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DESIGN AND WIND TUNNEL TESTING OF 1.5 M DIAMETER MODEL ROTORS

by

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1. SCOPE

In 1974, Aerospatiale set out to develop tests on model rotors with a diameter of about 1.50 m, in a 3 m diameter circular cross-section wind tunnel, in order to study quickly and economically the effect of any given parameter on the performance and dynamic behaviour of a rotor in hover or forward flight.

Tip speeds of 190 to 230 m/sec and tip speed ratios of up to 0.45 were set as objectives in order to comply with the Mach numbers encountered on a rotor and to analyse the effects of compressibility.

Equivalent rotor disc loads were obtained by using a solidity ratio analogous to those of helicopter main rotors. Hence, a 5 cm blade chord was necessary on a four-blade rotor.

The small size of these rotors raised a number of problems. It was more difficult to ensure an adequate fatigue strength for the blade and its mounts than on a full size rotor. Moreover, high precision machining and sure checking facilities were required to reproduce the blade aerodynamic profiles accurately. Finally, the blade surface condition had to be excellent, and called for hand finishing in a specialized shop.

The first part of this paper explains how these problems were solved ; and the results of the August 1977 test campaign (the effects of tip speed and blade twist on the performance and vibratory behaviour of a rotor in hover and in forward flight) are submitted in the second part.

List of Symbols and Abbreviations

S	Rotor disc surface area
R	Rotor radius
θ_0	Pitch at 0.75 R
Ω	Rotation speed
U	Tip speed
α_a	Rotor shaft tilt with respect to vertical datum, negative in nose dive
σ	Rotor disc solidity ratio
C_T	Lift/CSU^2
C_D	Traction/CSU^2
C_Q	Power/CSU^2

2. TECHNOLOGICAL DEVELOPMENT OF TEST FACILITIES

2.1 DESCRIPTION OF THE S_2CH WIND TUNNEL

The O.N.E.R.A. (Central Aeronautical Research Office) S_2 wind tunnel at Chalais-Meudon, in which these tests were conducted, is an Eiffel type tunnel with a guided air flow. The air duct is 3 m in diameter and its upper and side panels are removable. These panels were removed for the ground run test.

The rotor was placed in the middle of the air flow and was driven by a hydraulic motor. Discrete values ($-24^{\circ} \leqslant \alpha_Q \leqslant 24^{\circ}$) were used to calculate the variation in rotor shaft tilt with the rotor stopped. Figure 1 shows the rotor in the air duct.

Total loads on the blades were measured by means of a six-component balance and a torquemeter. A rotating switch was used to transmit blade stress signals.

During the test, the operator had the following monitoring equipment at his disposal :

- Display on two 8-channel oscilloscopes of blade stress bridges and balance dynamometer analogue signals.
- Display on an alpha-numerical console, peripheral of the local T2000/20 computer, of test parameters (V, U, Λ) and aerodynamic loads (C_T/σ, C_D/σ, C_Q/σ) corrected by rotor head effects.

– Closed circuit television network to monitor the model in the air flow.

The blade stress bridge and balance dynamometer analogue signals were recorded on magnetic tape and batch processed.

2.2 ROTOR HEAD DEVELOPMENT

The first rotor head, onto which 2, 3, 4 or 6 blades could be fitted, was built in 1975 to study amongst other parameters, the effect of the number of blades on the rotor performance and dynamic behaviour.

In order to maintain simplicity, this rotor head was hinged in flapping and drag modes (plain bearing hinges), and collective pitch was set manually when the rotor was stopped, using a clinometer mounted on a support.

The tests conducted in 1975 at Marignane and later at Chalais-Meudon showed that blade stresses were excessively high with this type of hinge, and that the bearing race service-life was too short (approximately two test days).

Therefore, a new rotor head, fitted with needle bearing hinges, was built (figure 2). In order to limit the flapping weight and to allow for the installation of drag frequency adapters, the drag hinge was located nearer to the rotor axis than the flapping hinge. Pitch was still adjusted manually when the rotor was stopped. The pitch hinge was furthest away from the rotor axis.

Drag frequency adapters must be fitted on this type of rotor head to prevent ground resonance. Rotor head simplicity and space factors lead us to decide on four blades and consequently a 5 cm blade chord.
This four-blade bearing hinge head (Figure 3) is used since April 1976.

2.3 BLADE DEVELOPMENT

The first blades, which were made in 1975, had a carbon spar and a glass cloth skin. At this time, glass cloth was not pre-impregnated ; impregnation by hand resulted in significant manufacturing variations in blade twist.

A new technology was therefore adopted to make the blades used in 1976 : fiberglas roving spars and a pre-impregnated glass cloth skin.

The 1976 tests were designed to check the effects of blade twist on the performance and dynamic behaviour of a rotor. Two sets of blades, with a nominal linear twist of -8^{o} and -14^{o} respectively, were tested. The airfoil, constant along the blade spar, was a BV 23010-1-5-8. The tests, which began in April 1976, have been used to specify performance data for a rotor with a -8^{o} twist.

In hover flight, C_z/σ values of 0.15 were reached, but in forward flight, the envelope covered was limited at high parameters for $U \geqslant 210$ m/sec, because of significant stress both in the blades and the balance and of the onset of blade track instability. Moreover, for constant θ_o, α_a and Λ values, rotor lift fell considerably when the tip speed was increased.

An analysis of these problems, which was conducted during a campaign in August 1976, attributes them to unsteady behaviour of the airfoil pitching moment at high Mach numbers.

The following design modifications were introduced to try to overcome these problems :

- addition of a trailing edge tab set at 3^{o} nose-up to the airfoil definition to counterbalance its nose-down C_m value.
- stiffening of the blade in torsion to reduce blade torsion deflection (first natural torsion frequency at about 6 Ω max.).
- improved blade surface condition by reducing the mold manufacturing tolerance to 0.04 mm.
- a -3^{o} pre-drag angle in order to reduce drag static moments in the blades.

The blade manufacturing technique was modified to meet these requirements : the spars were still made from R-glass cloth but the skin consisted of 2 plies of high tensile carbon crossed at 45^{o} (figure 4).

Tests conducted in June and August 1977 justified the new design criteria ; the dynamic problems encountered in 1976 did not reappear at all.

The only problem which occurred during these tests was caused by excessive stiffness of the drag adapters, which were of the viscous-elastic type. «Edge effects» were very important, because of the small size of these adapters and they made it very difficult to assess the stiffness. The adapters were in fact stiffer than was required, and the drag stress level in the blades was so high that it was necessary to reduce the stiffness by sawing the elastomer. Cracks, which resulted from this operation, spread in the elastomer, resulting in ground resonance when the stiffness value dropped too low.

The correct stiffness value was obtained subsequent to adapter design modifications, and, last May, two rotors were tested under conditions closely resembling current aircraft flight envelopes. For these tests, the airfoil BV 23010–1-5-8 was replaced by OA 209, described in reference 1, which was designed jointly by the O.N.E.R.A. and Aerospatiale.

3. RESULTS OF THE AUGUST 1977 TEST CAMPAIGN : EFFECT OF ROTATION SPEED AND BLADE TWIST ON ROTOR PERFORMANCE AND DYNAMIC BEHAVIOUR.

3.1. Open Test Envelope

For $\Lambda \leqslant 0.35$, the open test envelope covered the standard flight envelope of a helicopter.

At high tip speed ratio values ($\Lambda \geqslant 0.4$) the tension values obtained were low since there was no cyclic pitch fore-and-aft control on this type of rotor.

According to the tip speed, the test envelope in forward flight was limited for rotor lift by drag stresses (U = 196 and 211 m/sec) or torsion stresses (U = 226 n/sec).

3.2. Hover Flight Performance

The two rotors were compared on the following three significant points :

- Blade stall level
- Compressibility effect
- Aerodynamic efficiency

3.2.1. Blade stall level

The following lift range, which was limited by torsion stresses in both cases, was obtained.

- $0.07 \leqslant C_T/\sigma \leqslant 0.12$ for the rotor with a -8^o twist
- $0.06 \leqslant C_T/\sigma \leqslant 0.13$ for the rotor with a -14^o twist

Since it was not possible to determine accurate lift limits, because of discrete variations in collective pitch which was set manually with the rotor stopped, it was considered that blade twist had no effect on the stall level of a rotor.

3.2.2. Compressibility Effect

At restricted high lift values, the beginning of blade tip drag divergence resulted in a deterioration in rotor operation and, therefore, in its figure of merit. Figure 5 shows that blade twist delays and reduces compressibility effects, since the more the blade is twisted, the smaller is the angle of attack at the blade tip (Reference 2).

3.2.3. Aerodynamic Efficiency

At given restricted lift values, the differences in power consumption by the two rotors was only significant for $C_T/\sigma \leqslant 0$ or for $C_T/\sigma \geqslant 0.065$.

The rotor with a -8^o twist was more economical for negative thrusts and the one with a -14^o twist was more economical for $C_T/\sigma \geqslant 0.065$. Figure 6 shows that for $C_T/\sigma \geqslant 0.065$ the higher the circumferential speed, then, the higher the power gains due to large twist angles: at $C_T/\sigma = 0.11$, 8% power gain for U = 226 m/sec ; 3% gain for U = 196 m/sec.

For negative thrust values, the power losses due to blade twist were unrelated to the tip speed. They reached 10 % at $C_T/\sigma = -0.055$.

3.2.4. Conclusion

Large blade twist angles delay and reduce compressibility effects in hover flight, and enable power gains with high disc load values. They are therefore advantageous for a main rotor, and more especially for a tail rotor, in hover flight.

For negative thrust values, the behaviour of a rotor with large blade twist angles is not so good, but this is of no great significance for a tail rotor since the power breakdown for a helicopter is only critical in sidewards flight to the left (positive thrust for the tail rotor).

It must be remembered that all the numerical values given in this paper were obtained on 5 cm chord blades with a BV 23010 - 1.58 airfoil, and are likely to change for other airfoils and chords.

Flight. tests have since confirmed the results obtained at Chalais; indeed, a good qualitative correspondance can be seen between the two sets of results (figure 7).

3.3. FORWARD FLIGHT PERFORMANCE

3.3.1. Effect of Tip Speed

Figure 8, which illustrates the rotor with a -14^o blade twist angle, shows that for given tip speed ratio, lift and tension values, the reduced torque only increased with tip speed if the Mach number on the advancing blade tip exceeded 0.85. This value was comparable with the drag divergence Mach number of the BV 23010-158 airfoil: 0.81 when $C_L = O$. In view of the effect of tip relief, the presence of compressibility effects was certainly due to the beginning of drag divergence at the advancing blade tip.

On the rotor with a -8^o blade twist angle, compressibility effects were felt when the Mach number at the advancing blade tip exceeded 0.82. As in hover flight, therefore, blade twist delayed the onset of compressibility effects by reducing the angle of attack on the tip of the advancing blade.

It should be pointed out that below a certain value of C_L, depending on the airfoil ($C_L = 0$ for the NACA 0012; 0.05 for the BV 23010–1.58), the drag divergence Mach number decreased as the C_L decreased. Hence, too large a blade twist angle could result in premature drag divergence of the advancing blade.

The effect of rotation speed on the aircraft performance is shown in figure 9. Of the three speeds considered, the one which constituted the best compromise between performance at high and low speeds was that corresponding to U = 211 m/sec. It must be remembered that these results were obtained from tests conducted on a 5 cm chord rotor with a BV 23010–1.58 airfoil.

3.3.2. Effect of Blade Twist

In the standard flight envelope of a helicopter, blade twist affects its performance only slightly. The blade twist angle is only important on an aircraft with a high $C_D S$ or on one which must fly at high speeds, i.e.: when the tension values to be maintained are high, as shown in figures 10 and 11.

3.3.3. Conclusion

In both hover and forward flight, the main effect of blade twist on rotor performance is to delay the onset and attenuate the consequences of compressibility effects.

In hover flight, a larger twist angle provides power gains with high disc loads and is therefore particularly advantageous on the main rotor of a crane-helicopter, and especially on a tail rotor.

In forward flight, the blade twist angle affects the helicopter performance only slightly throughout the standard flight envelope. It is only when high tension values are to be maintained that it becomes critical.

4. ROTOR DYNAMIC BEHAVIOUR

Strain gauges were fitted on the blades and were used both to monitor the blades during testing and to examine stresses. They were bonded to the blade root to prevent damage to the airfoils. For future test campaigns, it is planned to incorporate the gauges in the airfoil, probably at the expense of reducing blade fatigue strength.

The signals from these strain gauge bridges and from the dynamometers were recorded during the tests, as shown in the example in figure 12. These signals were batch processed and could be subjected to harmonic or spectral analysis. In this way, the development of stresses at the blade root and excitation at 4 Ω on the balance, representative of an aircraft vibratory level, may be studied, with respect to the tip speed, tip speed ratio, disc loading and blade twist.

No significant effect of blade twist on the flapping stresses was recorded at the blade root. However, torsion stresses were increased considerably on a blade with a larger twist angle: stress values at 1 Ω were twice as large on the rotor with a -14° twist angle as on the one with a -8° twist angle. This could be unpractical on an aircraft since an increase in pitch control loads may reduce the service life of control linkage components and may well necessitate the installation of a double control system on light aircraft.

5. CONCLUSION

The test facilities are now fully operational and wind tunnel test programmes may begin three months after the decision to go ahead has been made.

The aim to reduce test costs, compared with those on other available facilities, has been achieved. A test campaign conducted in the SIMA Modane wind tunnel is six to eight times more expensive than one conducted in Chalais. However, it must be pointed out that much more data may be collected at Modane because of the provision for cyclic and collective pitch control from the test room. This also means that it is possible to reach higher rotor tension values. Moreover, it is easier to fit gauges onto the 4 m rotor at Modane than onto our 1.5 m model.

Rather than competing with each other, the two wind tunnel rotor test facilities at Modane and Chalais are complementary : the small scale model may be used to investigate trends quickly and economically, whereas more detailed tests may be conducted in a wider test envelope at Modane.

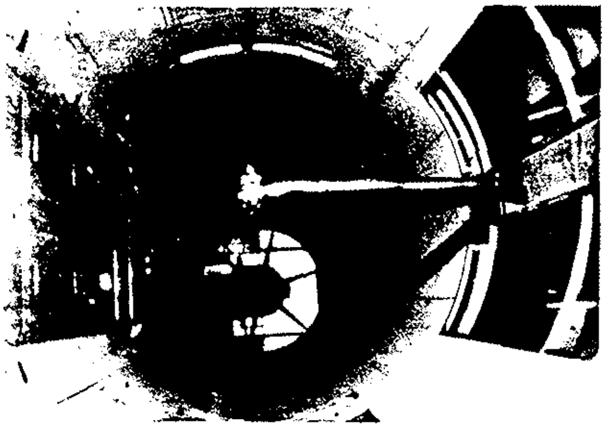

Fig. 1

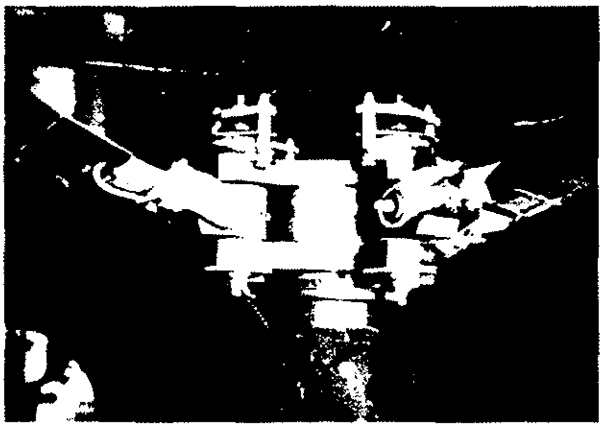

Fig. 2

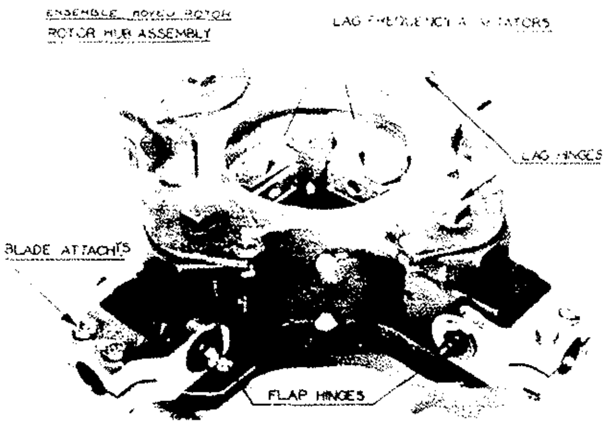

Fig. 3

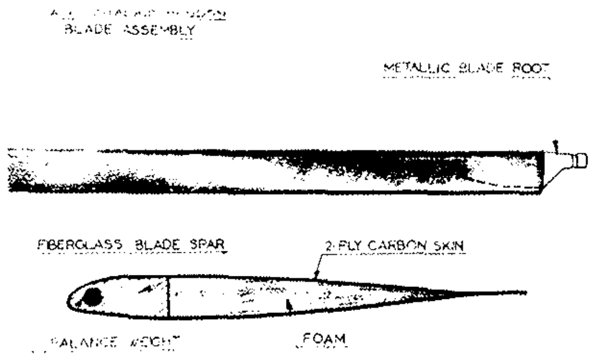

Fig. 4

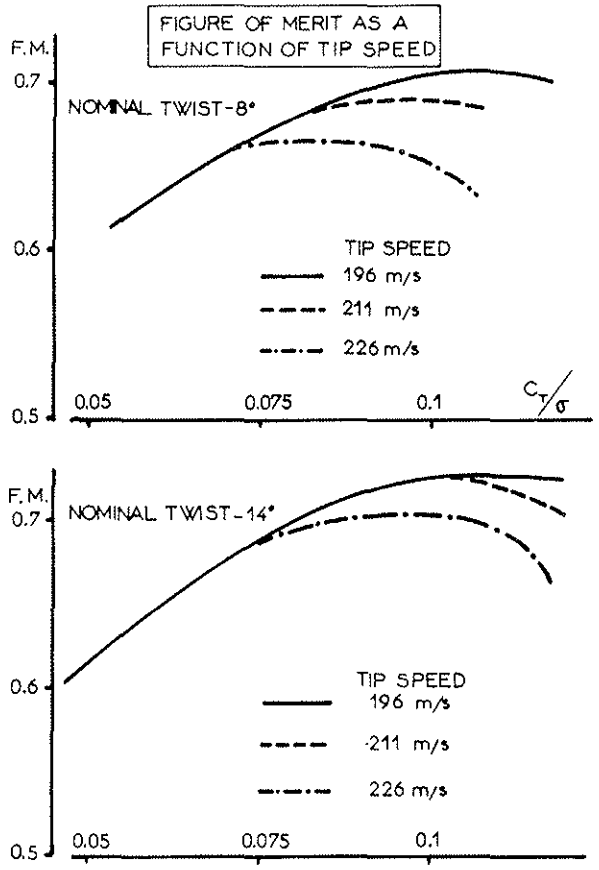

Fig. 5

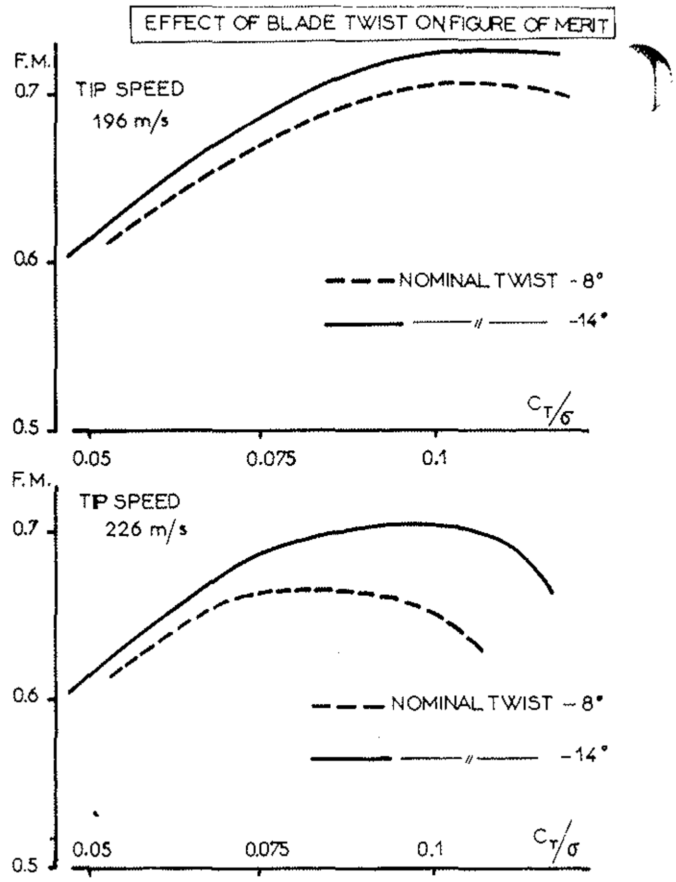

Fig. 6

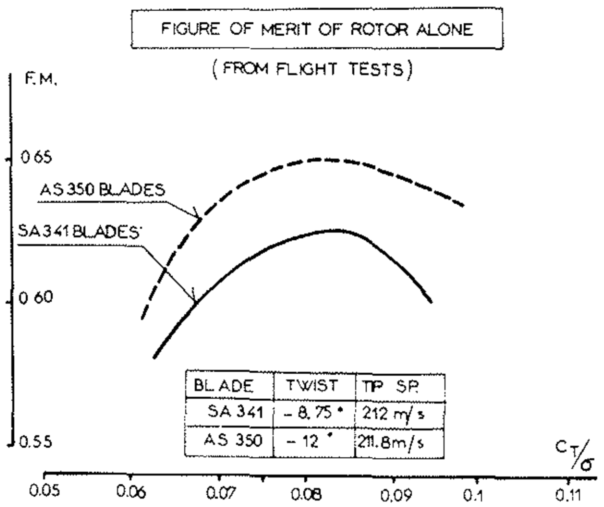

Fig. 7

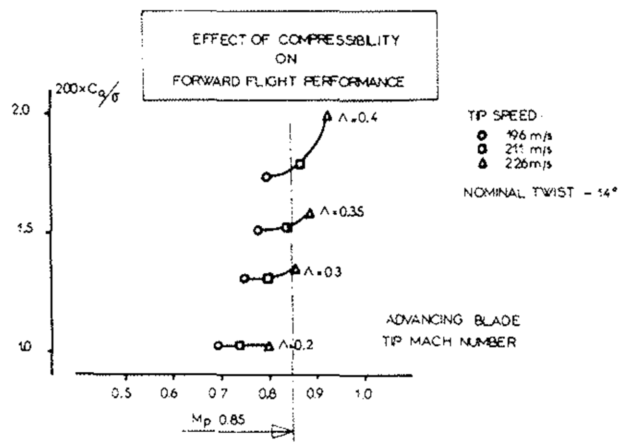

Fig. 8

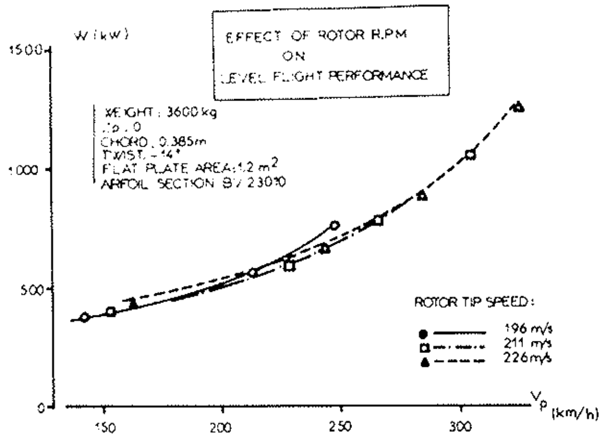

Fig. 9

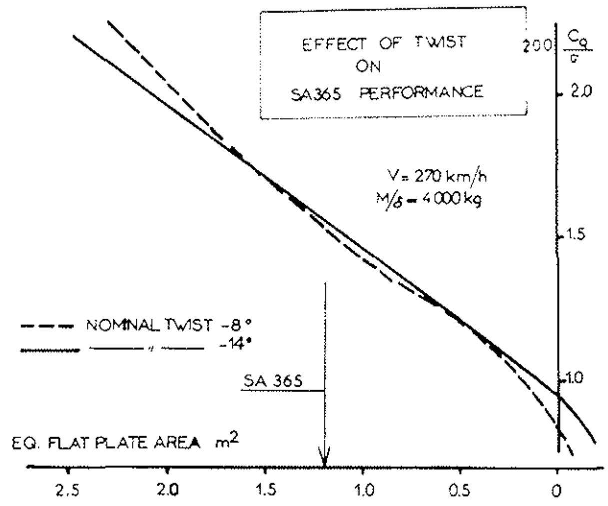

Fig. 10

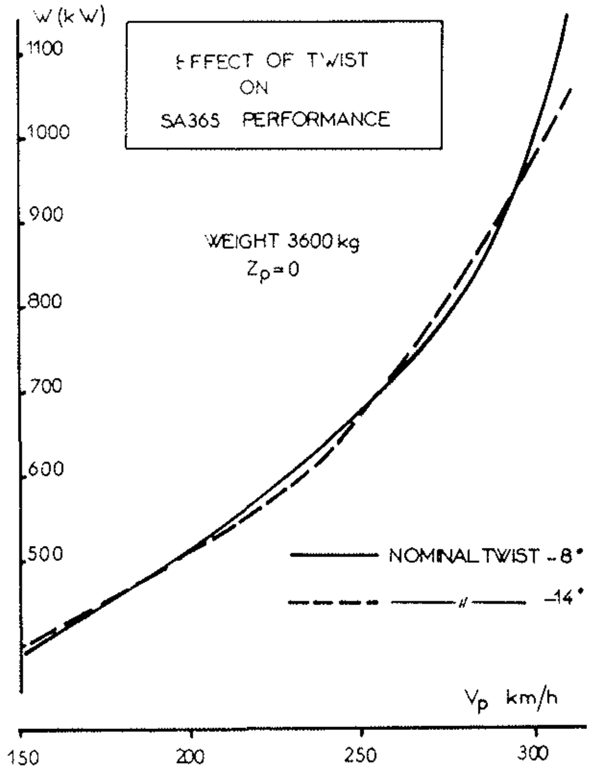

Fig. 11

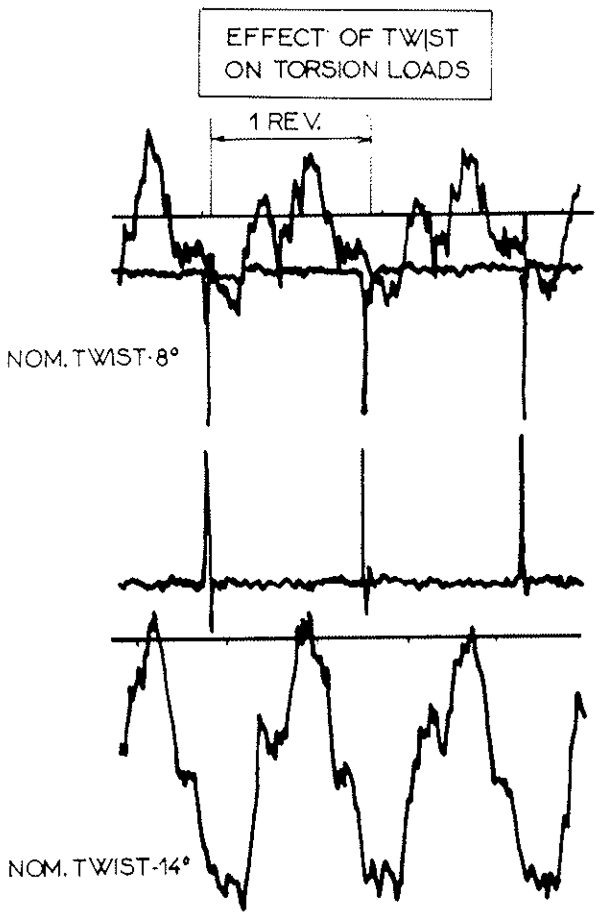

Fig. 12

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