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Calculation of the Flight Characteristics of the Aircraft, AN-225

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Abstract

Flight dynamics - The science of the laws of motion of aircraft under the influence of wind, gravity, and reaction forces. It is a combination of mainly three classic disciplines: solid mechanics, fluid dynamics, and mathematics. Among the wide range of problems in the dynamics of flight of great practical importance are the problems connected with the study of the steady rectilinear motion of the aircraft. The solution allows them to determine the flight characteristics of the aircraft, characterized by the range of possible speeds and heights, rate of climb, range, flight time, and so on.

Keywords: Calculation aerodynamic characteristics of the aircraft, AN-225; Thrust required and thrust available; Practical ceiling of aircraft; Building a polar flight; Flight dynamics

Introduction

Building a polar flight, making level flight at various speeds (Mach number 0.4 to 0.9) and on the same altitude, the aircraft as it passes from one polar to another, it is the flight of the aircraft polar.

From the equilibrium conditions of the lift Y_a gravity (weight) G (G = mg) in a horizontal flight:

$$C_{Y_a} = \frac{2G}{\rho SV^2} = \frac{2G}{\rho Sa^2M^2} = \frac{A}{M^2}$$

Where, $A = \frac{2G}{\rho SV^2}$ on the height and a constant weight of the aircraft,the value is constant.

All calculations are carried out in SI.

From this formula, it follows that, in a steady horizontal flight, each Mach number M complies to a specific lift coefficient $C_{Ya.}$

For the aircraft, AN-225 with turbojet engines must use the curves of required and available thrust. Calculation and construction of required thrust P_{Req} by the formula:

$$P_{Req.} = \frac{G}{K}, K = \frac{C_Y}{C_X}$$, the aerodynamic quality of the aircraft.

When determining the flight characteristics of the aircraft used by the equation of power in the projection on the axis of the trajectory of the coordinate system, considering at the same plane as the material point of variable mass. And when the aircraft stability and controllability of the calculations it is regarded as solid [1-4].

Initial data for the implementation of the research is of course work in aerodynamics,

"Calculation of aerodynamic characteristics of the aircraft AN-225", its geometrical parameters, the aerodynamic characteristics and polar cruising.

Research includes calculations, graphics and drawings, explanation and justification of the calculation of performance, the characteristics of longitudinal stability and controllability of the aircraft.

To calculate the flight characteristics of the aircraft AN-225, used the method of N.E. Zhukovsky,

A method based on the construction of curves thrust required and thrust available, which is determined by the parameters of steady flight modes.

Initial Data

Characteristics of the standard atmosphere are as follows:

Calculate thrust required and thrust available

Calculation of the algorithm:

Specifies flight height, H, m.; H=0

Specifies the number of flight Mach; M=0.3

Determine the relative density of the air; Δ (Table 1); Δ=1

Determine the density of the air; ρH (Table 1); $\rho H = 1.225 \frac{kg}{m^3}$.

Determine the ratio; K_a (Table 1); $K_a = 1$

Determine the speed of sound at a given height with a given number M, (m/s).

aH= [aH=0 *Ka]=340.28×1=340.28 m/s.

Where, $a_{H=0}$=340.28 m/s - the speed of sound at sea level H=0

Determine the flight speed (m/s); V= [M*$a_{H=0}$] =0.3×340.28=102.08 m/s.

Determine the dynamic pressure (N/m²); $q = \frac{\rho H \times V^2}{2} = \frac{1.225 \times 102.08^2}{2} = 6382 \frac{N}{m^2}$

Determine the average gross weight of the aircraft, (N).

$$G_{Average} = \left(m_0 - \frac{m_f}{2}\right) \times 10^3 \times 9.81 = \left(640 - \frac{128}{2}\right) \times 10^3 \times 9.81 = 5650560N$$

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Height, H, (km)	Relative density, Δ	Relative speed of sound, K_a	Density, $\rho H, \frac{kg}{m^3}$	$\rho * a^2 \frac{kg}{m^2 * S}$
0	1	1	1.225	141.8*10³
2	0.822	0.977	1.0067	111.3*10³
4	0.699	0.954	0.8194	86.3*10³
6	0.538	0.930	0.6602	66, 08*10³
8	0.429	0.905	0.5259	50*10³
10	0.337	0.880	0.4136	37.06*10³
11	0.298	0.867	0.3648	31.75*10³
12	0.2536	0.867	0.3156	27.48*10³
14	0.185	0.867	0.2306	20.08*10³
16	0.135	0.867	0.1654	14.4*10³
18	0.0983	0.867	0.1207	10.51*10³
20	0.0718	0.867	0.0889	7.74*10³

Table 1: Characteristics of the standard atmosphere.

Changing the ratio of the polar blade, K_a (M)	Mach number (M)	Change parasitic drag coefficient, K_{cx0} (M)	Changing the maximum lift coefficient, K_{cymax} (M)
1	0	1	1
1	0.2	1	1
1	0.4	1	1
1.09	0.6	1.03	0.94
1.16	0.7	1.2	0.89
1.27	0.8	1.5	0.81
1.4	0.9	1.9	0.73
1.6	1.0	2.0	0.65

Table 2: Odd changes in the number of M.

Take-off weight, m_0	Fuel weight, m_f	Wing area, S, m²	Take-off thrust, P_0, k N	Parasitic drag coefficient, C_{x0}	Wingspan, L_{wing}, m	Specific fuel consumption, $C_{spec}, \frac{kg}{(N*hr)}$
640	128	905	6*234	0.021	88.4	0.057

Table 3: Personal data on the aircraft AN-225.

$C_{y\ max\ cruiser}$	M_{max}	$Q_{qmaxmax}, \frac{kN}{m^2}$	C_{max}, T.off	$K_{T.off}$	$C_{y\ max\ Const}$	K_{const}
1.7	0.88	22	2.5	8.5	3.1	5.0

Table 4: Personal data on the aircraft AN-225.

where, W_0= 640 ton-takeoff weight of the aircraft;

W_f=128 ton-mass of the fuel

The coefficient of aerodynamic lift in a horizontal flight.

$$C_{y\,h.f.} = \frac{G_{Av.}}{s*q} = \frac{5650560}{905 \times 6382} = 0.98 \le 0.9 * C_{y\max cruiser} = 0.9*1.7 = 1.53$$

where,

$C_{ymax\ cruiser}$ is the maximum lift coefficient of the wing when stowed mechanization (Tables 2-4).

Determine the effective extension of the wing:

$$\lambda ef = \left[\frac{L_{wing}^2}{S} \times (1-\delta)\right] = \left[\frac{88.4^2}{905} \times (1-0.03)\right] = 8.37$$

Where, δ =0.02….0.04;

S=905 m²- wing area;

L_{wing} = 88.4 - wingspan

Determine the ratio of the blade of the polar;

$$A_0 = \frac{1}{\pi * \lambda_{ef.}} = \frac{1}{\pi * 8.37} = 0.038$$

Determine the rate of change of blade polar depending on the number of M, K_A (Table 2); K_A=1

Determine the rate of change of parasitic drag coefficient as a function of the number of M, K_{CX0} (Table 2).

K_{CX0}=1

Determine the drag coefficient in horizontal flight;

$$C_{xah.f.} = K_{CX0} \times C_{X0} + A_0 \times K_a \times C_{yh.f.}^2 = 1 \times 0.021 + 0.038 \times 0.98^2 = 0.0574$$

where,C_{x0}=0.021- parasitic drag coefficient (Table 3).

Define flight aerodynamic efficiency;

$$K = \frac{C_{y\,h.f.}}{C_{xa\,h.f.}} = \frac{0.98}{0.0574} = 17.04$$

Identify thrust required for level flight, N.

$$P_{Req.} = \frac{G_{Av}}{K} = \frac{5650560}{17.04} = 331510N$$

Determine the ratio of thrust change the number of M.

$$\xi = 1 - 0.32 * M + 0.4 * M^2 - 0.01 * M^3 =$$

$$=> 1 - (0.32 * 0.3) + (0.4 * 0.3^2) - (0.01 * 0.3^3) = 0.94$$

Determine the takeoff thrust engines (N) (Table 3).

$$P_0 = 6 * 234000 = 1404000N$$

Identify the thrust available in horizontal flight.

When,

$$H \leq 11km \dashrightarrow P_{Avail} = \xi \times \Delta^{0.85} \times P_0 = 0.94 \times 1^{0.85} \times 1404000 = 1319760N.$$

$$H > 11km \dashrightarrow P_{Avail} = \xi \times \Delta \times 1.2 \times P_0 = 0.94 \times 1 \times 1.2 \times 1404000 = 1583712N.$$

By algorithm, using a program in Excel.

The calculation results are shown in Tables 5 - 13 and in Figures 1 - 13.

"Area velocity values, at which horizontal flight is possible at a fixed weight of the aircraft and the altitude, it called horizontal flight speed range.

At this altitude:

$V_{min}=V_{most\ advantageous}=V_{max}$

At this altitude, the curve of Thrust available (or power) not intersects the curve of thrust required, but only touches it."

Determination of the Flight Range of the Aircraft H = f (v)

At the intersection point of the curve Thrust Required and Thrust Available, we define the boundaries of the possible limits of the aircraft (Table 14).

Drawing a schedule of possible aircraft flight boundaries under the conditions of thrust required and thrust available (Figures 11-13).

On the left side, should be restrictions on the minimum flight speed of the conditions for safe values of the coefficient of lift of the aircraft.

$C_{y\ safe\ values}$ = 0.9*$C_{ymaxcruiser}$ =0.9×1.7=1.53

The significance of this factor determines the minimum speed of horizontal flight of the conditions of a possible lift of an aircraft wing.

Flight altitude, км	**0**									
The number of flight Mach	0.1	0.2	0.3	0.4	0.6	0.7	0.8	0.85	0.9	0.95
Relative density of the air, Δ	1	1	1	1	1	1	1	1	1	1
The air density, ρH, kg/m³	1.225	1.225	1.225	1.225	1.225	1.225	1.225	1.225	1.225	1.225
Coefficient K_a	1	1	1	1	1	1	1	1	1	1
The speed of sound a, m/S	340.28	340.28	340.28	340.28	340.28	340.28	340.28	340.28	340.28	340.28
Flight speed , V, m/s	34.03	68.06	102.08	136.11	204.17	238.20	272.22	289.24	306.25	323.27
Dynamic pressure, N/m²	709	2837	6383	11347	25532	34752	45390	51241	57447	64007
The average gross weight of the aircraft, Gav. N	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560
The coefficient of aerodynamic lift in horizontal flight, $C_{yh.f.}$	8.80	2.20	0.98	0.55	0.24	0.18	0.14	0.12	0.11	0.10
Effective of wing extension , λ	8, 38	8.38	8, 38	8, 38	8.38	8.38	8.38	8.38	8.38	8.38
Factor Blade of the polar, A_0	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038
Coefficient K_A	1	1	1	1	1.09	1.16	1.27	1.33	1.4	1.5
Coefficient K_{CX0}	1	1	1	1	1, 03	1.2	1.5	1.7	1.9	1.95
The coefficient of aerodynamic drag in horizontal flight, C_{xa} h.f.	2.9664	0.2051	0.0574	0.0325	0.0241	0.0266	0.0324	0.0365	0.0405	0.0415
Flight aerodynamic quality, K	2.97	10.73	17.05	16.93	10.14	6.75	4.24	3.34	2.68	2.35
Thrust required $P_{required,}$ N	1903984	526541	331364	333815	557029	837300	1331466	1690318	2107042	2403497
The Rate of changes in the thrust of the number of flight Mach , ξ	0.972	0.952	0.940	0, 935	0, 950	0.969	0.995	1.011	1.029	1.048
Take-off thrust engines, $P_{0,}$ N,	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000
Thrust available, P_{avail} , N	1364674	1336496	1319381	1313245	1333575	1359872	1396812	1419246	1444309	1471990
Vertical speed, $V_{y,}$ m/s	-3.25	9.76	17.85	23.59	28.06	22.03	3.15	-13.88	-35.92	-53.29

Table 5: The calculation of thrust required and thrust available, H = 0 km.

Flight altitude, км					2					
The number of flight Mach	0.1	0.2	0.3	0.4	0.6	0.7	0.8	0.85	0.9	0.95
Relative density of the air, Δ	0.822	0.822	0.822	0.822	0, 822	0.822	0.822	0.822	0.822	0.822
The air density, $\rho H, \frac{kg}{m^3}$	1.0067	1.0067	1.0067	1.0067	1.0067	1.0067	1.0067	1.0067	1.0067	1.0067
Coefficient K_a	0.977	0.977	0.977	0.977	0.977	0.977	0.977	0.977	0.977	0.977
The speed of sound a, m/S	332.454	332.454	332.454	332.454	332.454	332.454	332.454	332.454	332.454	332.454
Flight speed , V, m/s	33.25	66.49	99.74	132.98	199.47	232.72	265.96	282.59	299.21	315.83
Dynamic pressure, N/m^2	556	2225	5007	8901	20028	27260	35605	40195	45063	50209
The average gross weight of the aircraft, $G_{Av,}$ N	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560
The coefficient of aerodynamic lift in horizontal flight, $C_{yh.f.}$	11.22	2.81	1.25	0.70	0.31	0.23	0.18	0.16	0.14	0.12
Effective of wing extension , λ	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38
Factor Blade of the polar, A_0	0.038	0.038	0.038	0.038	0.038	0, 038	0.038	0.038	0, 038	0.038
Coefficient K_A	1	1	1	1	1.09	1.16	1.27	1.33	1.4	1.5
Coefficient K_{cx0}	1	1	1	1	1.03	1.2	1.5	1.7	1.9	1, 95
The coefficient of aerodynamic drag in horizontal flight, $C_{xa\ h.f.}$	4.8078	0.3202	0.0801	0, 0397	0.0257	0.0275	0.0330	0.0369	0.0409	0.0418
Flight aerodynamic quality, K	2.33	8.76	15.57	17.67	12.15	8.32	5.32	4.21	3.39	2.97
Thrust required $P_{Required,}$ N	2420616	644803	362940	319796	465019	678749	1062836	1342999	1668846	1900779
The Rate of changes in the thrust of the number of flight Mach , ξ	0.972	0.952	0, 940	0.935	0.950	0.969	0.995	1.011	1.029	1.048
Take-off thrust engines, P_0, N	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000
Thrust available, P_{avail} , N	1155234	1131380	1116892	1111698	1128908	1151169	1182439	1201430	1222647	1246080
Vertical speed, $V_{y,}$ m/s	-7.44	5.73	13.31	18.64	23.44	19.46	5.63	-7.08	-23.63	-36.59

Table 6: The calculation of thrust required and thrust available, H = 2 km.

Flight altitude, км					4					
The number of flight Mach	0.1	0.2	0.3	0.4	0.6	0.7	0.8	0.85	0.9	0.95
Relative density of the air, Δ	0.699	0.699	0.699	0.699	0.699	0.699	0.699	0.699	0.699	0.699
The air density, $\rho H, \frac{kg}{m^3}$	0.8194	0.8194	0.8194	0.8194	0.8194	0.8194	0.8194	0.8194	0.8194	0.8194
Coefficient K_a	0.954	0.954	0.954	0.954	0.954	0.954	0.954	0.954	0.954	0.954
The speed of sound a, m/S	324.627	324.627	324.627	324.627	324.627	324.627	324.627	324.627	324.627	324.627
Flight speed , V, m/s	32.46	64.93	97.39	129.85	194.78	227.24	259.70	275.93	292.16	308.40
Dynamic pressure, N/m^2	432	1727	3886	6908	15543	21156	27632	31194	34972	38966
The average gross weight of the aircraft, G_{Av}, N	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560
The coefficient of aerodynamic lift in horizontal flight, $C_{yh.f.}$	14.46	3.62	1.61	0.90	0.40	0.30	0.23	0.20	0.18	0.16
Effective of wing extension , λ	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38
Factor Blade of the polar, A_0	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038

Coefficient K_A	1	1	1	1	1.09	1.16	1.27	1.33	1.4	1.5
Coefficient K_{cx0}	1	1	1	1	1.03	1.2	1.5	1.7	1.9	1.95
The coefficient of aerodynamic drag in horizontal flight, $C_{xa.h.f.}$	7.9686	0.5177	0.1191	0.0520	0.0283	0.0290	0.0340	0.0377	0.0416	0.0424
Flight aerodynamic quality, K	1.81	6.98	13.49	17.37	14.19	10.16	6.65	5.31	4.29	3.78
Thrust required $P_{Required,}$ N	3113632	809179	418897	325377	398284	555998	849348	1065002	1316496	1495674
The Rate of changes in the thrust of the number of flight Mach , ξ	0.972	0.952	0.940	0.935	0.950	0.969	0.995	1.011	1.029	1.048
Take-off thrust engines P_0, N	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000
Thrust available, P_{avail}, N	1006548	985764	973141	968616	983611	1003006	1030252	1046799	1065285	1085702
Vertical speed, Vy, m/s	-12.11	2.03	9.55	14.78	20.18	17.98	8.31	-0.89	-12.99	-22.38

Table 7: The calculation of thrust required and thrust available, $H = 4$ km.

Flight altitude, км	6									
The number of flight Mach	0.1	0.2	0.3	0.4	0.6	0.7	0.8	0.85	0.9	0.95
Relative density of the air, Δ	0.538	0.538	0.538	0.538	0.538	0.538	0.538	0.538	0.538	0.538
The air density, $\rho H, \frac{kg}{m^3}$	0.6602	0.6602	0.6602	0.6602	0.6602	0.6602	0.6602	0.6602	0.6602	0.6602
Coefficient K_a	0.93	0.93	0.93	0.93	0.93	0.93	0.93	0.93	0.93	0.93
The speed of sound a, m/S	316.46	316.46	316.46	316.46	316.46	316.46	316.46	316.46	316.46	316.46
Flight speed , V, m/s	31.65	63.29	94.94	126.58	189.88	221.52	253.17	268.99	284.81	300.64
Dynamic pressure, N/m^2	331	1322	2975	5289	11901	16199	21157	23885	26777	29835
The average gross weight of the aircraft, G_{Av}, N	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560
The coefficient of aerodynamic lift in horizontal flight, $C_{yh.f.}$	18.89	4.72	2.10	1.18	0.52	0.39	0.30	0.26	0.23	0.21
Effective of wing extension , λ	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38
Factor Blade of the polar, A_0	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038
Coefficient K_A	1	1	1	1	1.09	1.16	1.27	1.33	1.4	1.5
Coefficient K_{cx0}	1	1	1	1	1.03	1.2	1.5	1.7	1.9	1.95
The coefficient of aerodynamic drag in horizontal flight, $C_{xah.f.}$	13.5773	0.8683	0.1884	0.0740	0.0330	0.0317	0.0357	0.0392	0.0428	0.0434
Flight aerodynamic quality, K	1.39	5.44	11.14	15.96	15.88	12.14	8.27	6.68	5.45	4.82
Thrust required $P_{Required,}$ N	4062046	1039072	507185	354010	355765	465442	683629	846343	1037020	1173100
The Rate of changes in the thrust of the number of flight Mach , ξ	0.972	0.952	0.940	0.935	0.950	0.969	0.995	1.011	1.029	1.048
Take-off thrust engines, P_0, N	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000
Thrust available, P_{avail}, N	805738	789101	778996	775373	787377	802903	824713	837959	852756	869100
Vertical speed, Vy, m/s	-18.24	-2.80	4.57	9.44	14.50	13.23	6.32	-0.40	-9.29	-16.17

Table 8: The calculation of thrust required and thrust available, H = 6 km.

Flight altitude, км	**8**									
The number of flight Mach	0.1	0.2	0.3	0.4	0.6	0.7	0.8	0.85	0.9	0.95
Relative density of the air, Δ	0.429	0.429	0.429	0.429	0.429	0.429	0.429	0.429	0.429	0.429
The air density, ρH, kg/m³	0.5259	0.5259	0.5259	0.5259	0.5259	0.5259	0.5259	0.5259	0.5259	0.5259
Coefficient K_a	0.905	0.905	0.905	0.905	0.905	0.905	0.905	0.905	0.905	0.905
The speed of sound a, m/S	307.953	307.953	307.953	307.953	307.953	307.953	307.953	307.953	307.953	307.953
Flight speed , V, m/s	30.80	61.59	92.39	123.18	184.77	215.57	246.36	261.76	277.16	292.56
Dynamic pressure, N/m²	249	997	2244	3990	8977	12219	15960	18017	20199	22506
The average gross weight of the aircraft, G_{AV}, N	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560
The coefficient of aerodynamic lift in horizontal flight, $C_{Y\,h.f.}$	25.04	6.26	2.78	1.56	0.70	0.51	0.39	0.35	0, .1	0.28
Effective of wing extension , λ	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38
Factor Blade of the polar, A_0	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038
Coefficient K_A	1	1	1	1	1.09	1.16	1.27	1.33	1.4	1.5
Coefficient K_{CXO}	1	1	1	1	1.03	1.2	1.5	1.7	1.9	1.95
The coefficient of aerodynamic drag in horizontal flight, $C_{xa.h.f}$	23.8454	1.5100	0.3151	0.1141	0.0417	0.0367	0.0389	0.0418	0.0450	0.0453
Flight aerodynamic quality, K	1.05	4.15	8.83	13.72	16.69	13.92	10.06	8.30	6.87	6.12
Thrust required $P_{Required,\,N}$	5381413	1363125	640062	411870	338526	405953	561663	681076	822303	923415
The Rate of changes in the thrust of the number of flight Mach , ξ	0.972	0.952	0.940	0.935	0.950	0.969	0.995	1.011	1.029	1.048
Take-off thrust engines, P_0, N	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000
Thrust available, P_{avail} , N	664688	650963	642627	639639	649541	662349	680341	691268	703475	716958
Vertical speed, V_y, m/S	-25.71	-7.76	0.04	4.97	10.17	9.78	5.17	0.47	-5.83	-10.69

Table 9: The calculation of thrust required and thrust available, H = 8 km.

Flight altitude, км	**10**									
The number of flight Mach	0.1	0.2	0.3	0.4	0.6	0.7	0.8	0.85	0.9	0.95
Relative density of the air, Δ	0.337	0.337	0.337	0.337	0.337	0.337	0.337	0.337	0.337	0.337
The air density, $\rho H, \frac{kg}{m^3}$	0.4136	0.4136	0, 4136	0.4136	0.4136	0.4136	0.4136	0.4136	0.4136	0.4136
Coefficient K_a	0.88	0.88	0.88	0.88	0.88	0.88	0.88	0.88	0.88	0.88
The speed of sound a, m/S	299.446	299.446	299.446	299.446	299.446	299.446	299.446	299.446	299.446	299.446
Flight speed , V, m/s	29.94	59.89	89.83	119.78	179.67	209.61	239.56	254.53	269.50	284.47
Dynamic pressure, N/m²	185	742	1669	2967	6676	9086	11868	13398	15020	16735
The average gross weight of the aircraft, G_{Av}, N	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560
The coefficient of aerodynamic lift in horizontal flight, $C_{yh.f.}$	33.67	8.42	3.74	2.10	0.94	0.69	0.53	0.47	0.42	0.37
Effective of wing extension , λ	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38
Factor Blade of the polar, A_0	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038
Coefficient K_A	1	1	1	1	1, 09	1, 16	1, 27	1, 33	1, 4	1, 5
Coefficient K_{CX0}	1	1	1	1	1.03	1.2	1.5	1.7	1.9	1.95

The coefficient of aerodynamic drag in horizontal flight, $C_{xa\ h.f.}$	43.1065	2.7138	0.5529	0.1893	0.0579	0.0460	0.0449	0.0467	0.0491	0.0489
Flight aerodynamic quality, K	0.78	3.10	6.77	11.12	16.16	14.93	11.73	9.98	8.47	7.63
Thrust required $P_{Required}$, N	7234021	1821721	835106	508293	349600	378392	481800	565957	667341	740384
The Rate of changes in the thrust of the number of flight Mach , ξ	0.972	0.952	0.940	0.935	0.950	0.969	0.995	1.011	1.029	1.048
Take-off thrust engines, P_0, N	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000
Thrust available, P_{avail} , N	541395	530216	523426	520992	529058	539490	554145	563045	572988	583970
Vertical speed, V_y, m/S	-35.47	-13.69	-4.96	0.27	5.71	5.98	3.07	-0.13	-4.50	-7.87

Table 10: The calculation of thrust required and thrust available, H = 10 km.

Flight altitude, км	11									
The number of flight Mach	0.1	0.2	0.3	0.4	0.6	0.7	0.8	0.85	0.9	0.95
Relative density of the air, Δ	0.298	0.298	0.298	0.298	0.298	0.298	0.298	0.298	0.298	0.298
The air density, $\rho H, \frac{kg}{m^3}$	0.3648	0.3648	0.3648	0.3648	0.3648	0.3648	0.3648	0.3648	0.3648	0.3648
Coefficient K_a	0.867	0.867	0.867	0.867	0.867	0.867	0.867	0.867	0.867	0.867
The speed of sound a, m/S	295.023	295.023	295.023	295.023	295.023	295.023	295.023	295.023	295.023	295.023
Flight speed , V, m/s	29.50	59.00	88.51	118.01	177.01	206.52	236.02	250.77	265.52	280.27
Dynamic pressure, N/m²	159	635	1429	2540	5715	7779	10161	11470	12859	14328
The average gross weight of the aircraft, $G_{Av,}$ N	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560
The coefficient of aerodynamic lift in horizontal flight, $C_{yh.f.}$	39.33	9.83	4.37	2.46	1.09	0.80	0.61	0.54	0.49	0.44
Effective of wing extension , λ	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38
Factor Blade of the polar, A_0	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038
Coefficient K_A	1	1	1	1	1.09	1.16	1.27	1.33	1.4	1.5
Coefficient K_{CX0}	1	1	1	1	1.03	1.2	1.5	1.7	1.9	1.95
The coefficient of aerodynamic drag in horizontal flight, $C_{xah.f.}$	58.8019	3.6948	0.7467	0.2506	0.0711	0.0536	0.0497	0.0507	0.0524	0.0518
Flight aerodynamic quality, K	0.67	2.66	5.85	9.81	15.37	14.97	12.36	10.74	9.26	8.42
Thrust required $P_{Required,}$ N	8448432	2123422	965534	576114	367586	377343	457240	526053	610317	671356
The Rate of changes in the thrust of the number of flight Mach , ξ	0.972	0.952	0.940	0.935	0.950	0.969	0.995	1.011	1.029	1.048
Take-off thrust engines, P_0, N	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000
Thrust available, P_{avail} , N	487655	477586	471470	469277	476542	485939	499139	507156	516112	526004
Vertical speed, $V_{y,}$ m/S	-41.56	-17.19	-7.74	-2.23	3.41	3.97	1.75	-0.84	-4.43	-7.21

Table 11: The calculation of thrust required and thrust available, H = 11 km.

Flight altitude, км	**12**									
The number of flight Mach	0.1	0.2	0.3	0.4	0.6	0.7	0.8	0.85	0.9	0.95
Relative density of the air, Δ	0.2536	0.2536	0.2536	0.2536	0.2536	0.2536	0.2536	0.2536	0.2536	0.2536
The air density, $\rho H, \frac{kg}{m^3}$	0.3156	0.3156	0.3156	0.3156	0.3156	0.3156	0.3156	0.3156	0.3156	0.3156
Coefficient K_a	0.867	0.867	0.867	0.867	0.867	0.867	0.867	0.867	0.867	0.867
The speed of sound a, m/S	295.023	295.023	295.023	295.023	295.023	295.023	295.023	295.023	295.023	295.023
Flight speed , V, m/s	29.50	59.00	88.51	118.01	177.01	206.52	236.02	250.77	265.52	280.27
Dynamic pressure, N/m²	137	549	1236	2198	4944	6730	8790	9923	11125	12396
The average gross weight of the aircraft, $G_{Av,}$ N	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560
The coefficient of aerodynamic lift in horizontal flight, $C_{yh.f.}$	45.46	11.36	5.05	2.84	1.26	0.93	0.71	0.63	0.56	0.50
Effective of wing extension , λ	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38
Factor Blade of the polar, A_0	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038
Coefficient K_A	1	1	1	1	1.09	1.16	1.27	1.33	1.4	1.5
Coefficient K_{CX0}	1	1	1	1	1.03	1.2	1.5	1.7	1.9	1.95
The coefficient of aerodynamic drag in horizontal flight, $C_{xa\ h.f.}$	78.5576	4.9295	0.9906	0.3278	0.0877	0.0631	0.0559	0.0557	0.0567	0.0554
Flight aerodynamic quality, K	0.58	2.31	5.10	8, 67	14.40	14.69	12.72	11.29	9.91	9.09
Thrust required $P_{Required,}$ N	9764610	2450941	1108159	651889	392361	384584	444301	500309	570447	621625
The Rate of changes in the thrust of the number of flight Mach , ξ	0.972	0.952	0.940	0.935	0.950	0.969	0.995	1.011	1.029	1.048
Take-off thrust engines, P_0 , N	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000
Thrust available, P_{avail} , N	425163	416384	411052	409141	415475	423667	435176	442165	449973	458598
Vertical speed, Vy, m/S	-48.76	-21.25	-10.92	-5.07	0.72	1.43	-0.38	-2.58	-5.66	-8.09

Table 12: The calculation of thrust required and thrust available, H = 12 km.

Flight altitude, км	**12.4**									
The number of flight Mach	0.1	0.2	0.3	0.4	0.6	0.7	0.8	0.85	0.9	0.95
Relative density of the air, Δ	0.2399	0.2399	0.2399	0.2399	0.2399	0.2399	0.2399	0.2399	0.2399	0.2399
The air density, $\rho H, \frac{kg}{m^3}$	0.2986	0.2986	0.2986	0.2986	0.2986	0.2986	0.2986	0.2986	0.2986	0.2986
Coefficient K_a	0.867	0.867	0.867	0.867	0.867	0.867	0.867	0.867	0.867	0.867
The speed of sound a, m/S	295.023	295.023	295.023	295.023	295.023	295.023	295.023	295.023	295.023	295.023
Flight speed , V, m/s	29.50	59.00	88.51	118.01	177.01	206.52	236.02	250.77	265.52	280.27
Dynamic pressure, N/m²	130	520	1170	2079	4678	6367	8317	9389	10526	11728
The average gross weight of the aircraft, G_{Av}, N	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560	5650560
The coefficient of aerodynamic lift in horizontal flight, $C_{yh.f.}$	48.05	12.01	5.34	3.00	1.33	0.98	0.75	0.67	0.59	0.53
Effective of wing extension , λ	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38	8.38
Factor Blade of the polar, A_0	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038	0.038
Coefficient K_A	1	1	1	1	1.09	1.16	1.27	1.33	1.4	1.5
Coefficient K_{CX0}	1	1	1	1	1.03	1.2	1.5	1.7	1.9	1.95

Citation: Ahmed Soliman M.Sherif (2017) Calculation of the Flight Characteristics of the Aircraft, AN-225. J Aeronaut Aerospace Eng 6: 183. doi: 10.4172/2168-9792.1000183

The coefficient of aerodynamic drag in horizontal flight, $C_{xah.f.}$	87.7547	5.5044	1.1041	0.3637	0.0954	0.0676	0.0587	0.0581	0.0586	0.0571
Flight aerodynamic quality, K	0.55	2.18	4.84	8.26	13.99	14.51	12.79	11.46	10.12	9.32
Thrust required $P_{Required,}$ N	1E+07	2589322	1168646	684376	403975	389474	441832	493270	558414	606117
The Rate of changes in the thrust of the number of flight Mach , ξ	0.972	0.952	0.940	0.935	0.950	0.969	0.995	1.011	1.029	1.048
Take-off thrust engines, P_0, N	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000	1404000
Thrust available, P_{avail} , N	405560	397185	392099	390276	396318	404133	415110	421777	429226	437452
Vertica' speed, $V_{y,}$ m/S	-51.77	-22.89	-12.16	-6.14	-0.24	0.54	-1.12	-3.17	-6.07	-8.37

Table 13: The calculation of thrust required and thrust available, H = 12.4 km.

Height Flight, H, km	0	2	4	6	8	10	11	12	12.4
Minimum speed, $Vmin$, m/s (left)	46	55	60	70	94	118	140	170	200
Maximum(Full) speed, $Vmax$, m/s (right)	277	275	275	267	264	255	248	230	200

Table 14: Calculation of a possible aircraft flight boundaries

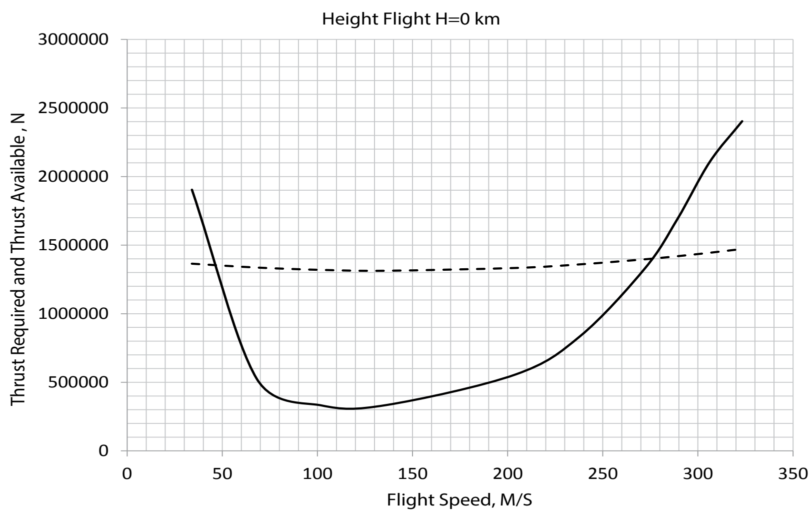

Figure 1: Thrust required and thrust available at a height of H = 0 km.

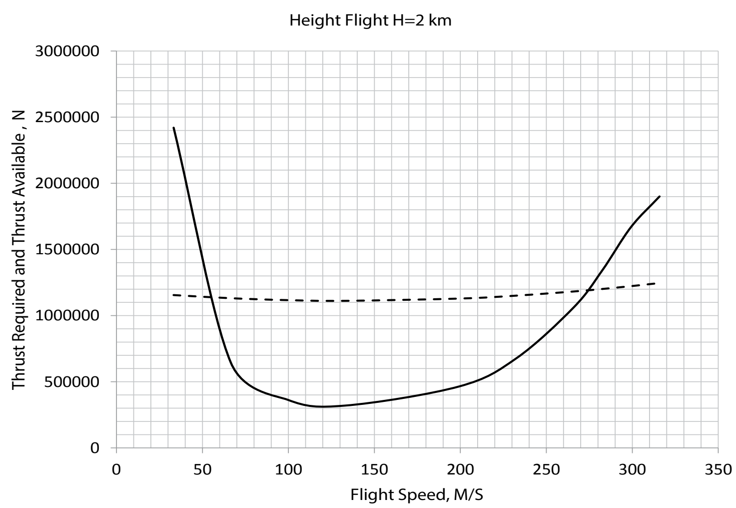

Figure 2: Thrust required and thrust available at a height of H = 2 km.

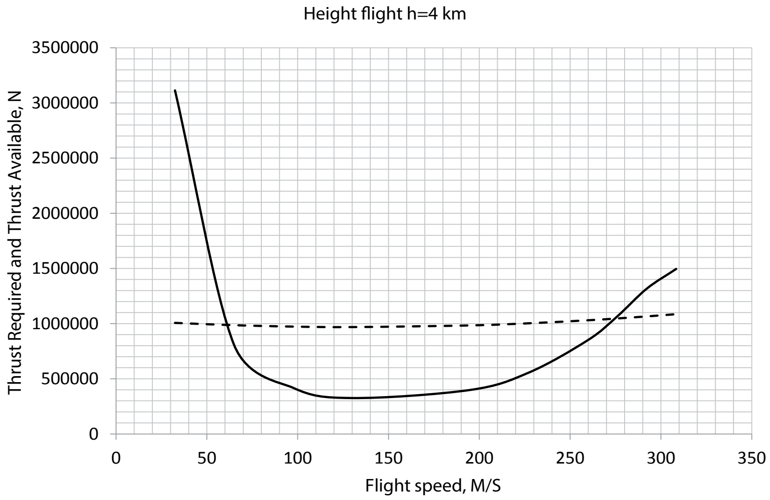

Figure 3: Thrust required and thrust available at a height of H = 4 km.

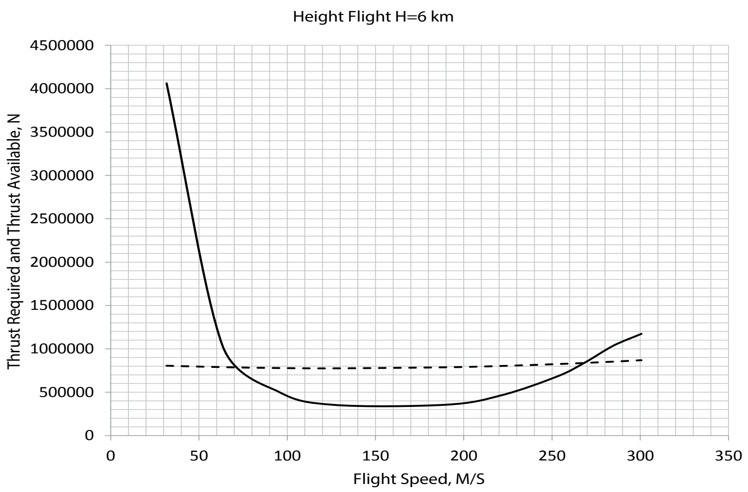

Figure 4: Thrust required and thrust available at a height of H = 6 km.

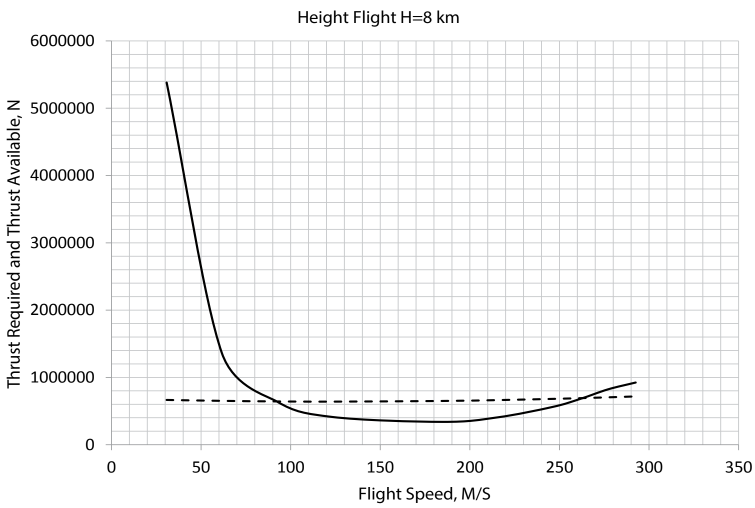

Figure 5: Thrust required and thrust available at a height of H = 8 km.

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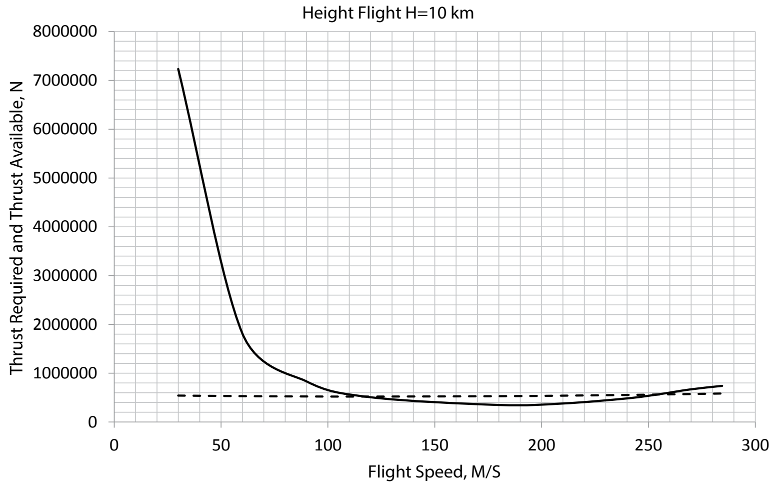

Figure 6: Thrust required and thrust available at a height of H = 10 km.

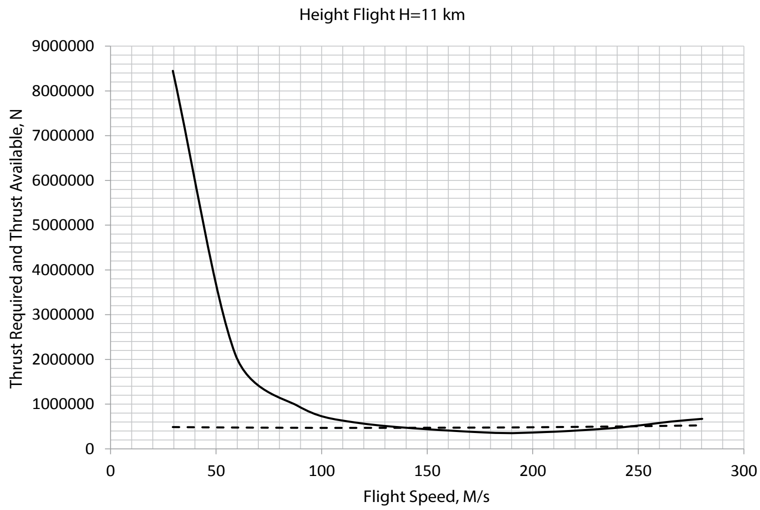

Figure 7: Thrust required and thrust available at a height of H = 11 km.

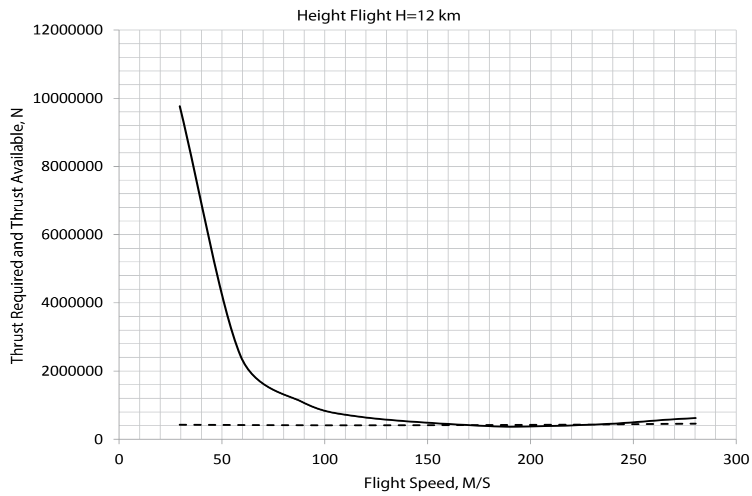

Figure 8: Thrust required and thrust available at a height of H = 12 km.

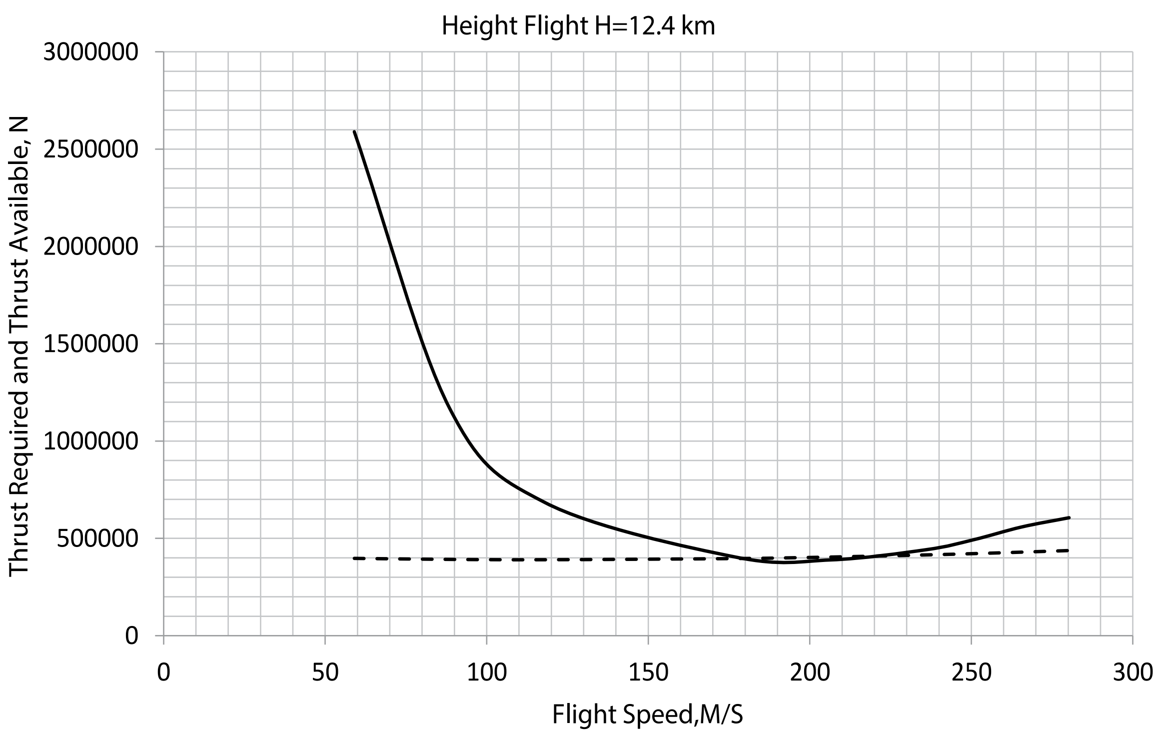

Figure 9: Thrust required and thrust available at a height of H = 12.4 km.

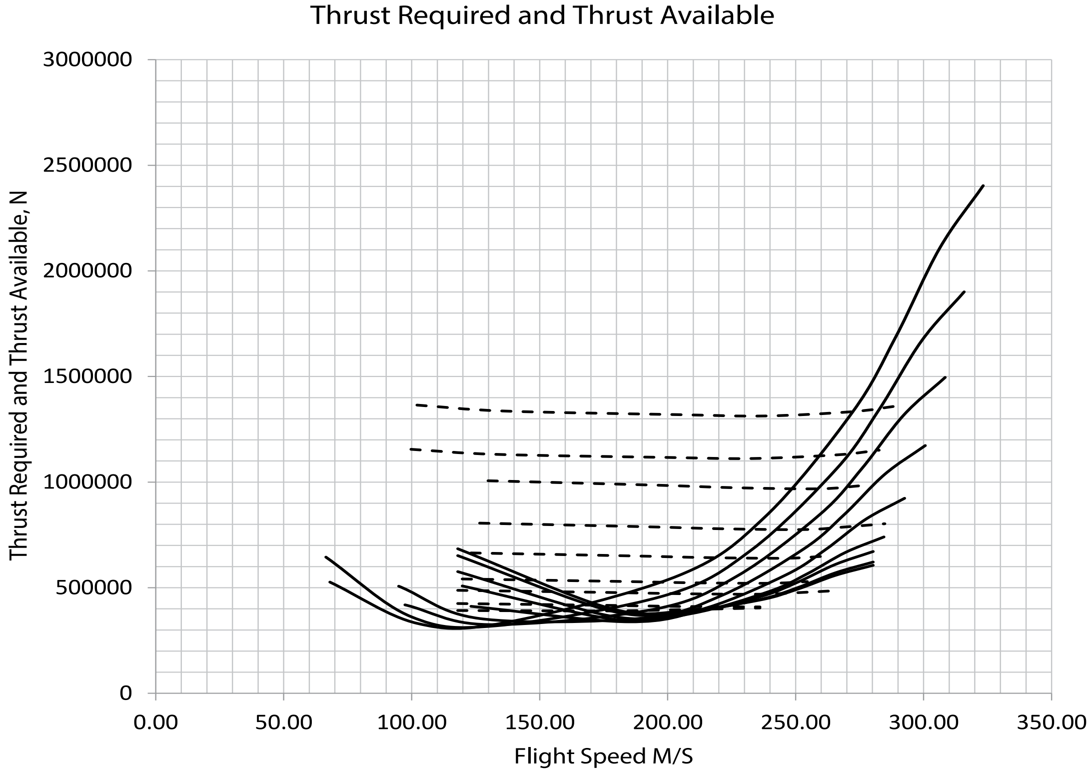

Figure 10: Thrust required and thrust available at a height of H = 0…. 12.4 km.

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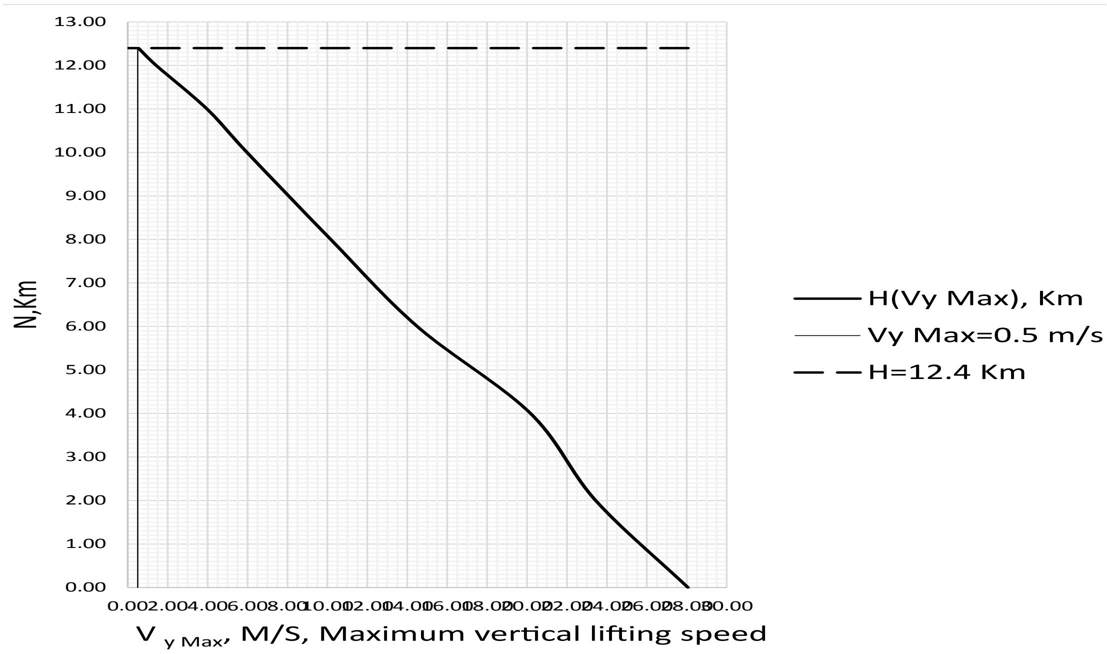

Figure 11: Finding a practical ceiling of the aircraft.

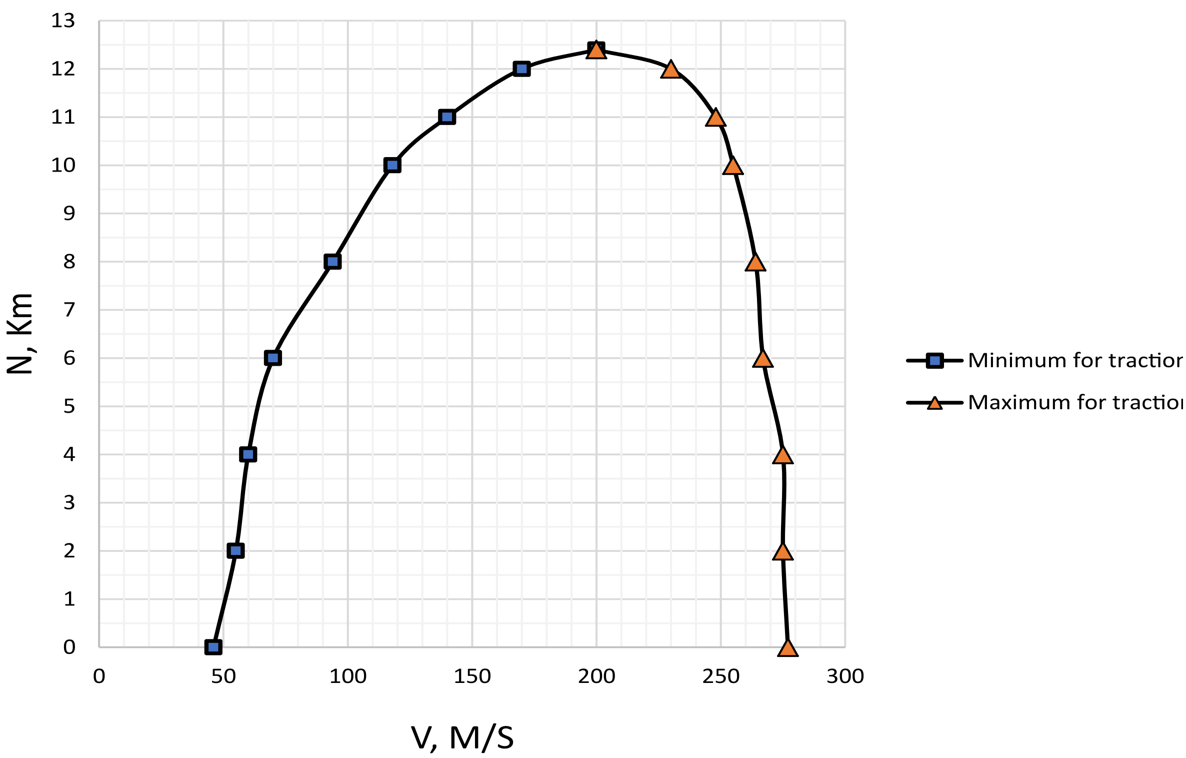

Figure 12: Real possible area of the aircraft (maximum for traction).

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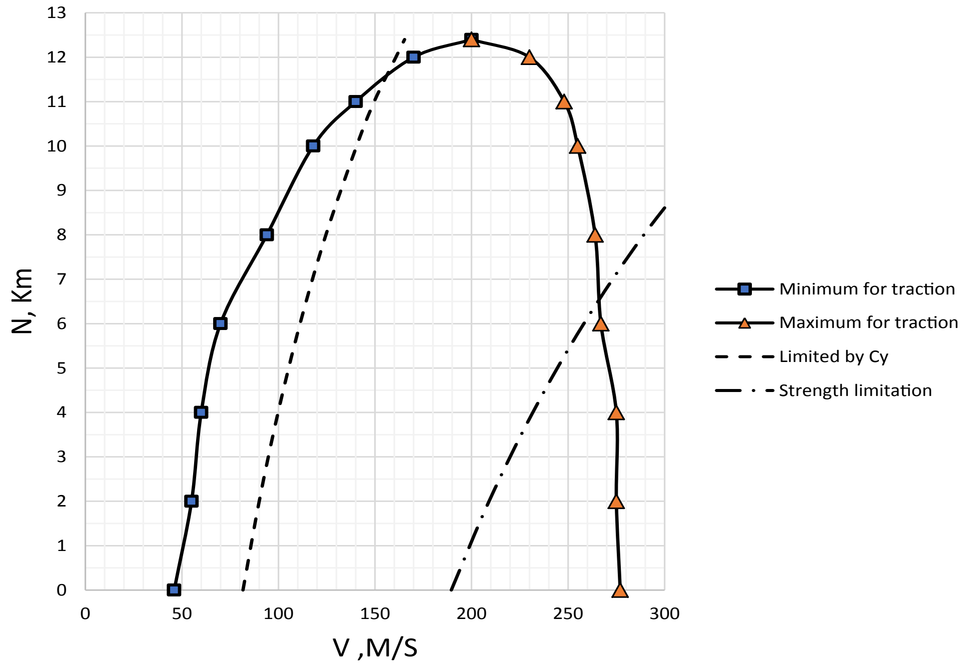

Figure 13: Real possible area of the aircraft (minimum for traction).

The minimum flight speed by the formula:

$$V_{min} = \sqrt{\frac{2\times G_{Av.}}{C_y \times \rho H \times S}} = \sqrt{\frac{2\times 5650560}{1.53\times \rho H\times 905}} = \sqrt{\frac{8161.7}{\rho H}}$$

On the right side, should be limits on the N/m² maximum velocity head from the condition of strength $q_{max,max} = 22000$.

Defining this condition maximum speed flight according to the formula (Tables 15 and 16):

$$Vmax = \sqrt{\frac{2\times q_{maxmax}}{\rho H}} = \sqrt{\frac{2\times 22000}{\rho H}} = \sqrt{\frac{44000}{\rho H}}$$

Height, H, km	0	2	4	6	8	10	11	12	12.4
Density of air, *ρH, kg/m3*	1.225	1.0067	0.8194	0.6602	0.5259	0.4136	0.3648	0.3156	0.2986
Minimum speed, V_{min},m/s	81.6	90	99.8	111.2	124.6	140.5	149.6	160.8	165.3

Table 15: Calculation of the minimum flight speed.

Height, H, km	0	2	4	6	8	10	11	12	12.4
Density of air, *ρH, kg/m3*	1.225	1.0067	0.8194	0.6602	0.5259	0.4136	0.3648	0.3156	0.2986
Minimum speed, V_{min},m/s	189.5	209.1	231.7	258.2	289.3	326.2	347.3	373.4	383.9

Table 16: The calculation of the maximum flight speed of conditions limitations on the maximun velocity head.

Conclusion

1. By, considering level flight at various altitudes with the same weight of flight and angle of attack, when performing level flight at any altitude is necessary to ensure equality of lifting forces and gravity of the aircraft, as Y_a =G. To fulfill this condition for constant weight and angle of attack at high altitudes where the air density is less true speed horizontal flight should be more, but the airspeed remains constant.

2. In carrying out flight on a modern passenger airplane Flight weight is significantly reduced due to fuel production. Such a change of flight mass causes a significant change in the aircraft flight characteristics. To perform horizontal flight of flight with less weight requires less lifting force, hence for the same attack and altitude angle requires less speed and less traction.

$$V_{h.f.} = \sqrt{\frac{2\times G}{C_{Y_a}\times \rho H\times S}}$$

3. As can be seen from the graphs of Thrust Required and Thrust Available (power), the speed range is reduced by raising at height, So all the speed characteristics increases by raising at height, with the exception of the V_{max}, because its value is determined by the characteristics of the engine.

4. With increasing altitude, the air density decreases, which leads to an increase in required speed and reduction of vertical speed(climb). Characteristics of climb is getting worse due to the fall of the engine thrust. At a certain height excess thrust is reduced to zero, so a further climb is not possible.

5. With increasing altitude, the excess thrust is reduced and at some certain height becomes zero. This means that the vertical velocity of the steady rise is also reduced to zero. At this altitude and above the aircraft is not able to make a steady recovery.

6. Flight altitude at which the vertical velocity of the steady rise equal zero is called a theoretical (or static), the ceiling of the aircraft.

7. There's not an excess thrust On a theoretical ceiling therefore the only possible is horizontal flight, and only the most advantageous angle of attack (and only in the most advantageous rate) at which the lowest Required thrust power. Speed range at this moment equal zero.

8. With the steady rise of the plane, almost cannot reach the theoretical ceiling, because as you get closer to it excess thrust becomes so small, that in order to set the height of the rest needs to spend too much time and fuel. Due to the lack of excess flying thrust on a theoretical ceiling is almost impossible, because any violation of the flight mode cannot be eliminated without excessive traction. For example, when randomly formed even small roll plane loses a considerable height (falls). Therefore, in addition to theoretical concepts (static) Ceiling introduced the concept of practical ceiling.

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