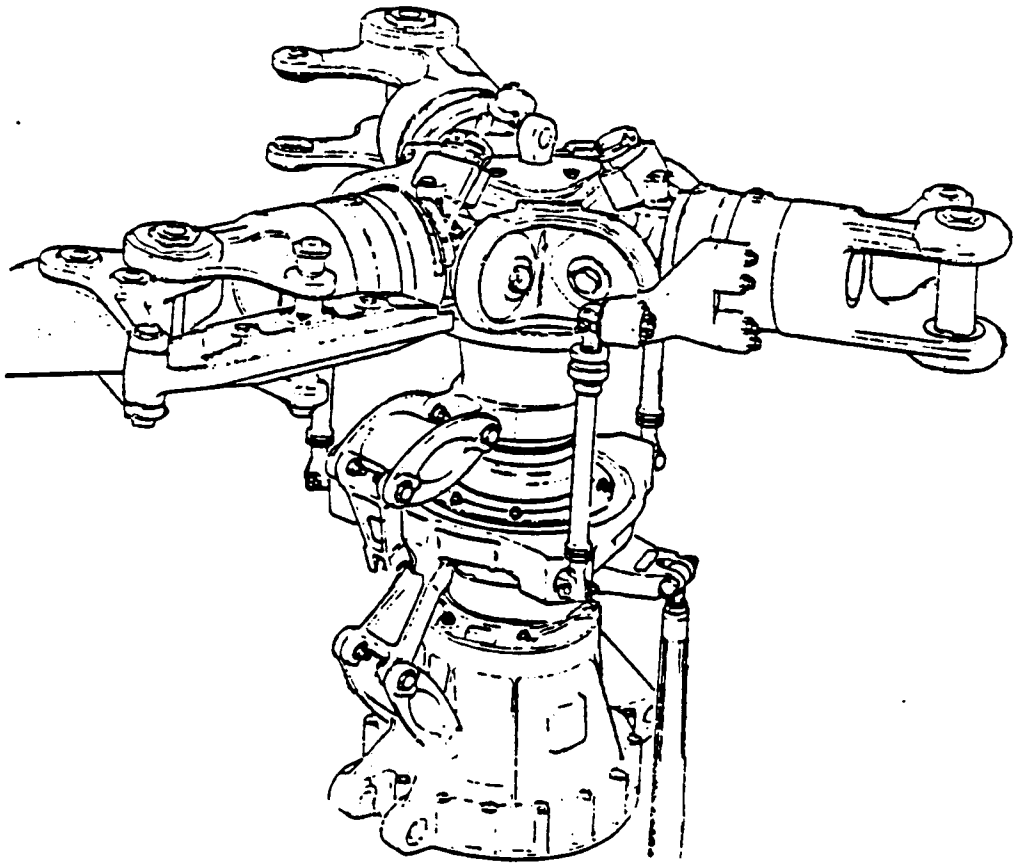


FIGURE 8

GAZELLE HT MK-3: MAIN ROTOR HEAD AND SHAFT ASSEMBLEY

APPENDIX E
ENCLOSURE (1)

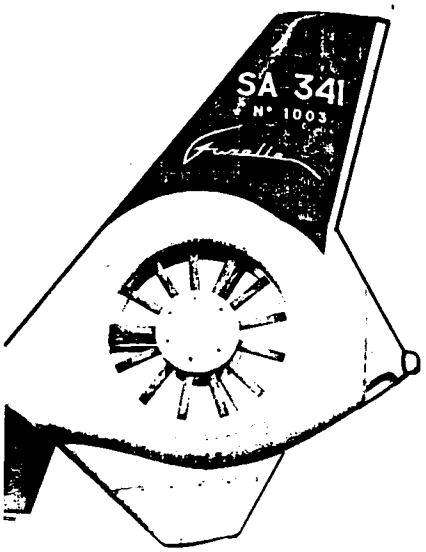


FIGURE 9

GAZELLE HT MK-3: FENESTRON SYSTEM

APPENDIX E
ENCLOSURE (1)

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TABLE 1

TESTS & TEST CONDITIONS

TEST/TASK CATEGORY	TEST/TASK (1,2)	AVE OAT-DEG C	AVE AIRSPEED-KCAS	AVE ALTITUDE (3)	AVE WEIGHT/CG-LBS/INCHES	REMARKS
COCKPIT EVALUATION	ON DECK:DAY	-	-	-	N/A	EXTERNAL ELEC/HYD POWER, ENGINE & ROTOR STATIC. PLUMB-BOB FOR FOV
	IN-FLIGHT: DAY	-8 TO +8.3	0-125	350 -10,000'	3955-3288/ 116.9-115.8	QUALITATIVELY EVALUATED CONCURRENTLY DURING ALL TESTS/TASKS
MECHANICAL CHARACTER.	ON-DECK ROTOR STATIC	-	-	-	-	EXTERNAL ELEC/HYD POWER, ENGINE & ROTOR STATIC, TAPE MEAS & FORCE GAUGE
	ON DECK ROTOR TURNING	-	-	-	-	ROTOR RPM=378 FORCE GAUGE AND QUALITATIVE
	IN FLIGHT	-8 TO +8.3	0-125	350 -10,000'	3955-3288/ 16.9-115.8	QUALITATIVE
ENGINE ASSESSMENT	POWER AVAILABLE	-8 TO +8.3	0-125	350 -10,000'	3621/116.4	CONDUCTED CONCURR W/ENG PERF TESTS TIMED FUEL FLOW CTR
	START	0.6-5.7	-	350-500	-	BATTERY STARTS: HEADWIND, LEFT AND RIGHT CROSSWIND 5 TO 20 KNOTS
	ENG/ROTOR ACCEL					
	STATIC DROOP	8.0	0	350 - 1000	3900/116.9	3 INCREMENTAL TORQUE INCREASES FROM 80 -100%
	TRANSIENT DROOP	-8 TO +8.3	0-125	350 -10,000'	3955-3288/ 16.9-115.8	QUALITATIVELY ASSESSED AS SATISFACTORY
	SHUTDOWN	0.6-5.7	-	350-500	-	ROTOR BRAKE HEADWIND, LEFT CROSSWIND 5 TO 20
MISSION MANEUVERS	VERTICAL TAKEOFF TO HOVER	.6-8.3	5-25 KTS WIND	0-5'	3925/116.9	3925/116.9
	AIR TAXI	8.3	5 KTS GS	5'	3920/116.9	FWD, SIDE, REAR, PED TRNS
	TRANSITION TO FWD FLT	8.3	0-60	5'-300'	3915/116.9	INTO WIND 80%-100% POWER
	APPROACH TO HOVER	8.3	80-0	500'-5'	3910/116.9	INTO WIND
	HOVER IGE	8.3	AMBIENT WIND 5-25 KTS	5'	3905/116.9	INTO WIND, XWIND, DOWNWIND
	HOVER OGE	8.3		80'	3900/116.9	INTO WIND, XWIND
	VERTICAL LANDING FROM HOVER	0.6-8.3	AMBIENT WIND	5'-0	3885/116.9	INTO WIND, XWIND, DOWNWIND
	SIDEWARD AND REARWARD FLIGHT	8.0	0-30	10'-20'	3875/116.9	INTO WIND

TABLE 1 (CONT.)

TESTS & TEST CONDITIONS

TEST/TASK CATEGORY	TEST/TASK (1,2)	AVE OAT-DEG C	AVE AIRSPEED-KCAS	AVE ALTITUDE (3)	AVE WEIGHT/CG-LBS/INCHES	REMARKS
MISSION MANEUVERS (CONT)	HOVERING PEDAL TURNS	8.0	5-25 KTS WIND	5'-10'	3970/116.9	360 DEGREE PEDAL TURNS ABOUT TRANSMISSION, TAIL, & NOSE
	BEST RATE CLIMB	6.0	65	350' -10,000'	3960/116.9	MAX POWER
	DESCENT	6.0	65	1500-500	3950/116.9	MIN POWER
	EVASIVE ROLLS	7.0	100	500'	3940/116.9	TRIM RELEASED 45-45 DEG ROLL REVERSALS
(LOW LEVEL FLIGHT)	LEVEL MANEUVERING	7.0	90-100	100'-200'	3870/116.8	BUILD UP TO LOWER ALTITUDE
(CONTOUR FLIGHT)	BUNT, ROLL, QUICKSTOP	7.0	50-90	50'-100'	3870/116.8	BUILD UP TO LOWER ALTITUDE
(NAP OF EARTH (NOE))	BUNT, ROLL, QUICKSTOP, UNMASK/ REMASK BOTH VERT &HORIZ	7.0	<50	10'-50'	3870/116.8	BUILD UP TO LOWER ALTITUDE
	STEEP APPROACH	7.0	80-5	300'-10'	3870/116.8	FROM CIRCUIT TO GRASS
	NO HOVER LANDING	8.3	<5 GS	10'-0	3822/116.8	10 KT HEAD WIND
	RUNNING LANDING	3.5	20 KT TOUCHDOWN	350 MSL	3490/116.2	20 FOOT GROUND SLIDE ON GRASS SURFACE
	SLOPE LANDINGS	3.5	10 KTS WIND	350 MSL	3623/116.5	12 DEG NOSE UP, 9 DEG LEFT CROSS SLOPE
LOW AIRSPEED (4)	TRIMMED FLIGHT CONTROL POSITIONS/ CRITICAL AZIMIUTH	8.3	0-30	10'-20' 500' MSL	3612/116.5	000, 045, 090, 135, 180, & 270 REL WIND, 10 KT INCR, ESTIMATED SPEED AND AZIMUTH
	STATIC STABILITY	8.3	0-10	10'-125' 500' MSL	3320/115.9	ESTIMATED 5 KT INCREMENTS FORWARD, SIDEWARD, & REARWARD
	UNUSUAL ATTITUDE RECOVERY PRACTICE	5.4	AMBIENT WIND 10 KTS 0 GS	1000' AGL 1500 MSL	3620/116.5	+/-15, 30, 45 DEG NOSE PITCH
	LONG TERM RESOPONSE	5.4			3570/116.5	NATURAL EXCITATION & 5 DEG PITCH UP AGAINST TRIM COUPLED RESP
	LONG/LAT CONTROL RESPONSE	5.4			3500/116.2	STEP INPUTS: 1/4" INCR USING CONTROL FIXTURE MAX INPUT SIZE: LONG/LAT-1"
	PEDAL CONTROL RESPONSE	8.3		20' AGL 500 MSL	3430/116.2	PED-1/2"

TABLE 1 (CONT.)

TESTS & TEST CONDITIONS

TEST/TASK CATEGORY	TEST/TASK (1,2)	AVE OAT-DEG C	AVE AIRSPEED	AVE ALTITUDE (3)	AVE WEIGHT/CG-LBS/INCHES	REMARKS
CLIMB AND DESCENT PERFORMANCE	VERTICAL CLIMBS	7.3	AMBIENT WIND: 15-20	FLD ELEV TO 2000'	3857-3822/ 116.7-115.8	FREE HOVER AIR REFERENCE VERT CLIMBS POWER INCR OF 80,90,100%
	SAWTOOTH CLIMBS AND DESCENTS	2.5	38-110	2500	3846-3747/ 116.7	MAX POWER CLIMBS (MPS,102.5 % Q, OR 580 DEG T4) MIN POWER DESCENTS-(NEEDLES JUST JOINED)
LEVEL FLIGHT PERFORMANCE	STRAIGHT AND LEVEL	0.9-0.2	40-120	3100-3480	3725-3652/ 116.7-16.5	CONST W/SIGMA = 3970 LBS
FORWARD FLIGHT TRIMMED FLIGHT CONTROL POSITIONS	STRAIGHT AND LEVEL	0.9-0.2	40-120	3100-3480	3725-3652/ 116.7-16.5	CONCURRENT WITH LEVEL FLT PERFORMANCE
	STEADY LEVEL TURNS	3.2	100	2000'	3600/116.5	15, 30, 45 DEG AOB LEFT & RIGHT
	CLIMBING & DESCENDING FLIGHT	3.2	100	2000'	3590/116.5	CLIMBS 500 FPM, MAX PWR WINGS LEV, & MAX PWR W/30 DEG RIGHT AOB DESCENTS 500 FPM, MIN COLL PITCH WINGS LEV, & MIN COLL PITCH W/30 DEG LEFT AOB
FORWARD FLIGHT STATIC TESTS	LONGITUD STABILITY	3.0	100 TRIM	2000	3789/116.6	- 2 & +/- 5, 10, 15, 20KTS FROM TRIM COLL. FIXED
	LAT-DIRECT STABILITY (SHSS)	3.1			3765/116.6	FIX COLLECT MAINTAIN 100 VARY BETA: INH, 0, +/- 2,5,10,15,20 DEG
	CYCLIC-ONLY TURNS	3.1			3745/116.6	20 DEG AOB LEFT AND RIGHT
FORWARD FLIGHT STATIC TESTS (CONT)	PEDAL-ONLY TURNS	3.1	100	2000	3725/116.7	+/-5, 10 DEG BETA
UNUSUAL ATTITUDE RECOVERY PRACTICE	NOSE HIGH/LOW	3.1	100	2000	3700/116.7	BUILT UP TO +30, -20 DEG PITCH
	LEFT/RIGHT WING LOW					BUILT UP TO +/-45 DEG ROLL
FORWARD FLIGHT DYNAMIC TESTS	LONG TERM	3.1	100	2000	3524/116.3	-NATURAL EXCITATION & ARTIFICIAL EXCITATION: (SLOWED 10 KT BLOW TRIM & RESET CYCLIC) -ALL CONTROLS FIXED

TABLE 1 (CONT.)

TESTS & TEST CONDITIONS

TEST/TASK CATEGORY	TEST/TASK (1,2)	AVE OAT-DEG C	AVE AIRSPEED	AVE ALTITUDE (3)	AVE WEIGHT/CG-LBS/INCHES	REMARKS
FORWARD FLIGHT DYNAMIC TESTS (CONT)	SHORT TERM	3.1	100	2000	3500/116.3	LONG FREQ SWEEP FOR RESONANT FREQ = 1 1/4 Hz +/-1" LONG CYCLIC DOUBLET TO EXCITE RESPONSE ARTIFICIALLY
	SPIRAL STABILITY	3.1	100	2000	3562/116.3	FLY TO 20 DEG AOB RIGHT AND LEFT, RETURN LAT CONTROL TO TRIM & FIX
	LATERAL-DIRECT OSCILLAT. (LDO)	3.1	100	2000	3550/116.3	ARTIFICIAL EXCITATION: 1/2" PED DOUBLET,
FWD FLT CONTROL RESPONSE	LONGITUD. & LATERAL	3.1	100	2000'	3489/116.2	1/4 " MAX INCR, MAX STEP: 1 1/2" RT, 1 3/4" FWD, AFT, & LEFT
MANEUVERING STABILITY	COORDINATED CONSTANT AIRSPEED TURNS	3.1	100	2000'	3655/116.5	COLLECTIVE FIXED, BALL CENTERED, 15,30,45 DEG TURNS LEFT AND RIGHT
	SYMMETRIC PULLUPS AND PUSHOVERS	3.1	100	2000'	3625/116.5	PULLUPS TO 2.0g PUSHOVERS TO 0.3g 1/4" INCR MAX INPUT:+1 1/2, -1 3/4
GUST RESPONSE	LEVEL FORWARD FLIGHT	4.0	40-125	500	3425/116.2	NATURAL EXCITATION OPEN LOOP COUPLED AND CLOSED LOOP EVAL
DEGRADED MODES	SAS OFF APPROACH & LANDING	3.0	80-0	500-0	3520/116.3	TOTAL SAS OFF
	HYDRAULIC OFF FLIGHT, APPROACH, & NO HOVER LANDING	3.0			3475/116.2	SECURED FROM STRAIGHT & LEVEL,80 KCAS 500' REENGAGE ON DECK, WITH CONTROL FORCES ZERO
	AUTO-ROTATIONS	3.0	70-100 KCAS ENTRIES	500 & 1000 FOOT ENTRIES	3400/116.2	SAS-OFF, STRAIGHT-INS THROTTLE LEVER FULLY AFT WHEN LDG AREA ASSURED

Notes:
1. Crew of two, full fuel, Hydraulics, SAS, Stick Feel-On for all flights except degraded modes. Clean configuration, doors on, engine air bleed systems off.
2. Test gross weight will ranged between 1793-1491Kg (3970-3288 Lbs). 100 Kg ballast for all flights except degraded modes and light hover performance,Test CG range throughout the flights was (116.9-115.8 (mid). Ballast was positioned on the cabin floor, centered laterally, behind the front crew seats.
3. Altitudes above 500 feet are pressure altitudes; altitudes at or below 500' are AGL, unless otherwise noted.
4. Low airspeed tests were conducted in steady 10 knot winds.

TABLE 2

SAMPLE LOAD STATEMENT

TECHNICAL SERVICES DIV. **A&AEE LOAD STATEMENT** FORM T.S.18 ISS

AIRCRAFT:- GAZELLE Mk. 2 S/No. XZ 939

W & C of G INDEX No. 1888 ISSUE D SHEET No. FOUR

W & C.G. LIMITS IN ACC WITH:- APF / 167/026 D/D 27 Feb. 87

STATEMENT BASED ON T.S.14 DATED - 030789

A/C. LOADING DATA	LOAD No. (11)	(12)	(13)
CREW @ 81·6 Kg:- PILOT	ONE	ONE	ONE
CO-PILOT	ONE	ONE	ONE
FUEL @ 0·80 Sp.Gr.:-			
Kg	360	360	317
Litres	450	450	396
PILOT'S BACK CUSHION	ON	ON	ON
CO-PILOT'S BACK CUSHION	ON	ON	ON
PILOT'S PARACHUTE			
CO-PILOTS PARACHUTE			
OXYGEN RIG @ 210B AFT	OFF	OFF	OFF
PASSENGERS @ 81·6 Kg:-			
AFT PORT			
AFT CENTRE			
AFT STBD			
MOD S 581	EMBODIED	EMBODIED	EMBODIED
BALLAST @ STN 2251·5 mm AFT OF DATUM	50 Kg	100 Kg	150 Kg
NOTE: BALLAST WEIGHT INCLUDES ALL FIXINGS, TIES & LASHINGS.			
AIRCRAFT WEIGHT AT START-UP Kg	1743	1793	1800
C of G AFT FROM DATUM (U/C DOWN) mm	2967·8	2947·8	2920·1
ZERO FUEL WEIGHT Kg	1383	1433	1483
C of G AFT FROM DATUM (U/C DOWN) mm	2942·8	2918·7	2896·2
ISSUED BY:-		[signature]	
DATE OF ISSUE		23 03 89	

APPENDIX F

APPENDIX G: CURVES

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GAZELLE HT MK-3 HELICOPTER
SER NO:XZ 939

CONFIGURATION: CLEAN, DOORS/ WINDOWS CLOSED

SAS: ON
HYDRAULICS: ON
STICK FEEL: ON
METHOD: CONSTANT COLLECTIVE, STEADY TURNS, PULL UPS & PUSH-OVERS

AVE GW: 3655 LBS
AVE CG: 116.5 IN(MID)
AVE Hp: 2000 FT
AVE OAT: 3.1 DEG C
ROTOR SPEED: 374 RPM
TRIM SPEED: 100 KCAS

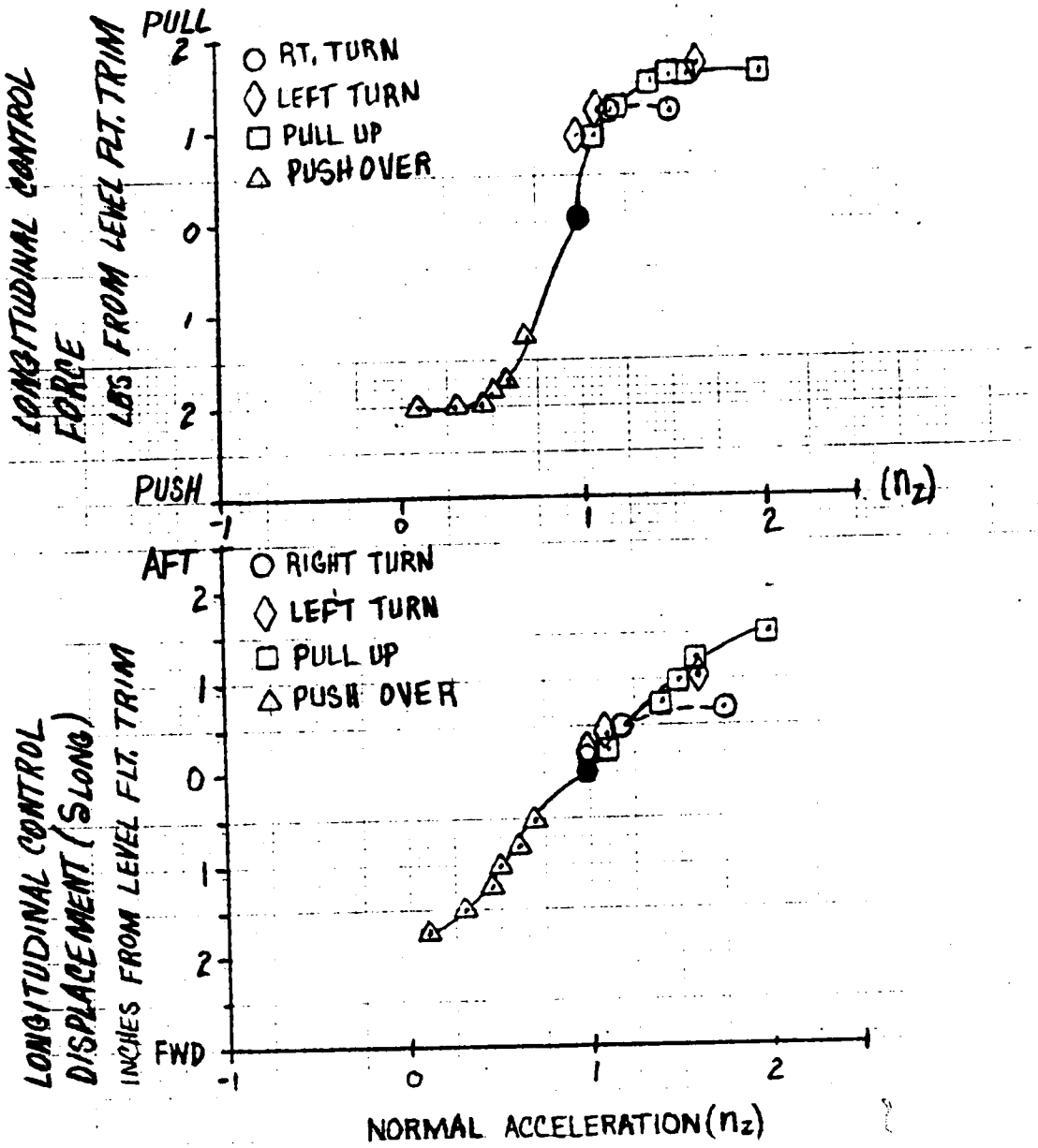


Figure 1
MANEUVERING STABILITY

APPENDIX G

GAZELLE HT MK-3 HELICOPTER
SER NO:XZ 939

CONFIGURATION: CLEAN, DOORS/
WINDOWS CLOSED

SAS: ON
HYDRAULICS: ON
STICK FEEL: ON
METHOD: ESTIMATED AIRSPEED

AVE GW: 3612 LBS
AVE CG: 116.5 IN(MID)
AVE Hp: 500 FT
AVE OAT: 8.3 DEG C
ROTOR SPEED: 374 RPM
SKID HEIGHT: 10-20 FT

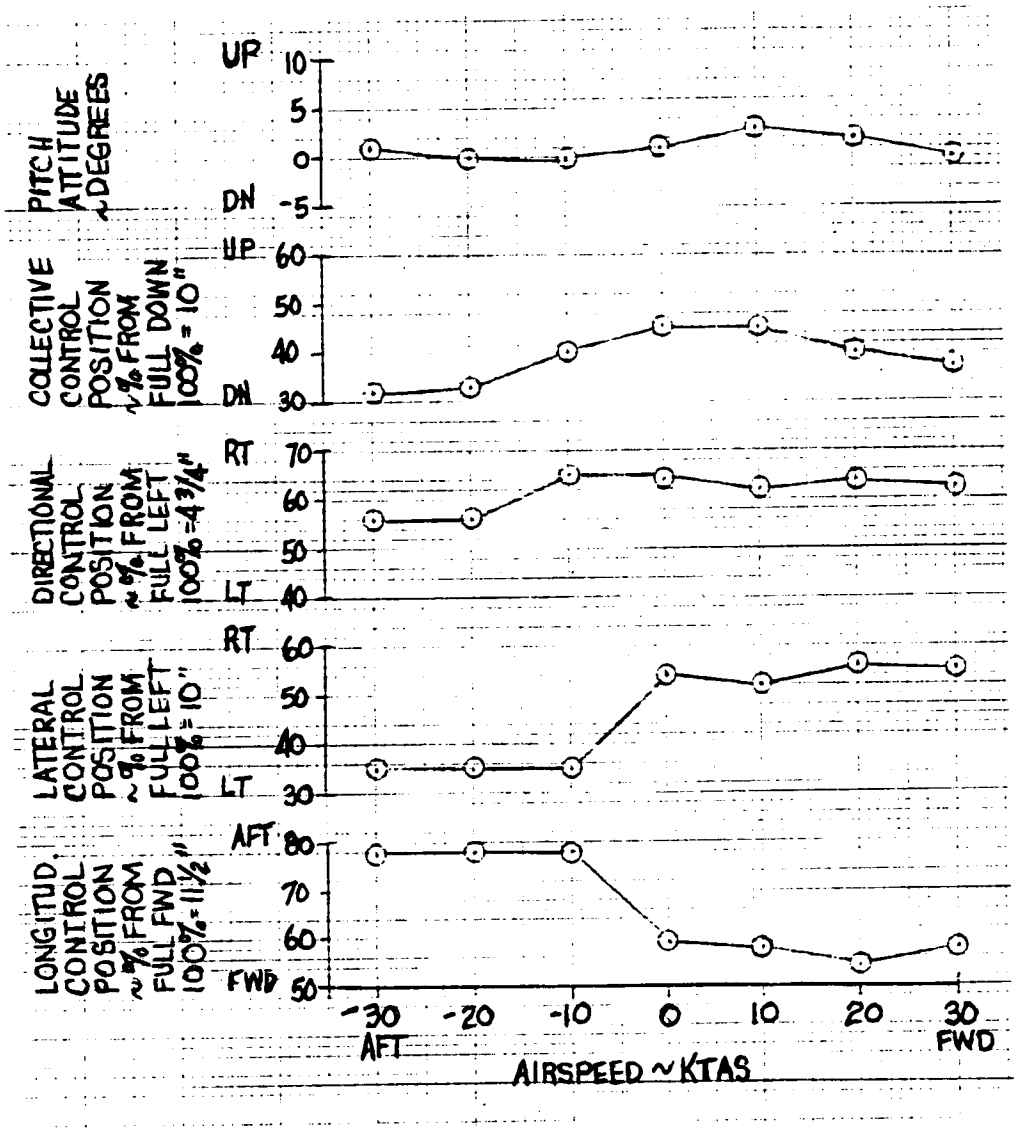


Figure 2
LOW AIRSPEED TRIMMED FLIGHT CONTROL POSITIONS:
FORWARD & REARWARD

APPENDIX G

GAZELLE HT MK-3 HELICOPTER
SER NO:XZ 939

CONFIGURATION: CLEAN, DOORS/ WINDOWS CLOSED
SAS: ON
HYDRAULICS: ON
STICK FEEL: ON
METHOD: ESTIMATED AIRSPEED

AVE GW: 3612 LBS
AVE CG: 116.5 IN(MID)
AVE Hp: 500 FT
AVE OAT: 8.3 DEG C
ROTOR SPEED: 374 RPM
SKID HEIGHT: 10-20 FT

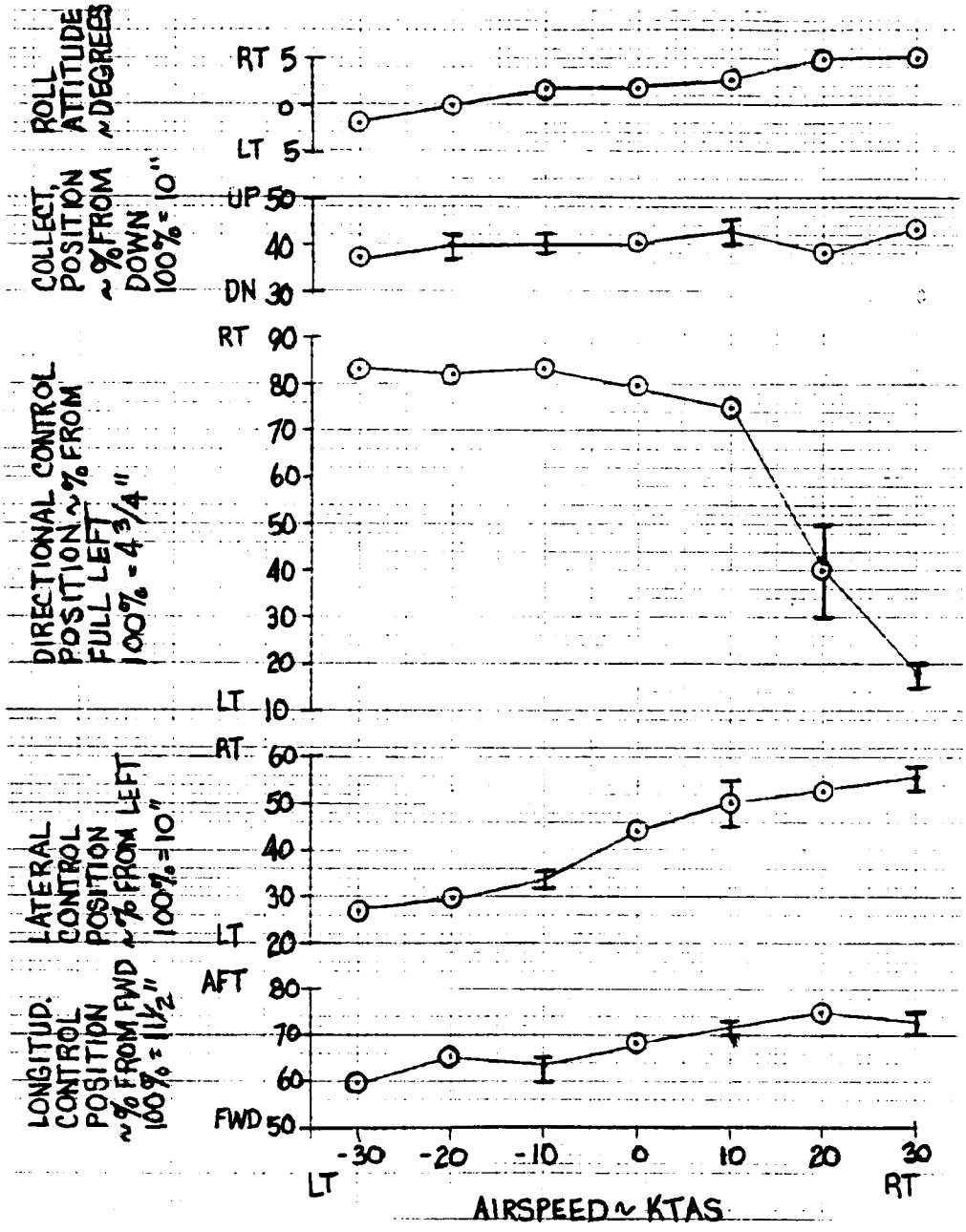


Figure 3
LOW AIRSPEED TRMMED FLIGHT CONTROL POSITIONS: SIDEWARD

GAZELLE HT MK-3 HELICOPTER
SER NO:XZ 939

CONFIGURATION: CLEAN, DOORS/WINDOWS CLOSED
SAS: ON
HYDRAULICS: ON
STICK FEEL: ON
METHOD: ESTIMATED AIRSPEED

AVE GW: 3612 LBS
AVE CG: 116.5 IN(MID)
AVE Hp: 500 FT
AVE OAT: 8.3 DEG C
ROTOR SPEED: 374 RPM
SKID HEIGHT: 10-20 FT

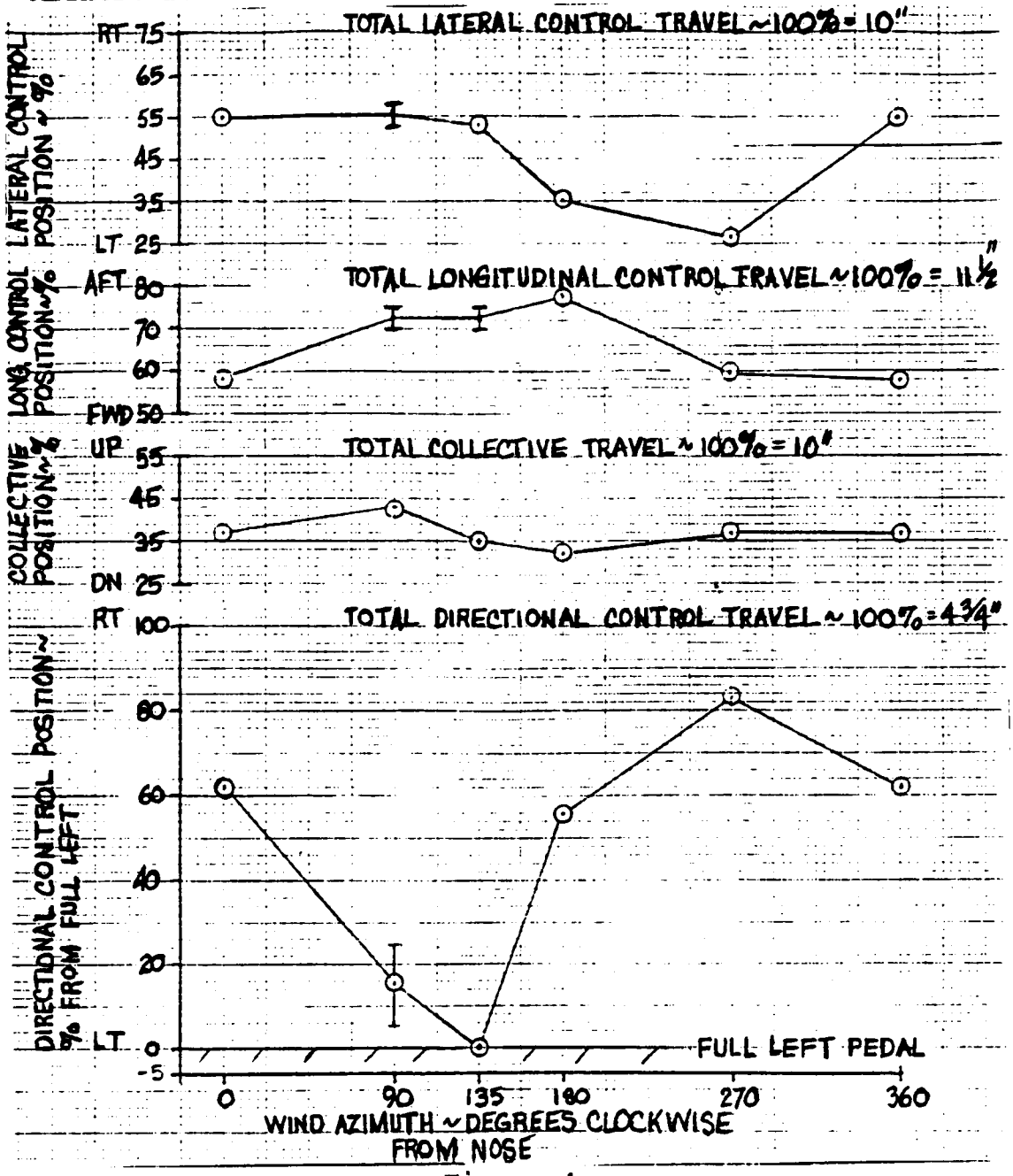


Figure 4
LOW AIRSPEED CRITICAL AZIMUTH CHARACTERISTICS

GAZELLE HT MK-3 HELICOPTER
SER NO:XZ 939

CONFIGURATION: CLEAN, DOORS/ WINDOWS CLOSED
SAS: ON
HYDRAULICS: ON
STICK FEEL: ON
METHOD: ESTIMATED AIRSPEED

AVE GW: 3612 LBS
AVE CG: 116.5 IN(MID)
AVE Hp: 500 FT
AVE OAT: 8.3 DEG C
ROTOR SPEED: 374 RPM
SKID HEIGHT: 10-20 FT

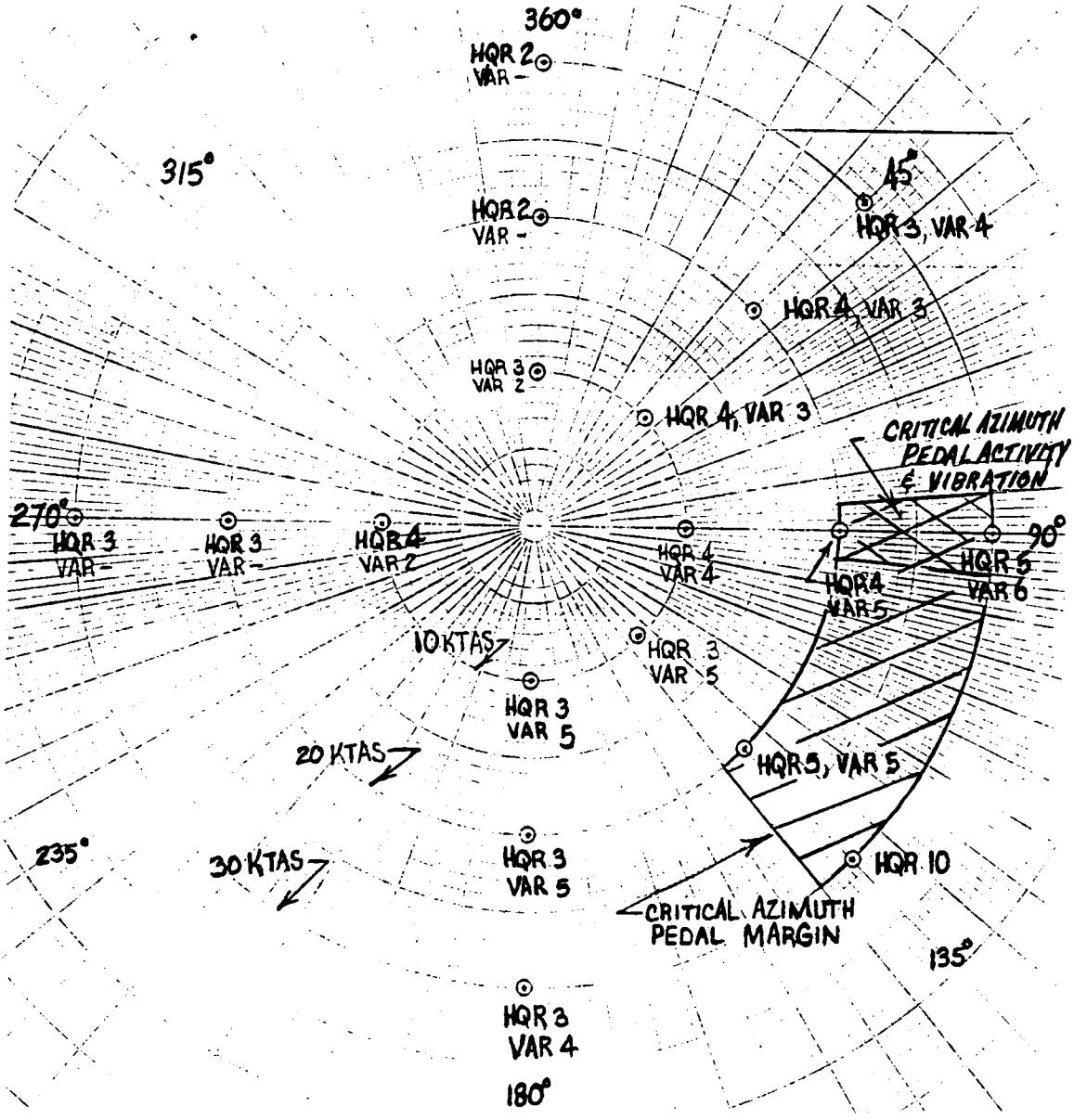


Figure 5
SUMMARY OF LOW AIRSPEED CRITICAL AZIMUTHS

APPENDIX H: MECHANICAL CHARACTERISTICS AND FORWARD FLIGHT, TRIMMED FLIGHT CONTROL POSITIONS

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APPENDIX H, ENCLOSURE (1): MECHANICAL CHARACTERISTICS

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H.1 FLIGHT CONTROL SYSTEM MECHANICAL CHARACTERISTICS

H.1.1 GENERAL

The flight control system, mechanical characteristics were evaluated with the hydraulics, SAS, and stick feel on during a 2.0 hour ground evaluation with external electrical and hydraulic power applied, engine and rotor static, and main rotor spread. The mechanical characteristics were also evaluated on the ground with the engine running and rotor turning at 378 RPM and in flight under the conditions presented in table H-1, below. Control forces were measured using two force gauges, a Bramley 20 pound gauge, serial number 137-5661, calibrated 30 September, 1990 and graduated in 1/2 pound increments. The other force gauge, used for checking pedal damper rates, was an ETPS 60 pound gauge, serial number BD-EI-199, calibrated in November, 1988. Total and trim system freeplay was negligible in all controls. No control coupling nor any stick mass imbalance was noted throughout the tests. Operation of cyclic and collective friction collars were found to be satisfactory. No control mixing was encountered. Cyclic control envelope, presented in enclosure (1), figure 1, was rectangular with collective and pedals in all positions. Total control displacements are presented in table H-1, below. A detailed flight description is presented in appendix E.

Table H-1

TOTAL FLIGHT CONTROL TRAVEL

CONTROL	FULL RANGE OF TRAVEL - IN.
LONGITUDINAL CYCLIC	11 1/2
LATERAL CYCLIC	10
DIRECTIONAL PEDALS	4 3/4
COLLECTIVE LEVER	10

H.1.2 LONGITUDINAL CYCLIC CONTROL

Longitudinal cyclic control characteristics data are presented in enclosure (1), figure 2. Breakout plus friction forces were 1/4 pound forward and 1/2 pound aft. Average value of friction band was 1.2 pounds forward and 1 pound aft. Force gradient slope was positive or zero with no discontinuities noted. Average force gradient, both forward and aft, for the first one inch of travel was 0.8 pounds per inch. Average force gradient for the remainder of travel was 0.3 lbs/in. forward and 0.5 lbs/in. aft. Limit control force measured from the stick centered and trimmed position, was 4 1/2 pounds forward and 3 pounds aft. Limit control force from full aft to full forward was 6 1/2 pounds while full forward to full aft was 5 1/2 pounds. Longitudinal control centering was positive, nearly absolute; cyclic centered within 1% (or 1/10 of an inch) of trim. Longitudinal stick dynamics were heavily damped in that releasing the stick after a +/-1" displacement from trim generated only one 1/4"-1/2" overshoot. Stick jump for a one inch deflection from the mid-longitudinal position, was negligible. The longitudinal mechanical characteristics met the

requirements of Military Specification 8501A, with the exception of the following:

Paragraph 3.2.4: In that the longitudinal force gradient slope was not positive over the entire range of travel.

Paragraph 3.2.7: In that the forward longitudinal breakout plus friction force was 1/4 lbs, 1/4 lbs (100%) less than the minimum required 1/2 lbs.

H.1.3 LATERAL CYCLIC CONTROL

Lateral cyclic control data are presented in enclosure (1), figure 3. Breakout plus friction force was 3/4 pounds right and left. Average value of the friction band was 1.3 pounds right and 1.2 pounds left. Force gradient slope was zero or positive with no discontinuities noted. Average force gradient, both right and left, for the first inch of travel was 0.6 pounds per inch. The average force gradients decreased to 0.2 pounds/inch after the first one inch. Lateral limit control force measured from a stick centered and trimmed position was 2 1/2 pounds right and 2 pounds left. Limit control force measured from full left to full right cyclic was 5 pounds and from full right to full left was 3 1/2 pounds. Lateral control centering was absolute. The lateral stick jump for a one inch deflection from a control centered and trimmed position was negligible. The lateral mechanical characteristics met the requirements of Military Specification 8501A, with the exception of the following paragraphs:

Paragraph 3.2.4: In that the lateral force gradient slope was not positive across the entire range of travel.

H.1.4 DIRECTIONAL PEDAL CONTROL

Directional pedal control data are presented in enclosure (1), figures 4 & 5. The directional pedals had no trim system but incorporated a viscous damper to provide pedal displacement rate damping. Force to overcome friction was a light 3 pounds in both directions. To displace the pedals beyond 1/2" required only 5 pounds of force to the right and 4 1/2 pounds to the left. Limit control force to the right was a light 7 pounds, to the left was a light 6 1/2 pounds. Directional pedal damping barely limited pedal displacement rates, which yielded an average 3.45 in./second (73% of full travel in 1 sec) for an applied force of 30 pounds. During in-flight maneuvers above 50 KCAS requiring large collective inputs, a pilot induced yawing oscillation was encountered while attempting to maintain heading throughout the collective application. Practice autorotations, entered from 100 KCAS into 10 knot winds with full collective reduction in less than 2 seconds, caused a pilot induced yawing oscillation due to light pedal forces of +/- 3 to 5 degrees and balance ball oscillations of +/- 1 ball width every one to two seconds (PIO 4, HQR 7). The yaw PIO generated an out-of-balanced flight condition that increased rate of descent by approximately 500 FPM. During actual engine out autorotations, the yaw PIO encountered with large rapid collective displacement will increase the Scout pilot's rate of descent, decreasing the autorotational glide range and limiting the safe autorotational landing zones available. The yaw PIO due to light pedal forces encountered with large rapid collective displacements is a Part III deficiency that should be avoided in future designs. The directional pedal mechanical

characteristics met the requirements of Military Specification 8501A, with the following exception:

Paragraph 3.3.10: In that the directional pedals did not exhibit positive self-centering characteristics.

H.15 COLLECTIVE LEVER CONTROL

The collective lever control was evaluated with the OAT knob on the collective intermediate pitch stop set at 30 degrees (min setting). Collective mechanical characteristics are presented in table H-2

Table H-2

MECHANICAL CHARACTERISTICS
(COLLECTIVE LEVER CONTROL)

COLLECTIVE POSITION	RANGE OF DISPLACEMENT -INCHES/% FROM FULL DOWN	FORCE TO OVERCOME FRICTION - LBS	LIMIT CONTROL FORCE - LBS
FULL DOWN TO IPS (1)	7"/ 70 %	2 UP/ 2 DOWN	4 UP/ 2 1/2 DOWN
IPS TO MPS (UP) MPS TO IPS (DOWN) (2)	8"/ 80%	7 UP/ 3 DOWN	-- (3)
OPTIONAL AUTO RANGE	10"/ 100%	15 UP/ 0 DOWN (4)	20 UP/ O DOWN

Notes: 1. IPS- Intermediate pitch stop.

2. MPS- Maximum pitch stop.

3. The force to overcome friction at the IPS is greater than the force to displace the collective the

rest of the distance up to the MPS, and was unmeasurable.

4. In optional auto range, force to raise collective was breakout plus friction force as the upward force is applied against a spring as well as system friction.

Total collective travel available in powered flight was 8 inches; in autorotational flight the full 10 inches of travel was available. While operating at or below the IPS range the collective held its position once force was removed. The collective lever control mechanical characteristics met the requirements of the specification for which it was tested.

H.2 FORWARD FLIGHT TRIMMED FLIGHT CONTROL POSITIONS

H.2.1 GENERAL

Trimmed flight control position characteristics were evaluated during level flight from 40-120 KCAS (Vh at max power available) and during climbs and descents and level turns about a trim airspeed of 100 KCAS, under the conditions presented in table F-1. Data are presented in enclosure (2), figures 1 through 3. A summary of control margins is presented in table H-3.

TABLE H-3

MINIMUM CONTROL MARGINS

CONTROL	AIRSPEED (KCAS)	FLIGHT CONDITION	MARGIN (%)	MARGIN (IN.)	LMT DIR.
LONGITUDINAL	120	LEVEL FLIGHT	33	3.8	FWD
	100	MIN PWR DESCENT	40	4.6	AFT
LATERAL	40	LEVEL FLIGHT	50	5.0	RT
	100	RIGHT STEADY LEVEL TURNS	44	4.4	LEFT
DIRECTIONAL	100	MAX PWR CLIMB W/30 DEG AOB RIGHT	22	1.0	RT
	100	MIN PWR DESCENT W/30 DEG AOB LEFT	22	1.0	LEFT

Notes: 1. Longitudinal control throw was 11 1/2 inches.

2. Lateral control throw was 10 inches.

3. Directional control throw was 4 3/4 inches.

H.2.2 RESULTS

Throughout the test envelope, lateral and longitudinal control forces could be trimmed to zero, adequate control margins existed, no excessive control displacements were required, and no control coupling was noted. Adequate control margins existed throughout the test envelope and were considered satisfactory for the Scout mission, within the scope of this test.

GAZELLE HT MK-3 HELICOPTER
SER NO: XZ 939

ENGINE/ROTOR: STATIC
ELECTRICAL POWER: EXTERNAL
HYDRAULIC POWER: EXTERNAL
SAS: ON

STICK FEEL: ON
COLLECTIVE: ANY POS
PEDALS: ANY POS

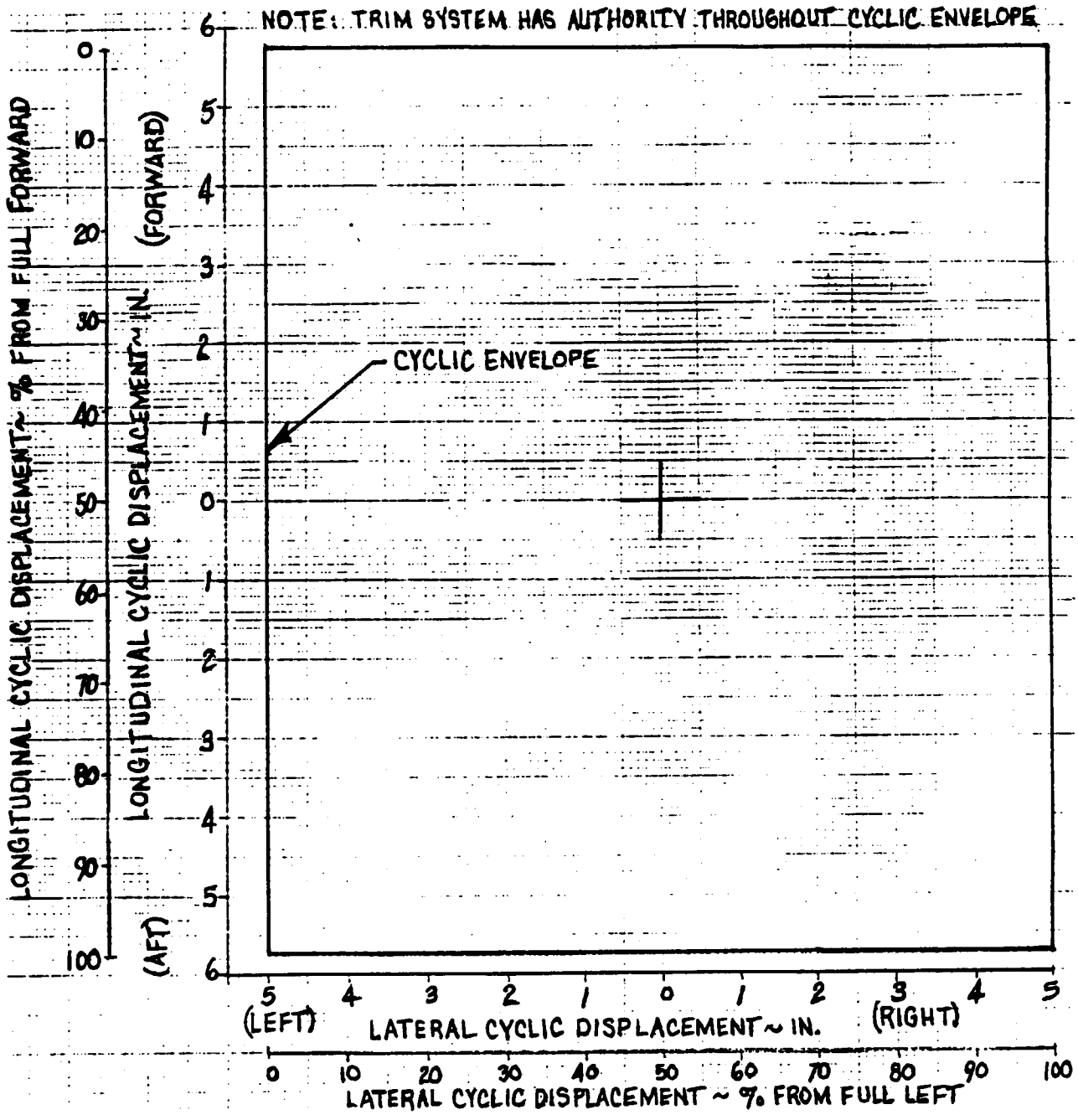


Figure 1
CYCLIC CONTROL & TRIM ENVELOPE

APPENDIX H
ENCLOSURE (1)

GAZELLE HT MK-3 HELICOPTER
SER NO: XZ 939

ENGINE/ROTOR: STATIC
ELECTRICAL POWER: EXTERNAL
HYDRAULIC POWER: EXTERNAL
SAS: ON

STICK FEEL: ON
COLLECTIVE: DOWN
PEDALS: CENTERED

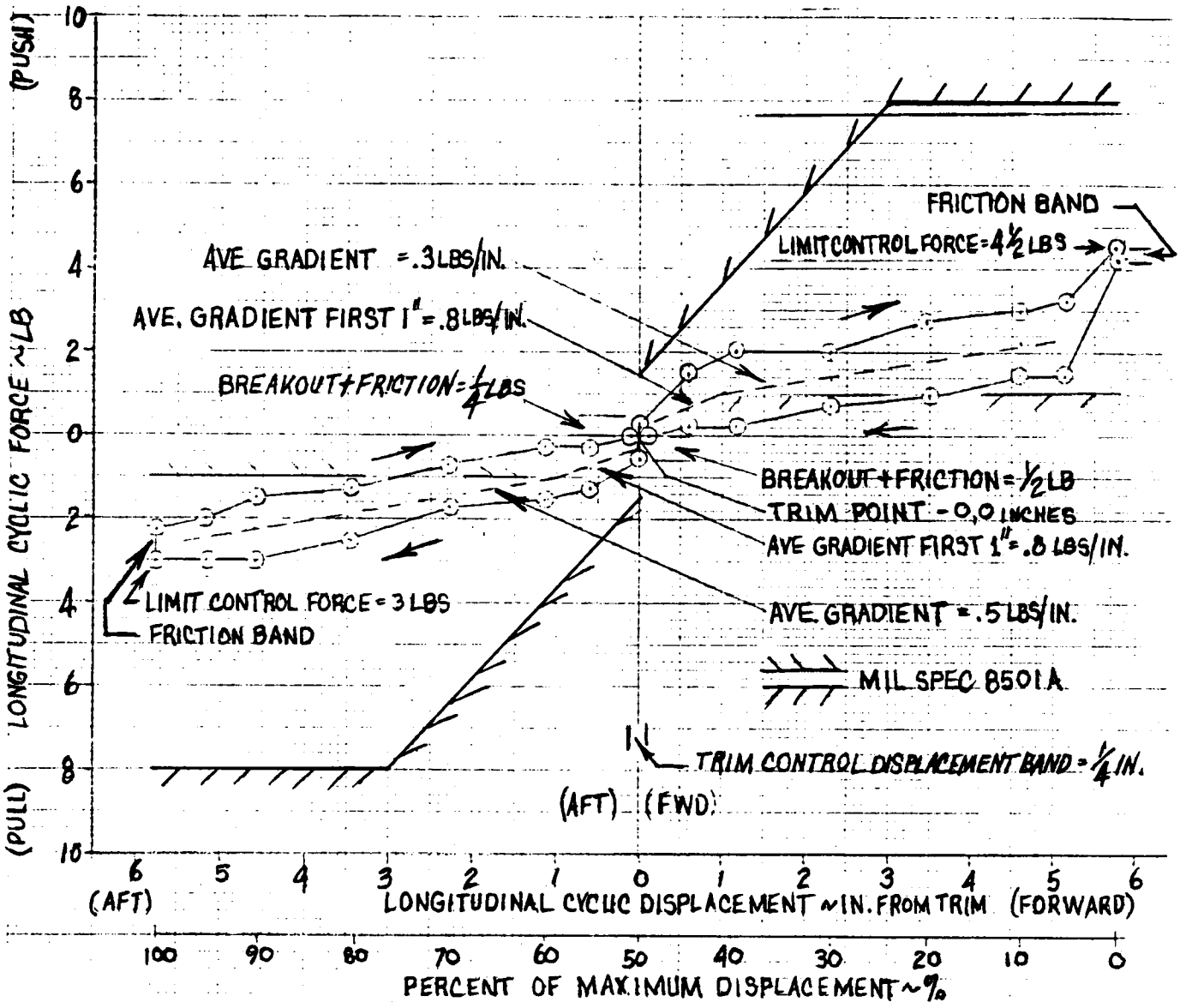


Figure 2
LONGITUDINAL CONTROL CHARACTERISTICS

APPENDIX H
ENCLOSURE (1)

GAZELLE HT MK-3 HELICOPTER
SER NO: XZ 939

ENGINE/ROTOR: STATIC
ELECTRICAL POWER: EXTERNAL
HYDRAULIC POWER: EXTERNAL

STICK FEEL: ON
COLLECTIVE: DOWN
PEDALS: CENTERED

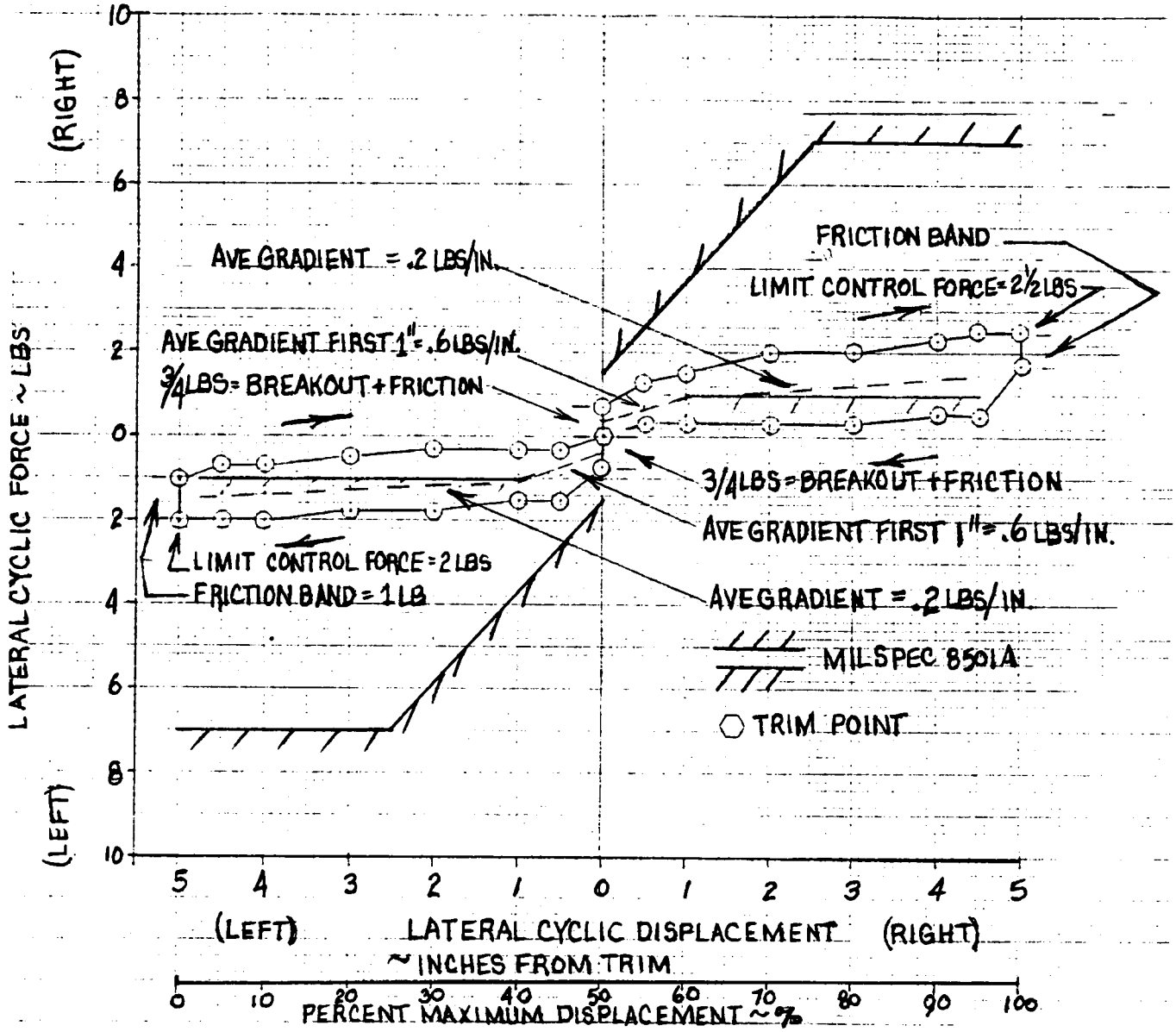


Figure 3
LATERAL CONTROL CHARACTERISTICS

APPENDIX H
ENCLOSURE (1)

GAZELLE HT MK-3 HELICOPTER
SER NO:XZ 939

ENGINE/ROTOR: STATIC
ELECTRICAL POWER: EXTERNAL
HYDRAULIC POWER: EXTERNAL
METHOD: RATE OF DISPLACEMENT-2%/SEC

SAS: ON
STICK FEEL: ON
COLLECTIVE: DOWN
CYCLIC: CENTERED

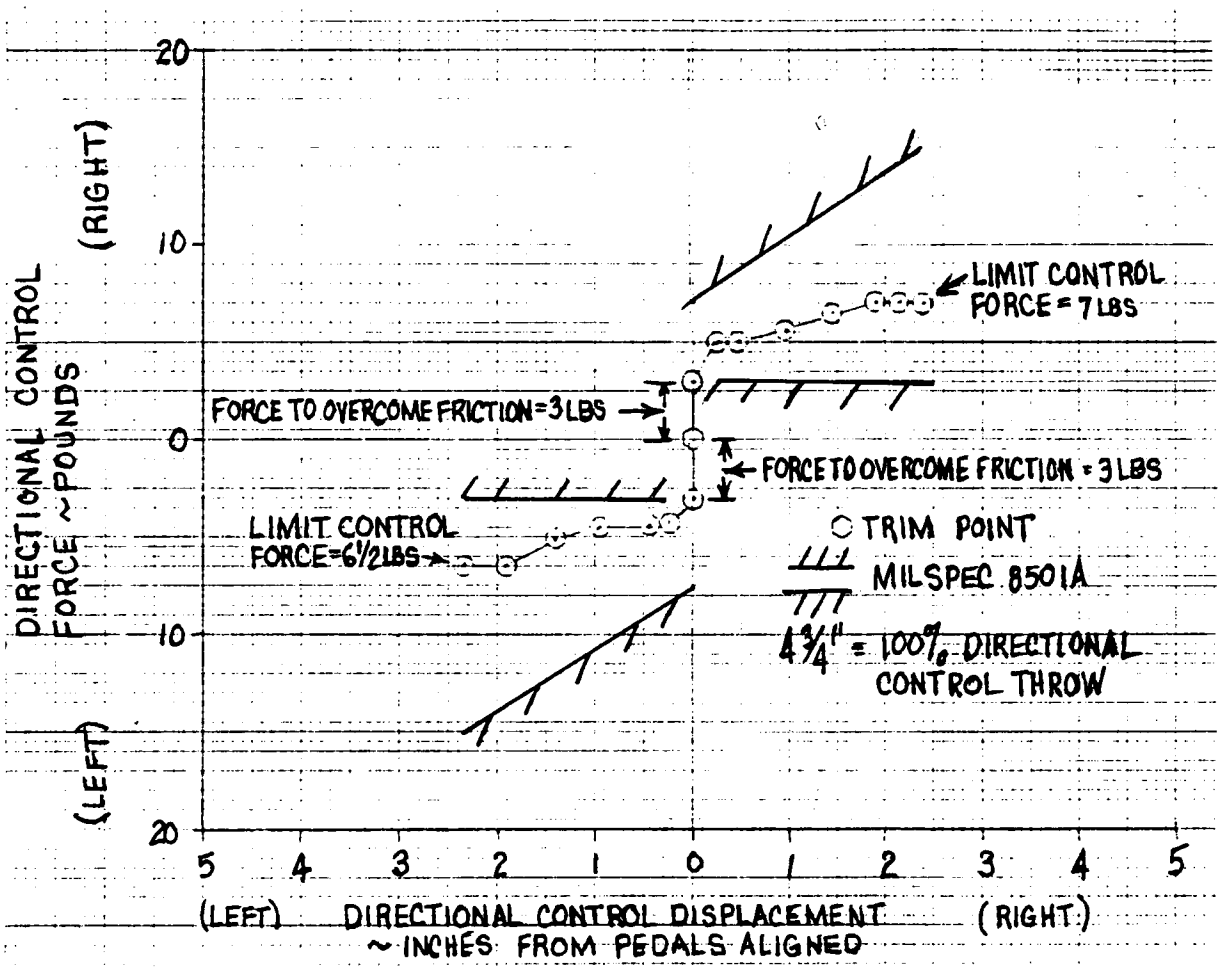


Figure 4
DIRECTIONAL CONTROL CHARACTERISTICS

APPENDIX H
ENCLOSURE (1)

GAZELLE HT MK-3 HELICOPTER
SER NO:XZ 939

ENGINE/ROTOR: STATIC
ELECTRICAL POWER: EXTERNAL
HYDRAULIC POWER: EXTERNAL
METHOD: CONSTANT FORCE APPLIED AS STEP INPUT MEASURED OVER TIME

SAS: ON
STICK FEEL: ON
COLLECTIVE: DOWN
CYCLIC: CENTERED

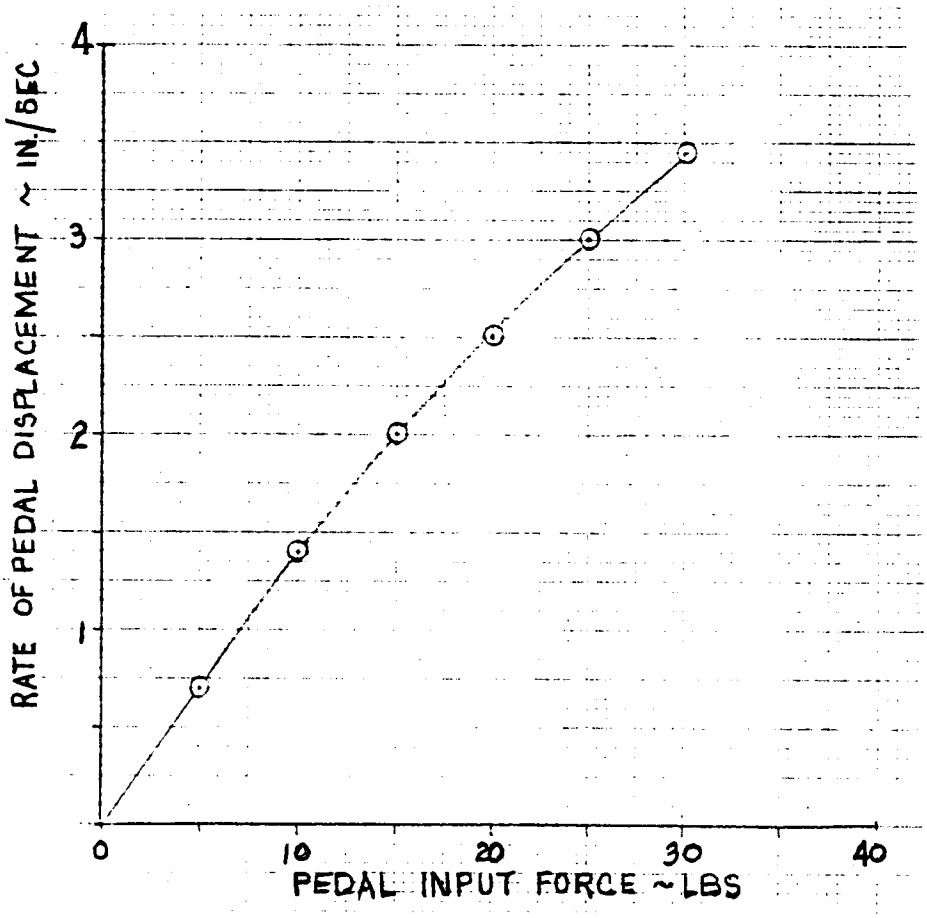


Figure 5
DIRECTIONAL CONTROL DAMPING

APPENDIX H
ENCLOSURE (1)

GAZELLE HT MK-3 HELICOPTER
SER NO:XZ 939

CONFIGURATION: CLEAN, DOORS/ WINDOWS CLOSED
SAS: ON
HYDRAULICS: ON
STICK FEEL: ON

GW RANGE: 3725-3652 LBS
CG RANGE: 116.7-116.5 IN(MID)
Hp RANGE: 3100-3480 FT
OAT RANGE: 0.9 TO 0.2°C
ROTOR SPEED: 374 RPM

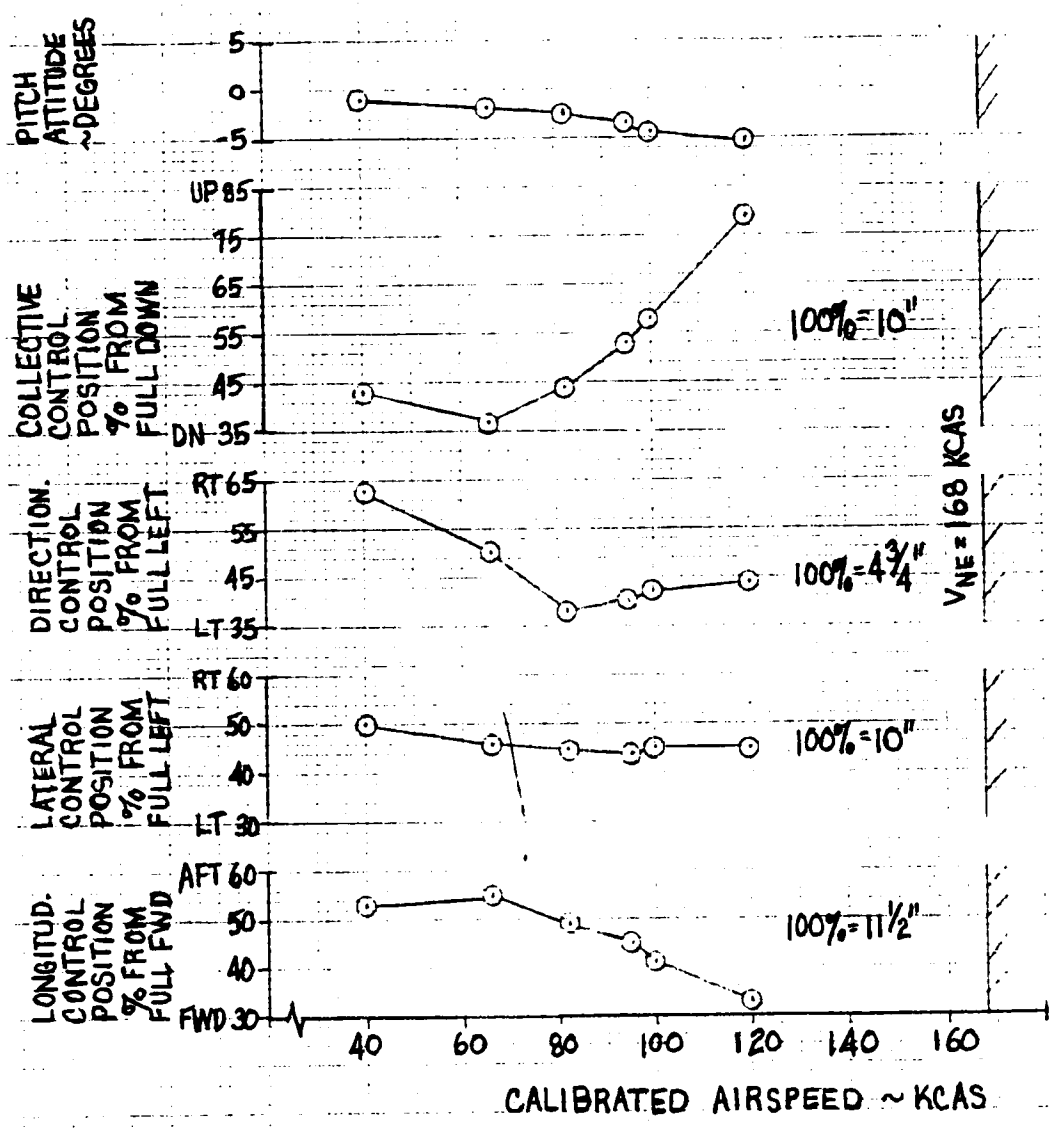


Figure 1
TRIMMED FLIGHT CONTROL POSITIONS: LEVEL FLIGHT

APPENDIX H
ENCLOSURE (2)

GAZELLE HT MK-3 HELICOPTER
SER NO:XZ 939

CONFIGURATION:	CLEAN, DOORS/ WINDOWS CLOSED	AVE GW: 3600 LBS
		AVE CG: 116.5 IN (MID)
SAS: ON		AVE Hp: 2000 FT
HYDRAULICS: ON		AVE OAT: 3.2 DEG C
STICK FEEL: ON		ROTOR SPEED: 374 RPM

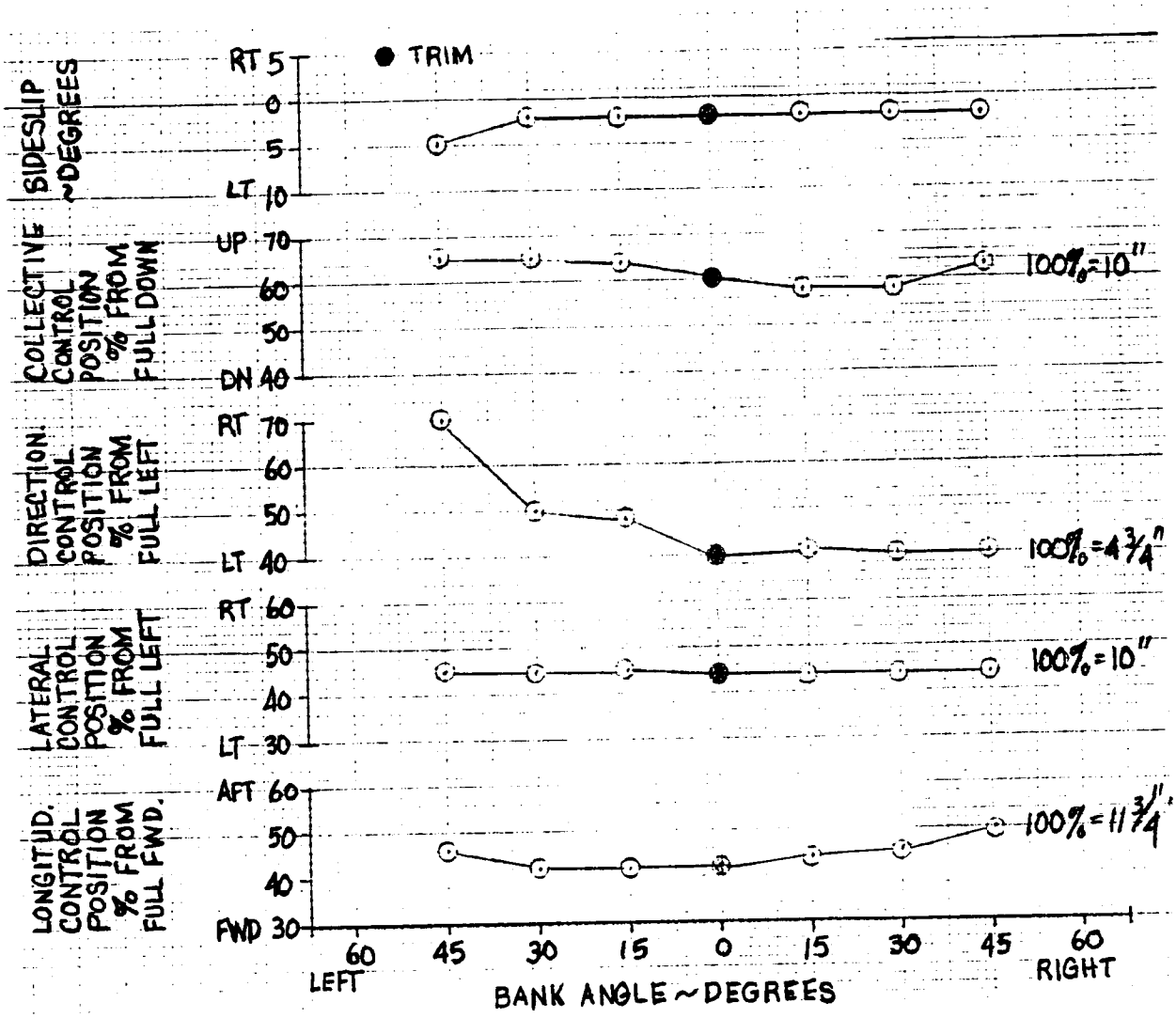


Figure 2
TRIMMED FLIGHT CONTROL POSITIONS: STEADY LEVEL TURNS
APPENDIX H
ENCLOSURE (2)

GAZELLE HT MK-3 HELICOPTER
SER NO:XA 939

CONFIGURATION:	CLEAN, DOORS/ WINDOWS CLOSED	AVE GW: 3590 LBS
		AVE CG: 116.5 IN (MID)
SAS: ON		AVE Hp: 2000
HYDRAULICS: ON		AVE OAT: 3.2
STICK FEEL: ON		ROTOR SPEED: 374 RPM

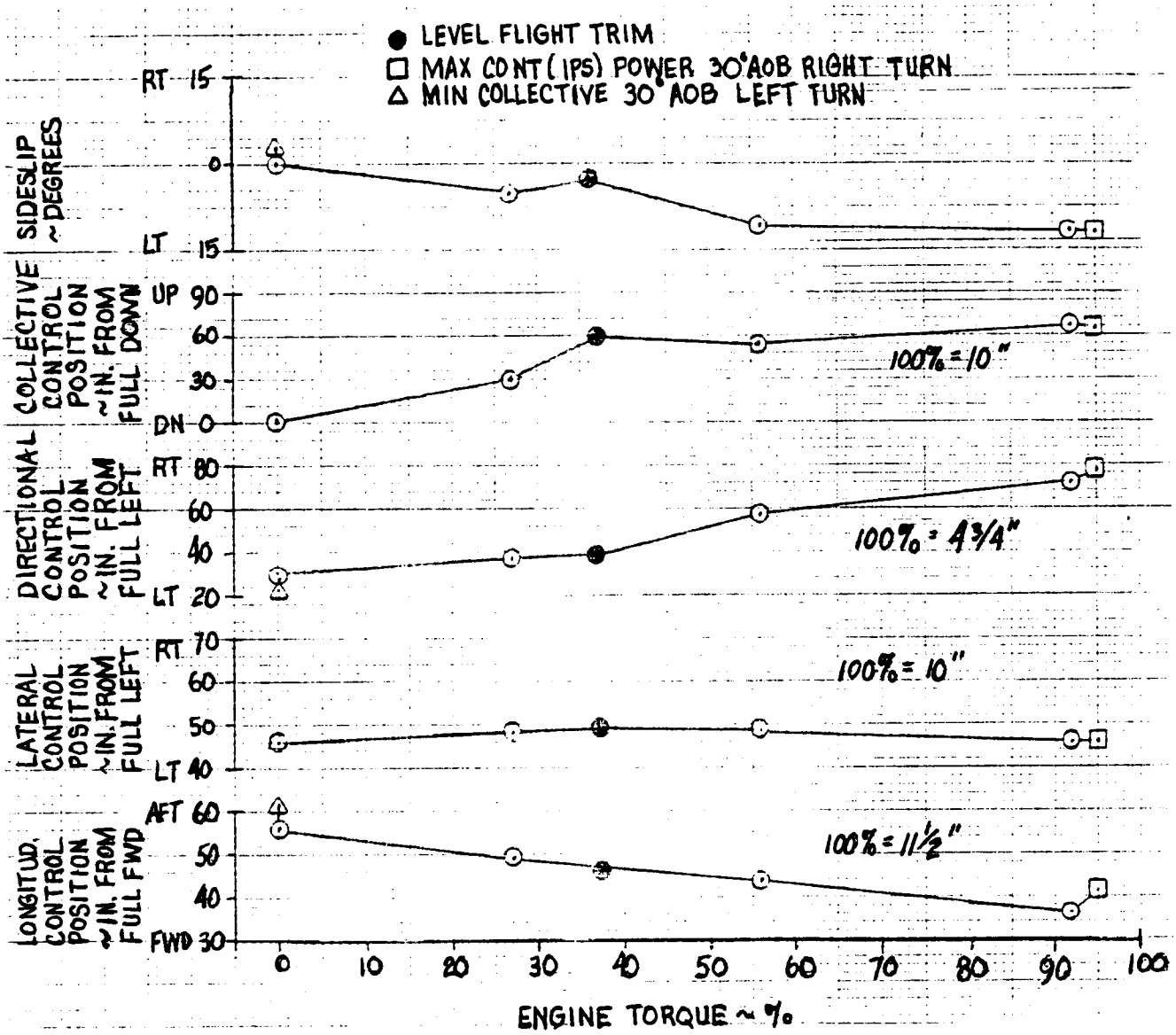
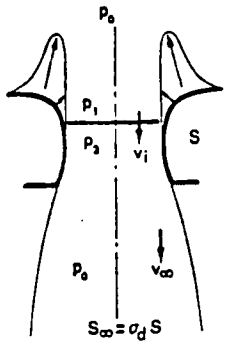


Figure 3
TRIMMED FLIGHT CONTROL POSITIONS: CLIMBS/DESCENTS

APPENDIX H
ENCLOSURE (2)

APPENDIX I: MOMENTUM THEORY & THE FENESTRON

MOMENTUM THEORY



BERNOUILLI EQ :
$$\begin{cases} P_1 + \frac{1}{2}\rho v_i^2 = P_0 \\ P_2 + \frac{1}{2}\rho v_i^2 = P_0 + \frac{1}{2}\rho v_\infty^2 \end{cases} \quad (1)$$

MASS CONSERVATION : $\rho S v_i = \rho(\sigma_d S) v_\infty$ (2)

MOMENTUM EQ : $T = T_{ROTOR} + T_{SHROUD} = (\rho S v_i) v_\infty$ (3)

ENERGY EQ : $P_i = (\rho S v_i)\frac{1}{2} v_\infty^2$ (4)

FROM : (1) $T_{ROTOR} = (p_2 - p_1) S = \frac{1}{2}\rho v_\infty^2 S$ (5)

(3,5) $\frac{T_{ROTOR}}{T} = \frac{v_\infty}{2v_i} = \frac{1}{2\sigma_d}$ (6)

(2,3) $v_i = \sqrt{\frac{\sigma_d T}{\rho S}}$ (7)

(4, 5, 6) $P_i = T_{ROTOR}\, v_i = \frac{T}{2\sigma_d}\sqrt{\frac{\sigma_d T}{\rho S}} = T\sqrt{\frac{T}{4\sigma_d \rho S}}$ (8)

FIGURE OF MERIT

FENESTRON $\sigma_d = 1$

CTR $\sigma_d = \frac{1}{2}$

$$FM = \frac{\frac{T}{2\sigma_d}\sqrt{\frac{\sigma_d T}{\rho S}}}{P}$$

EQUIVALENT FENESTRON DIAMETER AT CONSTANT T / R POWER AND THRUST (ON THE CONSTANT PORTION OF THE FM CURVE)

ROTOR DIAMETER ϕ

T/R - FIN BLOCKAGE FACTOR K

T/R - M/R INTERAXIS d

$$\left[\frac{\phi' \text{ (FAN)}}{\phi \text{ (CTR)}}\right] = 1 \Big/ \sqrt{2}\left(K\frac{d'}{d}\right)^{\frac{3}{2}} \frac{FM'}{FM}$$

VITA

Keith W. Danel was born in Windber, Pennsylvania in 1960. After graduating from Windber Area High School and attending the University of Pittsburgh at Johnstown for two semesters he was selected to attend the United States Air Force Academy. He graduated in 1983 with a Bachelor of Science degree and entered the United States Marine Corps as a Second Lieutenant. After attending The Basic School at Marine Corps Base Quantico, Virginia, he entered Naval Aviator training at Pensacola, Florida in April 1984 and earned his wings in September 1985. Following flight school, First Lieutenant Danel was assigned to fly AH-1 Cobra attack helicopters at Camp Pendleton, California. He served a 5 1/2 year tour with the "Vipers" of Marine Light/Attack Helicopter Squadron 169. Following service in Desert Shield and Desert Storm, Captain Danel was assigned to the United States Naval Test Pilot School in Patuxent River, Maryland. Following graduation from TPS in November 1992, he was assigned as a test pilot to the Rotary Wing Aircraft Test Directorate where he worked on various AH-1W and UH-1N test projects and served as the Dynamic Interface Department Head of rotary wing shipboard compatibility testing. Major Danel was reassigned in May 1995, as an operations plans officer, to the First Marine Division Staff at Camp Pendleton. Major Danel is currently assigned as a student of the U.S. Marine Corps Command and Staff College and will be reassigned to Marine Light/Attack Helicopter Squadron 369 in June of 1997.