THEORETICAL, EXPERIMENTAL AND IN-FLIGHT SPIN INVESTIGATIONS FOR AN EXECUTIVE LIGHT AIRPLANE

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Abstract

This paper describes the theoretical, experimental and in-flight spin investigations for an executive light airplane, named I-23. Spin analysis and adequate treatment to spin recovery were considered early in the design stage. The wind tunnel tests, performed on the 1:3 scaled airplane model at high angles of attack showed that there were no differences between effectiveness of the rudder alone configuration (horizontal tail removed) and that of the full configuration (including horizontal tail). Yawing moment derivative with respect to rudder deflection did not practically depend on the presence of horizontal tail unit. Flow visualization using tuft technology showed that although the flow over the vertical tail (with horizontal tail removed) at high angles of attack was well attached on the leeward side of the rudder, however simultaneously was directed span-wise around a very low aspect ratio wing, giving the extremely low side-force gradient. On the other hand, the dead-air region in the presence of the horizontal tail unit downwashes the surrounding streams and slightly increases the side force gradient. During the whole spin flight test program 265 spins have been performed. Three typical time histories of a spin entry, spin developing and spin recovery are included into this paper.

1 General Introduction

The spin is known to be a very complex and still dangerous phenomenon in aviation. Even though pilots are often trained in the basic methods of recovering from various types of spin, quite a few of them fail to win the fight with the spin they get into from time to time. This is mostly to pilot errors, but at times it is due to aircraft failures. Certain types of aircraft are incapable of recovering from some types of spin altogether due to their design, moment of inertia and aerodynamic features. Such a situation is unacceptable especially for light general-aviation airplanes, by nature flown through not very experienced pilots. And although a huge number of papers and reports devoted to spin technology have been published, there is still a number of challenges and questions which can not be easy solved and answered for an individual design project and aircraft prototype. It is mainly because the spinning motion is very complicated and involves simultaneous rolling, yawing, and pitching while airplane is at high angles of attack and sideslip. Since it involves strongly separated flows in the region beyond the stall, the aerodynamic characteristics of the airplane are very nonlinear and time dependent. Spin is not very amenable to theoretical analyses and to the best authors knowledge it is not numerically solved yet using coupled CFD/Flight Dynamics model.

The very well known three principal factors, overriding importance in the spinning of light airplanes, were carefully investigated. Among them were: relative distribution of the mass of the airplane between the wing and fuselage (Fig.1), relative airplane density and tail configuration. In setting up the tail-design requirements, the so-called tail-damping power factor was computed using the unshielded-rudder volume coefficient and the tail-damping ratio.

Dynamics of spinning has received relatively high attention through the whole aviation history, from a preliminary design of a new airplane to its in-flight tests and operation. It is especially true for aerobatic airplanes, trainers and highly maneuverable combat aircraft. However, most of general-aviation airplanes are no longer required to enter and then recover from a fully developed spin and it is the reason that spin training is no longer required for a non-professional pilot's license. Understanding of the basic principles of spinning is absolutely essential when a new design is considered and tested. Such understanding has to be based both on theoretical analysis (see e.g. Kotik [9], Toms [19]) and experimental investigations (see e.g. Bowman [3]; Weissman [20]). Theoretical analysis usually begins from the equations of motion for quasi-steady-state spin, derived for the body principal-and-central-axes system ("principal" means that we can ignore the products of inertia I_{xy}, I_{xz}, I_{yz}, "central" means that the axes system is connected to the airplane mass center). This set of equations of motion has the following form

$$
\begin{aligned}
m\left(\dot{U} + qW - rV\right) &= F_x \\
m\left(\dot{V} + rU - pW\right) &= F_y \quad , \\
m\left(\dot{W} + pV - qU\right) &= F_z
\end{aligned}
\tag{1a}
$$

$$
\begin{aligned}
\dot{p} I_x + qr(I_z - I_y) &= L_x \\
\dot{q} I_y + pr(I_x - I_z) &= M_y \quad , \\
\dot{r} I_z + pq(I_y - I_x) &= N_z
\end{aligned}
\tag{1b}
$$

where L_x, M_y, N_z are aerodynamic moments.

Because in a steady-state spin the accelerations are equal to zero

$$\dot{U} = \dot{V} = \dot{W} = 0 \quad , \tag{2}$$

so the velocity components U, V, W can be given as

$$
\begin{aligned}
U &= U_0 \cos\alpha \\
V &= -\Omega R \quad , \\
W &= U_0 \sin\alpha
\end{aligned}
\tag{3}
$$

where U_0 – velocity of descent, Ω- angular velocity of spin, R – spin radius.

It can be shown that the angular velocity vector $\boldsymbol{\Omega}$ at the steady-state spin has the following components in the body axes system

$$
\begin{aligned}
p &= \Omega \cos\alpha \cos\chi \\
q &= -\Omega \cos\alpha \sin\chi \quad , \\
r &= \Omega \sin\alpha
\end{aligned}
\tag{4}
$$

where χ denotes the angle of rotation about the z-body axis.

Substituting the above components into the steady-state equations of motion one get the following equations

$$
\begin{aligned}
F_x &= m\Omega^2 R \sin\alpha \\
F_y &= 0 \\
F_z &= -m\Omega^2 R \cos\alpha \\
L_x &= -\frac{\Omega^2}{2} \sin 2\alpha \sin\chi (I_z - I_y) \quad , \\
M_y &= -\frac{\Omega^2}{2} \sin 2\alpha \cos\chi (I_z - I_x) \\
N_z &= -\frac{\Omega^2}{2} \sin 2\chi \cos^2\alpha (I_y - I_x)
\end{aligned}
\tag{5}
$$

which establish the relations between aerodynamic forces and moments (F_x, F_y, F_z, L_x, M_y, N_z) and the inertia forces and moments. It also can be shown that the first three above equations are equivalent to following equations

$$
\begin{aligned}
D &= W \\
F_y &= 0 \quad . \\
L &= m\Omega^2 R
\end{aligned}
\tag{6}
$$

In the developed spin the attitude and some angles are repeatable from turn to turn. An example of an airplane spinning motion and the forces in spin is shown in Fig.2.

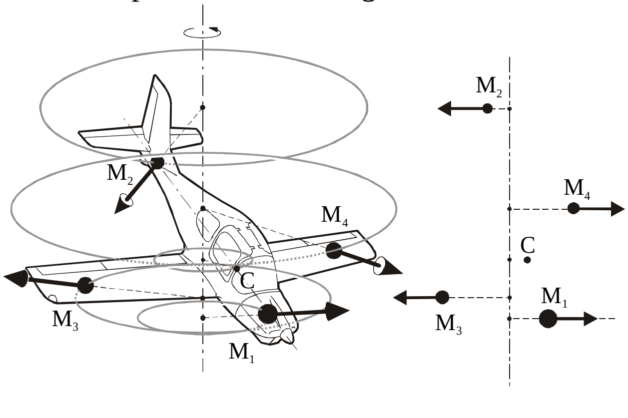
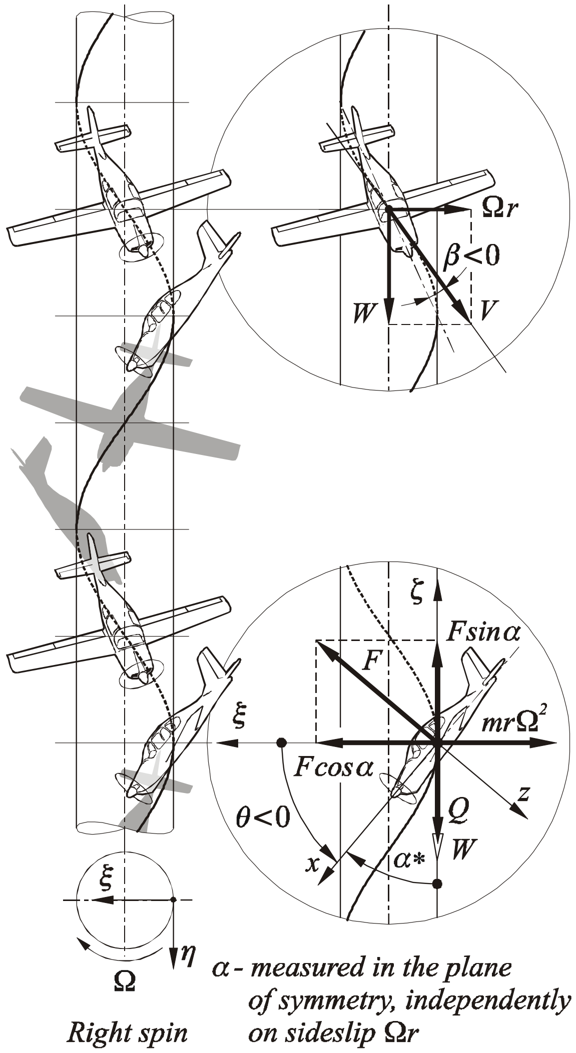


Fig.1 Airplane in spin –theoretical analysis [6])



Fif.2 Forces in a steady spin [3, 6]

For a long time in the airplane design development the aviation community was conscious of the great role of the control surfaces efficiency in the recovery from spin. In recovery the control surfaces (elevator, rudder and ailerons) have to be able to create aerodynamic moments, which can overcome the inertial moments – pitching, rolling and yawing. These aerodynamic moments created by control surfaces have to decelerate the aircraft rotation, decrease the pitch angle, decrease the angle of attack and then recover from diving vertical flight to the horizontal, level flight. Control surfaces have to be effective at high angles of attack and sideslip, which are typical in spin.

The angles of attack in spin depend first of all on characteristics of autorotation, $\omega(\alpha)$ (see e.g. Pamadi [15], Goraj [6]), moments of inertia and pitching moment $M_y(\alpha)$. Autorotation is a tendency of airplane to start the rotation about its longitudinal axis spontaneously at angles of attack beyond the stalling angle. For straight-wing light airplane the autorotation is one of principal reasons of entering into the spin. Whether an airplane develops a spin depends on the balance between the aerodynamic and inertial moments [15]. An important role in tendency of unswept wing for autorotation plays the stability derivative of rolling moment with respect to rolling velocity, C_{lp}, computed for 2-D flow around the wing section. An approximate formulae for this damping-in-roll stability derivative has the form (see e.g. Pamadi [15])

$$C_{lp,w} = -\tfrac{1}{6}(a_0 + C_D), \qquad (7)$$

where $a_0 = C_{l\alpha}$ is the sectional lift-curve slope, and C_D is the sectional drag coefficient. At low angles of attack, C_{lpw}, is negative, and for angles of attack above the stall, $C_{lp,w}$ can become positive what means that the wing is unstable in roll.

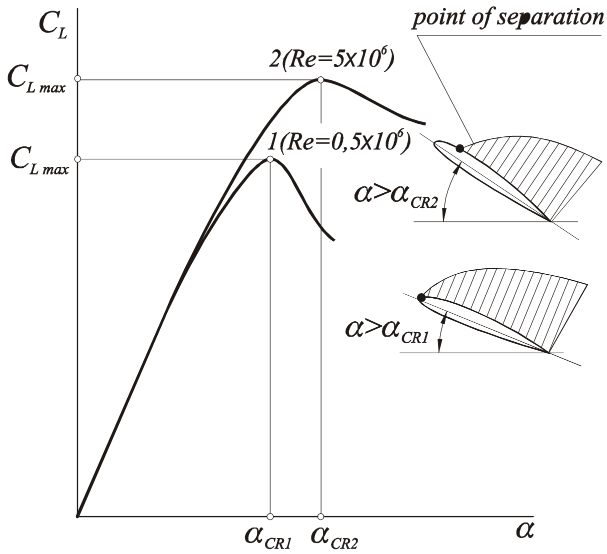


Fig.3 Lift coefficients dependent on Reynolds number [9]

The autorotational characteristics of an airplane depend on the lift curve versus angle of attack in the neighborhood of its critical angle of attack (Fig.3), which is a function of Reynolds number. The bigger is Reynolds number, the higher critical angle of attack and the airplane is more resistant to spin. Fig.4 presents a simple explanation of spin-resistant tendency (moment M_x acts in direction opposite to a disturbance p), pro-spin tendency (moment M_x acts in the same direction as a disturbance p) and a neutral-spin case ($M_x = 0$). In this figure there is also shown an influence of elevator deflection on pitching moment and a shifting of the total pitching moment due to its inertial component (this will be discussed further in the description of Fig.5).

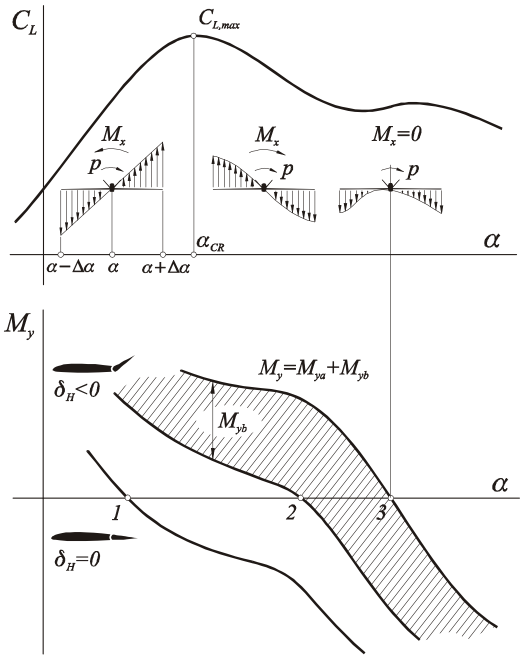


Fig. 4 Pro-spin and anti-spin tendencies, explained on the basis of the relative directions of a rolling disturbance p and the resulting rolling moment M_x

Autorotative tendency depends also on fuselage, especially on its cross-sectional shape in the plane *xy*. Fuselages with flat bottoms and sharp edges of intersection of the side walls to the fuselage bottoms generally are prone to autorotation. Much more details can be found in other textbooks and reports [15, 18].

The steady-state spin can be developed either at the angle of attack a little bit higher than the critical angle of attack (such a spin is called the steep spin) or at a really high angle of attack (such a spin is called the flat spin). Fig.5 shows pitching moments – inertial M_{yb} (proportional to $\Omega^2(I_z–I_x)$, eq.5) and aerodynamic M_{ya} (this aerodynamic pitching moment depends on the angle of attack α, elevator deflection δ_H and position of the airplane mass center x_c). Points of intersections of the curves $M_{yb}(\alpha)$ and -$M_{ya}(\alpha)$ define the angles of attack, for which inertial and aerodynamic pitching moments are in equilibrium (so, these intersection points correspond to the steady-state spins). From Fig.5 it can be concluded that the rear position of the airplane mass center (x_{cR}) decreases the angle of attack of the steep spin. However, because of a lower value of aerodynamic pitching moment and instability of equilibrium at lower angles of attack, the rear position of the airplane mass center (x_{cR}) increases the tendency for spin at higher angle of attack, i.e. for the flat spin.

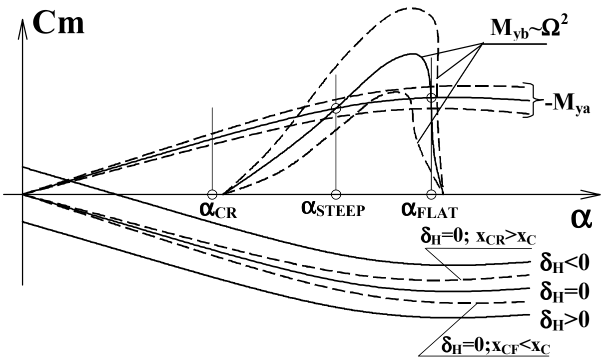


Fig.5 Inertial and aerodynamic pitching moments versus the angle of attack for different elevator deflections δ_H and positions of the airplane mass center x_c [9]

Recovery procedure from spin depends on the moment of time during of spinning, after which recovery begins. As rotation develops, the angle of attack increases until the spin is fully developed (which can be stable or unstable). In Fig.6 (see e.g. Pamadi [15], Baron & Goraj, [1]) there is shown a typical example of time history in spin. In the case under consideration about 13 s and 5 complete turns are needed for fully developed spin. According to the often-used definition the spin is fully developed if the vector of speed of the airplane mass center and the angular velocity vector are vertical and acting along the same line (it means that all spin parameters are constant). The requested effectiveness of the vertical tail, rudder and elevator at high angles of attack depends on spin characteristics associated with applied flying and handling qualities requirements. A very important requirement corresponds both to the initial instant of time (beginning of recovery process) and the time of this recovery process.

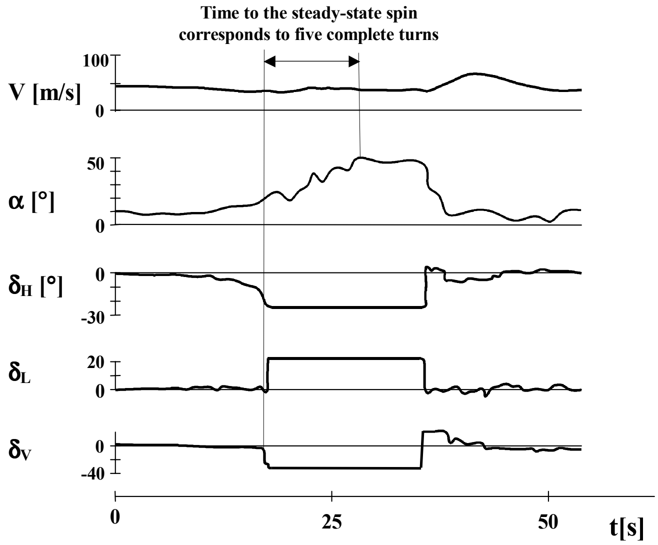


Fig.6 Selected parameters in spin [1,15]

2 Evaluation of the Control Surfaces Effectiveness at the Preliminary Design Stage

Basing on spin-tunnel tests [14] some criteria and design requirements, ensuring anti-spin characteristics, have been developed [13,14]. The results are usually presented in the form of the so-called "Tail Damping Power Factor", TDPF, the function of inertia yawing-moment parameter $(I_{xx}-I_{yy})/mb^2$ and relative density of airplane $\mu = m/\rho Sb$. In this paper such a graph for I-23 airplane (Fig.7) is also presented. This graph, shown in Fig.8, presents the areas of parameters where the recovery either by reversal rudder alone (δ_v) or by reversal rudder and elevator ($\delta_v+\delta_H$) are either satisfactory or unsatisfactory. The area in Fig.8 (being the set of pairs of points: TDPF and the inertia yawing-moment parameter) is divided into sub-areas of permissible values of TDPF, for which the successful recovery is guaranteed. A method of determining of the "Tail Damping Power Factor", TDPF, is presented in Fig.9 [3]. This method is based on the assumption that in spin (carried out at the overcritical angles of attack) arises an area (the so-called shielded area), located in the wake of horizontal tailplane, where the rudder loses its effectiveness. The higher is the angle of attack, the bigger is the shielded area.

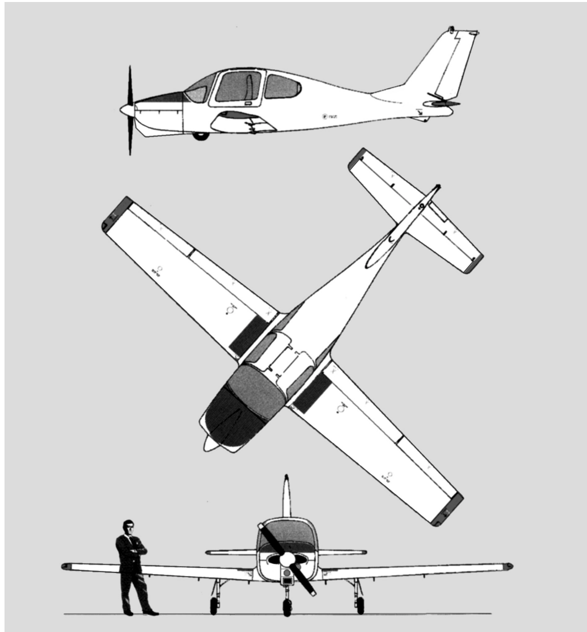


Fig.7 Personal & Business aircraft I-23 "Manager" – main views

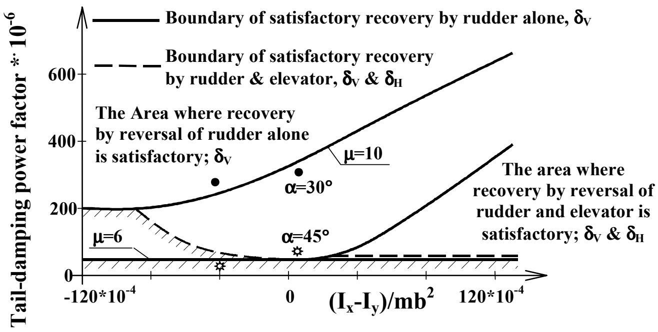


*Fig.8. Areas of satisfactory and unsatisfactory spin recoveries for light military airplanes [1] (for I-23 airplane the relative density μ= 10÷12. A symbolic location of I-23 airplane at the above figure is marked for two different angles of attack: • - for angle of attack α_h = 30^o and * - for angle of attack α_h = 45^o)*

The nature and extend of shielding of the vertical tail and rudder surfaces also depend on the relative displacement of the horizontal tail with respect to the vertical tail. The part of body and part of vertical surface below the horizontal tail contribute to the yawing moment developed by the fuselage. The unshielded part of rudder (including the part below horizontal tailplane) contributes to the rudder effectiveness in generating of yawing moments. For a satisfactory rudder effectiveness the empennage geometry should be chosen to fulfill at least partly the condition of "unshielding". It means that at the angle of attack corresponding to the spin recovery (and resulted from requirements defined by certification authorities), a part of rudder should be placed outside of the wake.

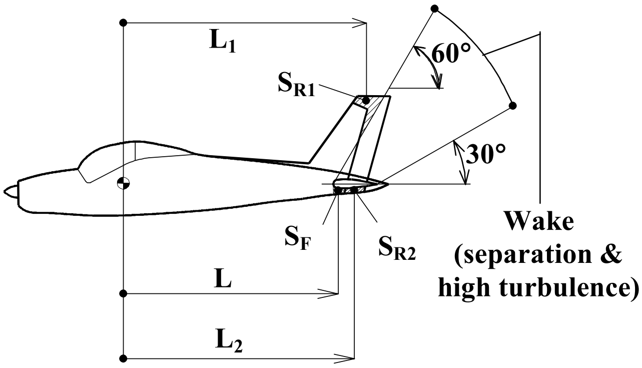


Fig.9 Symbols used in computing of the Tail Damping Power Factor, TDPF, [3,16,17]

The Tail Damping Power Factor, TDPF, is defined as follows:

$$TDPF = \frac{S_{R1}L_1 + S_{R2}L_2}{S\ b/2} * \frac{S_F L^2}{S\,(b/2)^2}\ , \qquad (10)$$

where mathematical symbols are shown in Fig.9.

Designing of empennage based on the method presented above can be considered as a typical engineering approach in the early design stage. This method is widely used in the light airplane design process and is presented in details in a number of university textbooks [16,17].

3 Investigation of Control Effectiveness in the Wind-Tunnel

The aerodynamic forces and moments have been measured and recorded as functions of angles of attack in the range from 2.5^o to 52.5^o and angles of sideslip in the range from –30^o to 30^o. Measurements have been made for different airplane configurations, including flap, elevator and rudder deflections, including and excluding horizontal tailplane. On the basis of these results a number of airplane aerodynamic characteristics have been derived in the extended range of angles of attack and angles of sideslip (Fig.10). These characteristics include gradients of rolling, pitching and yawing moments versus angle of attack and sideslip:

$$\frac{\partial C_m}{\partial \delta_H}(\alpha,\beta);\frac{\partial C_n}{\partial \delta_V}(\alpha,\beta);\frac{\partial C_l}{\partial \delta_L}(\alpha,\phi); \qquad (11)$$

where: C_l, C_m, C_n – dimensionless rolling, pitching and yawing moment coefficients, respectively and; δ_H, δ_V, δ_L – dimensional control deflections (rad), respectively of elevator, rudder and ailerons.

From Fig.10 it can be seen that effectiveness of control surfaces decreases as the angle of attack increases. At high angles of attach the elevator is more effective than others surfaces. To check the influence of the horizontal tailplane location onto the rudder effectiveness a number of measurements have been performed, also with the horizontal tailplane excluded. It was expected that if the shielding of vertical tail and rudder is removed than the rudder effectiveness essentially increased. In fact, the results of measurements have shown that excluding of the horizontal tailplane did not change the effectiveness of rudder. It was found that the decreasing of rudder effectiveness versus the angle of attack was caused not only by the increasing of the rudder shielding as it could be concluded from Fig.8.

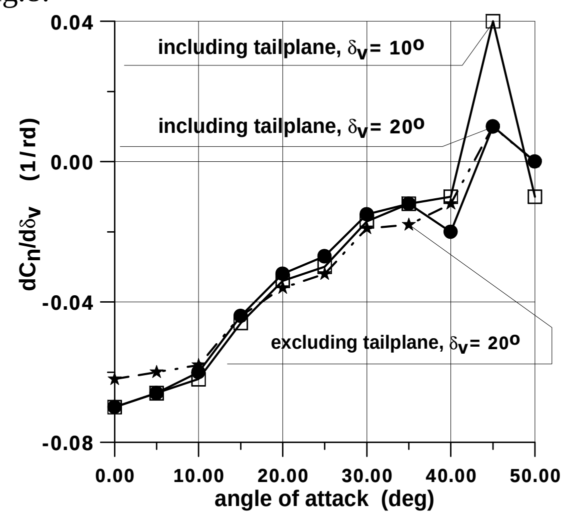


Fig.10 Effectiveness of rudder as a function of the angle of attack

To explain this phenomenon the authors have decided to perform the flow visualization in the extended range of angles of attack and sideslip. In Fig.11 the selected photos taken during these wind-tunnel tests are presented.

There are shown four airplane attitudes, corresponding to different angles of sideslip (two upper photos were taken at β=+30°, two lower photos were taken at β=-30°). The second and fourth photo (looking from the top to the bottom) shows the airplanes with the horizontal tailplane excluded. All these airplane attitudes correspond to the same angle of attack α=17.5°. In this figure the asymmetry of flow is clearly visible. There is a difference between leeward (first photo) and windward (third photo) sides of vertical stabilizer. On the windward side of vertical stabilizer the flow is fully attached and strongly deflected downwards due to downwash effect (i.e. the flow is approximately directed chord-wise). However, on the leeward side of vertical stabilizer, independently of the existence or inexistence of the horizontal tailplane, the flow is strongly turbulent, separated and undeflected downwards (i.e. the flow is directed approximately span-wise). It is the reason that the side-force on the vertical stabilizer does not appreciably change and as a consequence the rolling, pitching and yawing moments also do not change as the horizontal tailplane is removed. This phenomenon intensifies as the angle of attack increases, i.e. at high angles of attack the flow direction become more span-wise. It means that wing aspect ratio decreases from approximately of 1.5 to 0.5 and as a consequence it diminishes both the lateral-force coefficient and its effectiveness. If the horizontal tail is present then wake is generated, however the downwash is greater (the effect of downwash is positive because flow is directed more chord-wise). If the horizontal tailplane is absent then the wake behind tailplane is not generated, however the flow is not deflected downwards and remains directed span-wise. The total difference between both the cases is almost negligible.

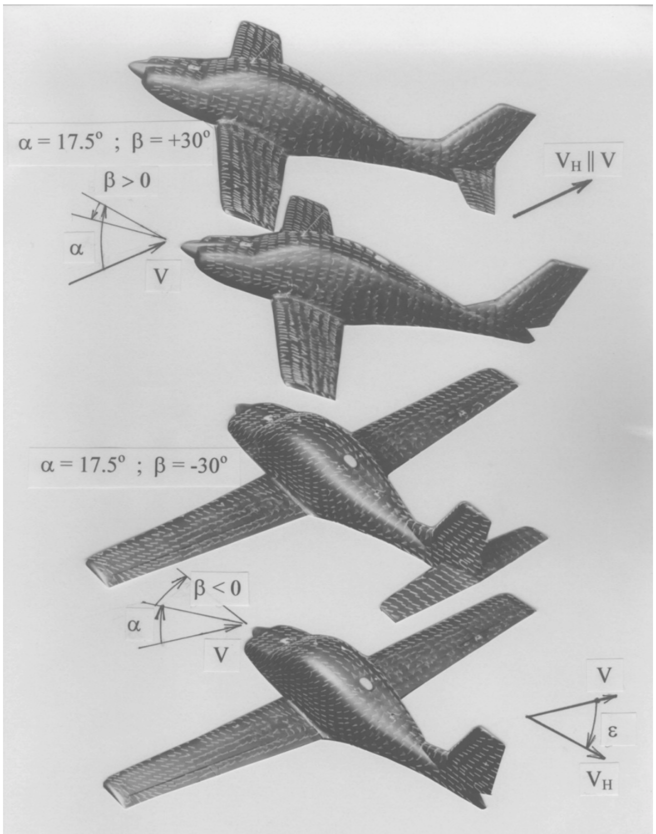


Fig.11.Flow visualization of the I-23 airplane [5,8]. Two upper attitudes of the airplane correspond to the angle of sideslip β=+30°, two lower attitudes correspond to the angle of sideslip β=-30°

4 Flight Tests

Spin flight test program for the I-23 airplane has been performed at the Institute of Aviation basing on the recommendations of the „Flight Test Guide For Certification of JAR.23 Airplanes (Section 23.221: Spinning [7])". According to these recommendations the aircraft has been tested [2] in the full scope of weight and the center of gravity (C.G.) envelope, for all possible configurations (flaps up, flaps landing, flaps take-off, gear down, gear up, full power, power reduced to idle, etc.). Normal and abnormal control usage during recovery from spin have been tested. During the whole spin flight test program 264 spins have been performed. For the reason of safety the spin recovery parachute was installed. The anti-spin system was carefully developed and assembled. This system consists of anti-spin parachute (placed at a special chute at rear part of fuselage), control system for deployment and a special protection system against the unwanted deployment. The whole anti-spin system was carefully tested to determine its structural integrity, reliability and susceptibility to inadvertent or unwanted deployment. System was assembled and tested, both on ground and in flight, in view its functionality and strength, NASA reports [10,11,12].

Fig.12 Anti-spin parachute placed at the chute (rear part of fuselage)

Fig.13 Cockpit of the Manager Aircraft. In emergency the black ring handle can deploy the auxiliary parachute, the red handle (right and above of the black ring handle) after recovery allows throwing away the main parachute

Airplane has been equipped with measuring apparatus, enabling data acquisition by the continuous recording. Among different recorded flight parameters there were calibrated airspeed, altitude, manifold pressure, shaft angular velocity, accelerations (measured in the center of gravity), control surface deflections and others. Moreover, a number of extensometers were stuck at the neuralgic points of airplane structure. These extensometers made it possible to monitor the structure loads and vibrations.

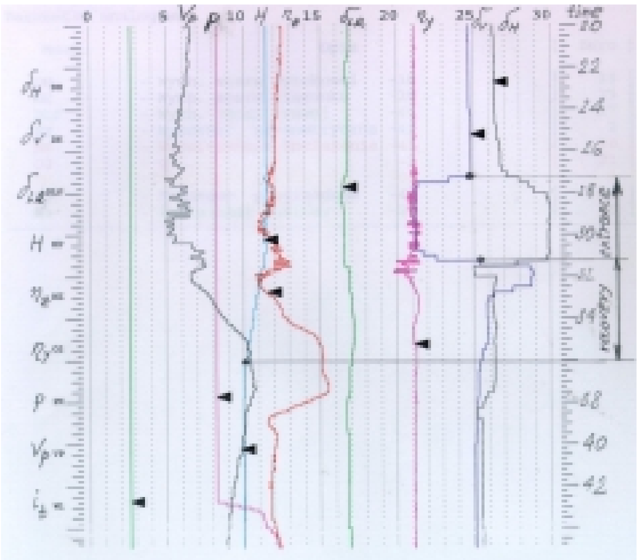


Fig.14 Flight analog data recorded at entry and recovery from one-turn spin. These data correspond to the following flight conditions: airplane weight = 1045 kg, position of the center of gravity = 25.6 % of MAC, flaps up, gear up, power reduced to 0.75 % of maximum power, entry from left turn, normal control usage (recovery ~ 5 s).

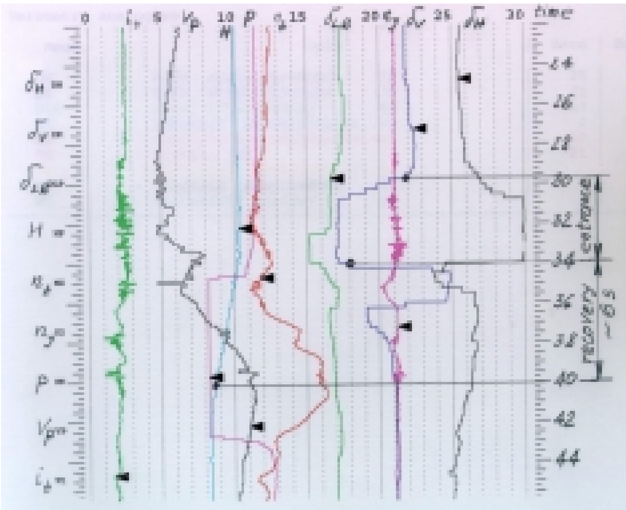


Fig.15 Flight conditions: weight = 1150 kg, position of the center of gravity = 31 % of MAC, flaps up, gear up, power reduced to 0.75 % of maximum power, entry from left turn, abnormal control usage, ailerons deflected (5^o) with the spin (recovery ~ 6 s).

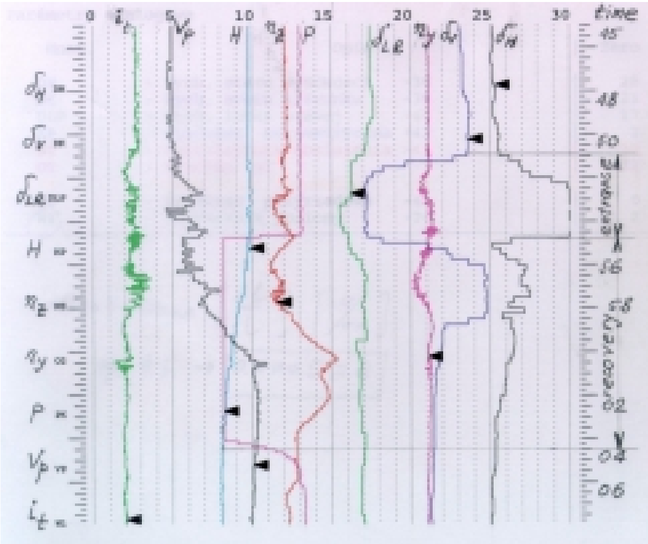


Fig.16 Flight conditions: airplane weight = 1150 kg, position of the center of gravity = 31 % of MAC, flaps up, gear up, power reduced to 0.75 % of maximum power, entry from left turn, abnormal control usage, ailerons deflected (8.5°) with the spin (flat spin, recovery ~ 9 s).

Description of the parameters recorded in Fig.14-16, including location of zero and corresponding scales and units, are placed in Tab.1.

Tab.1 Description of recorded parameters

Description	Location of zero	X (scale)*	Units
Elevator deflection, δ_H	25	5	deg
Rudder deflection, δ_V	21	5	deg
Altitude, H	2	200	m
Normal load coefficient, n_z	10	0.5	-
Air-speed, V_p	0	20	km/h
Right aileron deflection δ_{LR}	17	5	deg
Side load coefficient n_y	13	0.5	deg
Manifold pressure	5	100	hPa

* 1 degree (distance between the neighboring dotted lines) is equal to x units in SI System

5 Results of Investigation

The results can be summarized as follows: Entering into spin is not easy, independently of the considered configuration. It is caused by a relatively flat lift curve C_L above the critical angle of attack. For entering one has to reduce the speed to its minimum value and then abruptly deflect the rudder up to its limit. Recovery from spin is easy, independently of the configuration. One full turn lasts approximately 5÷6 s, loss of height is about 300 m and flight parameters (speed and load coefficients) do not exceed the admissible parameters. The recovery is possible at the time of one-turn.

Spin angle (the angle between body axis and a horizontal plane) measured by the end of the first turn was steep or very steep (>50°). Full throttle flattens the spin. The fully flat spin occurs only if the control is abnormal, throttle is fully opened and ailerons are deflected with spin. However, the time of recovery from the spin was not longer than that from the steep spin. It means that even in a flat spin the control surfaces are fully effective.

Change of the center of gravity within the range under consideration (20÷31 % of MAC) does not apparently influence on the measured flight parameters. Only for the rear extreme location of the center of gravity (flaps up, gear up configuration) a tendency towards spin flattening was observed. For all other configurations (flaps down, gear down) the spin was steep again, independently on the location of the center of gravity.

6 Verbal Evaluation Made by Test Pilot

"Independently on the spin entry, airplane mass and the aerodynamic configuration, recovery from spin is typical and easy to do for an average pilot. The airplane fulfils the requirements of regulations JAR.23.221 (a) and (b)".

7 Conclusions

Spin in-flight tests have shown that the method of evaluation of the control effectiveness, first published in forties in NACA reports TN-1045 (Neihouse & Lichtenstein, [14]) and TN-1329 (Neihouse, [13]) and after that advised to follow by other authors [16,17], is not very precise and authoritative for the evaluation of recovery from one-turn spin. Preliminary results obtained on the basis of NACA reports TN-1045 [14] and TN-1329 [13] are too pessimistic (following these results one would be on the so-called “safe side”). The wind-tunnel tests, performed under the steady-state conditions, are also not authoritative for checking the effectiveness in recovery. The only authoritative and reliable method of the evaluation of the control effectiveness in recovery can be found in the spin flight tests.

8 Acknowledgement

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