A98-31591

ICAS-98-4,4,5

MANUFACTURING AND TESTING OF GRAPHITE- EPOXY WING BOX AND FUSELAGE STRUCTURES FOR A SOLARPOWERED UAV-HAVE

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Abstract

The design of an automatical High Altitude Long Endurance (17-20 km) flying platform, capable of remaining aloft for very long periods of time, is presented. After that the vehicle should climb to 17 km by direct sun radiation, the level flight would be maintained by direct solar energy during the day and by a fuel cells energy storage system during the night. A computer program has been developed for carrying out a parametric study for the platform design, by taking into account solar radiation, latitude and altitude, mass and efficiency of solar cells and fuel cells, aerodynamics aspects, etc. A wide use og high modulus CFRP has been made in designing the structure in order to minimize the airframe weight. The final configuration of the platform is a monoplane, with 46 m of span, six brushless motors, twin-boom tail type and two rudders and TOGW of 6000 N. The platform is expected to cost about $ 3 million or 345 $ per hour of flight in one year.

The design of a flying scale-sized technological demonstrator has been completed aiming to the realization of a proof-of-concept aircraft. The main wing box structure, two C-spar and two skin sandwich panels, has been manufacturing in three pieces, each one 8m long by using a graphite/epoxy pre-preg tape, and will be tested to shear - bending - torsion up to the ultimate load to verify the theoretical behaviour. A particular design has been developed for manufacturing fuselage and boom-tail; a very light pin-jointed CFRP truss-structure 3.6 m long has been manufacturing to carry the applied load. CFRP square tubes are bonded one to each other, by CFRP joints, to form up to seven connections. It will be tested under shear-bending load to show the correlation with the theoretical FEM analysis.

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Nomenclature

b	Wing span
C_L	Coefficient of lift
C_D,C_{Do}	Total and profile drag coefficient
E	Dayly solar energy obtainable by 1 m^2
F	Fuel cells energy density
k	S_t/S_w
n	limit load factor
P_{req}	Horizontal flight required power
$S_{sc}\%$	Solar cell covered area
S_t	Horizontal tail area
S_w	Wing area
V	Platform Airspeed
W_{af},W_{sc}	Airframe and solar cells weight
W_{tot}	Total Platform weight
Z	Altitude
η_{blade}	Propellers efficiency
η_{fc},η_{sc}	Fuel cells and solar cells efficiency
η_{gears}	Reduction gear efficiency
η_{motor}	Electric motor+inverter efficiency
η_{prop}	Propulsion system efficiency
λ	Aspect ratio
ρ	Air density

Introduction

Some successful projects, both sport records and scientifically oriented (Solar Challenger, Icaré 2 and Pathfinder)[1,3], have demonstrated that it is possible to produce a flying vehicle capable of remaining sustained in flight for long periods just thanks to the power of sun. There is a great request, nowadays, for an automatical high altitude (17-25 km), flying platform capable of remaining aloft for very long periods of time [UAV-HAVE (Unmanned Air Vehicles - High Altitude Very-long Endurance]. It would play the role of an artificial satellite, with the advantage of being much cheaper and more flexible; in fact, it could be selflaunched and being easily recovered for maintenance, whenever necessary.

21st ICAS Congress
13-18 September 1998
Melbourne, Australia

The mission of such platform would cover a very large spread of applications: atmospherical pollution control and meteorological monitoring, real-time monitoring of seismic-risk areas, coast surveillance, agriculture monitoring, telecommunications, cellular - telephons networks, photogrammetry and so on. From a flying altitude of 17 km, an area of about 500 km of diameter would be covered for communication trasmission by properly choosing the onboard antenna irradiation diagram. Just four platforms should cover Italy and Corsica from North to South (Fig. 1).

One of the main advantages of this artificial satellites compared to real satellites, besides to the costs (about $ 200-300 million for satellite), would be the much higher resolution due to the "low" flying altitude (17-20 km). At the end of 1994, NASA has started the ERAST (Environmental Research Aircraft and Sensor Technology) program; one of the four drones is the solar plaftorm Pathfinder which exceeded 20 km of altitude in a 13 hours flight. Also in ESA activity, a study has been planned for the future on HALE-UAV platform. Under the financial support of the Italian Space Agengy (ASI), a research is being carried out at the Turin Polytechnical University aiming to the design of UAV-HAVE Platform and manufacturing of a smaller solarpowered piloted prototype.

Since today's solar cells convert such a little percentage of the sun energy to useful work, the platform must be a very large one. The vehicle should climb to 17-20 km by taking advantage, mainly, of direct sun radiation and maintaining, afteron, a level flight; electrical energy not required for propulsion and payload operation is pumped back into the fuel cells energy storage system and, during the night, the platform should have to maintain the altitude by the stored (solar) energy.

This hard sun-powered level flight can be achieved only by means of a suitable design and an accurate integration of the best standard achievable for each technological item involved such as the structural weight optimization, the aerodynamic, the solar cells, the fuels cells, the propulsion efficiencies and the electronic devices. Solar cells and fuels cells are, up to-day, the worst element of the project. Thin high efficiency (17%) crystalline silicon cells are today available at about 820 $\$/m^2$, and considered very useful for our purpose; however, when an higher effieciency is required, the cost increases incredibly to 35000 $\$/m^2$ for 21% of efficiency and to 110000 $\$/m^2$ for 23% of efficiency.

Regenerative Fuel Cells (RFCs) are of great interest for storing energy in various space applications. In general the systems are based on the electrolysis of water, the storage of hydrogen and oxygen, and the following recombination to water in a fuel cell during the power production. In the reversible design, each individual electrochemical cell is able to work as well in the electrolysis as in the fuel cell mode, simplifying in this way the configuration of the overall system to a great extent with the main advantages of reducing weight of the storage system. Furthermore, the fuel cells and the electrolysers, based on the solid polymer technology, meet the following requirements: a) Long lifetime (40,000 hours); b) Reliability of the components (more than two lifetime years); c) Voltage stability. The fuel cell efficiency is, up to-day, the second worst element of the project. The RFCs energy storage system gives an energy density of 400-600 Wh/kg, much higher than 100 Wh/kg given by better batteries. Indeed, today efficiency of a RFCs system is about 55% and its production cost is of a few hundrend thousands dollars.

Several types of high altitude solar powered platforms (HASPP) have been designed in the past [4,5]; however, the only flying platform is, up to-day, the UAV Pathfinder, used as demonstrator of the greater platform Helios (up to 67m of span).

As an alternative to the solar platform, unmanned aerial vehicles (Condor, Tier II+, Theseus, Raptor, etc.) are considered very promising for their higher payload capability (up to 10 kN) with a long endurance of 24 - 48 hours. Indeed, their cost is up to $ 60 - 80 million. Furthermore, since their limited range of 24 - 48 hours, is not possible to have a continous earth monitoring unless two or more UAV are flying simultaneously. Manned airplanes have similar constraints, which would be extremely expensive. Balloons are often used for observation purpose; however, they are limited by the weather conditions in which they have to operate and cost is very prohibitive ($ 60 million as minimum).

A parametric study has been carried out for the platform design, taking into account the wind speed up to 27 km, the minute-to-minute solar radiation change along one year, altitude, latitude, solar cells efficiency and weight, fuel cells efficiency and energy density, aerodynamic drag. The results obtained give very important indication of which parameters have to be improved to reduce weight and dimensions of the platform or in alternative to increase the payload.

Platform Design Considerations

The solar radiation available in Italy from the North to the South (from 45° North latitude to 37° N. lat.) has been first determined. At the operating altitude of 17-20 km, the platform will surely fly above all cloud cover not being, in this way, any daytime influence of the sunlight, although an higher power is required because of the thin air density. A good compromise would be a flying altitude of 17 km, where the jet-stream has a relatively low speed value.

Although the average solar radiation received at the edge of the atmosphere is 1353 W/m^2, by taking into account declination of the sun, latitude, solar hour angle, incidence angle, azimuth angle, atmospheric absorption, etc. a little/medium part of the solar irradiation is lost. The solar radiation available at 17 km of altitude is reported in Figure 2. In the worst day (22 Dec), the maximum specific solar power, for a short period of time, is 675 W/m^2 and 475 W/m^2, respectively, at 36° and 45° N. lat. (Fig.2a). In Figure 2c, the maximum specific solar power available at noon along one year is reported; a power greater than 1050 W/m^2 would be available for more than half year. However, in the winter solstice, more than 15 hours of darkness are present requiring a very high power from the fuel cells. The solar energy available, at several latitude, to 1 m^2 of solar cells, integrated all over the day, is reported in Fig. 2b. As it will be seen, in the worst day, only 4000 Wh/m^2 would be available at 36° N. lat. and 2600 Wh/m^2 at 45° N. lat. There will not be problems, however, in the best day of the year (22 June).

A wind study has been performed at several locations to examine the operating speed of the platform; the upper air data climatic tables recorded by the Italian Air Force have been elaborated to obtain the wind profiles as function of the altitude (from 1000 hPa to 10 hPa) for several latitudes [6]. The data were taken from 1963 to 1986. Along the many data there reported, the wind speed has been elaborated and plotted for several cities. Wind has a minimum speed between the altitudes 18 and 20 km, depending from the time of year; the maximum mean value recorded is less than 50 km/h (Fig.3). However, taking into account standard deviations and confidence interval, higher speed values have to be considered in the design[7]; a maximum speed of 86 km/h has to be considered at an 86% occurrence; considering a 95 percentile the speed raises to 115 km/h and at a 99 percentile would be 140 km/h. In conclusion, although in very few cases, the propulsion power should be able to contrast this environmental effect.

The design procedure followed in the analysis is based on the energy balance equilibrium between the available solar power and the required power; the former being dependent from the solar cell area installed on wing and stabilizer, the latter depending from the velocity and total drag of the platform. In particular, the available power, produced by solar cells, has not only to give the power for the daylight equilibrium, but also to rigenerate the fuel cells for the night flight power. The endurance parameter has to be fulfilled to minimize the power required for an horizontal flight, that means to minimize the parameter $C_D/C_L^{3/2}$; it has been preferred to reduce as the coefficient of drag C_D instead of only increase the coefficient of lift C_L, also because of the greater structural load. An iterative procedure has been followed untill the wing area and aspect ratio chosen can satisfy the energy equilibrium. By appropriate equations combination it can be written as:

$$(2/\rho)^{1/2}\,(W_{tot}/S_w)^{3/2}\,(C_D/C_L^{3/2})/(\eta_{prop}) = [(\eta_{sc})(\eta_{fc})(1+k)\,(S_{sc}\%)\,E]\,/\,[(\eta_{fc})2h+(24\text{-}2h)]$$

where the propulsion efficiency is given by:

$$\eta_{prop} = \eta_{blade}\ \eta_{gears}\ \eta_{motor}$$

and (24-2h) is the number of hours in which the fuel cells has to give power.

The total power required is obtain by adding the power required by the payload (500 W) and by the avionics (1.2 W/N where the weight of avionics is estimated to be 3% of the total weight).

The coefficient of drag of the platform is given by the sum of profile drag (wing, horizontal tail, rudders and winglets), induced drag (wing and tail) and parasite drag (pod, booms, interaction with wing and tail). Laminar profiles have been chosen for wing and tail surfaces. Wortmann profiles have extensively been used in the past since the many available experimental data at several Reynolds numbers; in these last years, however, new excellent profiles were developed by L. Boermans at Delft University of Technology (DU-89-134-14, etc.)[8]; a laminar flow is maintened up to 70-80% of the chord reducing in such way the drag coefficient up to 20-30% with respect to Wortmann profiles; since the platform will operate at an altitude of low density and at a relative low speeds, Reynolds number between 300,000 and 700,000 should be considered.

To complete the power equilibrium, an airframe weight estimation of the platform is necessary; based on previous papers on man-powered airplanes, high-altitude airplanes or ultralight cantilever twin-boom tails the following expression has been derived[5], as function of aspect ratio, ultimate load factor (maximum value of 4) and wing planform area, for the airframe weight:

$$W_{af} = 8.75\ (N^{0.311}\lambda^{0.4665}\ S_w^{0.7775})$$

The total platform weight, per wing area, is determined by adding to the airframe, the weight of the propulsion (32.37 N/kW), solar cells (6.87 N/m^2 of covered area), avionics (3% of the total weight) and payload (1000 N); the weight of the propulsion system includes brushless motors, reduction gears, inverter, conditioning and propellers.

Fuel cells weight, including fuel cells, electrolysers, tankage and reactants, is given by:

$$[(24\text{-}2h)g(P_{req}/S_w)\ S_w]\ /\ F$$

By working all the above equations is possible to obtain the total planform area as function of aspect ratio and weight, that would make the equilibrium between required and available power. The platform has been designed for a payload weight of 1000 N.

Parametric results

Hundreds of analytical results have been obtained by a parameteric study of the above equations. The total planform area (wing and horizontal tail) is plotted as function of the aspect ratio λ (Fig. 4). The platform has been designed by considering the worst day of the year (22 dec.). Any point below the curve should not be capable of collecting enough energy during the day for flying continuously also all the night and for one year. Of course, a significant reduction in platform size can be possible by restricting the time of year in which is required to fly. The following baseline values were assumed for the study:

η_{prop}=0.7618; η_{sc}=0.21; η_{fc}=0.7; k=0.36; F=600 Wh/kg; W_{sc}=6.87 N/m^2; C_{do}=0.008; Z=17 km; S_{sc}%=85% λ=20$\div$38.

Three latitude were considered in the analysis: 36°, 40° and 45° North latitude, corresponding to Malta, Naples and Turin, respectively.

The aspect ratio of the platform has been limited to 38 for not having a too much high wing span; in many cases this has not permitted to obtain the minimum of the planform area, although the results change only of little amount.

Starting from the value of the above quantities, only one parameter has been changed to see the performance results of the platform, obtaining the following results:

1)Effect of Altitude(Fig. 4a). As the flight altitude increases, the total surface increases tremendously; for an aspect ratio λ=36, changing the altitude from 15 to 20 km, the total area would increase of about 98% and 145% at 36° and 45° N. lat., respectively. The difference is greater for the lowest aspect ratio values.

2)Effect of Wing Drag and Winglet. The aerodynamic performances have been taken into consideration by considering the effect of both the profile drag C_{do} (Fig. 4b) as of the inductive drag, by including a winglet. A C_{do}=0.008 baseline value is today possible with the usual laminar profile. A possible reduction to 0.006 should be possible by a dedicated profile for low Reynolds number; its effect, however, should produce a 10% total area reduction. By introducing a winglet, 2 or 4 m high, the advantages have been also very poor givin about the same reduction.

3)Effect of Solar Cells Efficiency and Weight. Solar cells performances have a great influence on the size of HAVE solar powered platform. Efficiency influence is reported in Figure 4c; a 21% baseline value is today available although at very high cost (35000 \$/m^2); it is possible within five years that they should be obtained at much lower cost; a present day cheaper efficiency of 18% and a long term very expensive efficiency of 24% have been also considered. For λ=36, reducing the efficiency to 0.18, the total area would increase of about 30% and 49% at 36° and 45° N. lat., respectively; increasing the efficiency to 0.24, the total area would decrease of about 21% and 25% at 36° and 45° N. lat., respectively. The influence of the cells weight is also notable; with respect to the present day 6.87 N/m^2 baseline value, a specific weight of 5.4 N/m^2 would be obtained within near term and a value of 3.92 N/m^2 at long term. For λ=36, reducing the cells weight to 5.4 N/m^2, the total area would decrease of only 9% and 15% at 36° and 45° N. lat., respectively; reducing the cells weight to 3.92 N/m^2, the total area would decrease of about 18% and 28% at 36° and 45° N. lat., respectively.

4)Effect of Fuel Cells Efficiency and Energy Density. Fuel cells performances have a significant influence on the size of a very long endurance solar powered platform. Efficiency influence is reported in Figure 4d; with respect to the 70% efficiency baseline optimistical value, but possible to be obtain within five years, a present day efficiency of 50% and a near term efficiency of 60% have been also considered. For λ=36, reducing the efficiency to 0.6, the total area would increase of about 20% and 34% at 36° and 45° N. lat., respectively; reducing the efficiency to 0.5, the total area would increase of about 53% and 97% at 36° and 45° N. lat., respectively. The influence of the energy density is also notable; with respect to the 600 Wh/kg baseline optimistical value, but possible to be obtain within near term, a present day density of 400 Wh/kg has been also considered. For λ=36, reducing the energy density to 400 Wh/kg, the total area would increase of about 27% and 35% at 36° and 45° N. lat., respectively;

HELIPLAT configuration

As result of the above parametric study, a first final configuration of HELIPLAT (**HELIOS PLATFORM**) (Fig. 5) has been worked out. The platform is a monoplane with six brushless motors, twin-boom tail type with an oversized horizontal stabilizer and two rudders. The main characteristics obtained at a design altitude of 17 km carrying a payload of 1000 N, in four different flight configurations, are reported in Table 1; the nine months flight period starts from the first of february.

TABLE 1 - Main characteristics of 4 different platform configurations.

latitude[°N]	38	38	44	44
time[months]	12	9	12	9
W_{tot}[N]	6000	4515	8840	5750
S_w[m^2]	129	84	227	122
b [m]	46	30	82	44
1.2 P_{req}[W]	3020	2495	4000	2920

η_{fc}=0.65; η_{sc}=0.20; η_{prop}=0.7618 λ=40; S_{sc}=90% ; S_t=0.36 S_w

The power reported is 20% higher than that required in an horizontal flight; a power reserve of 5kW has also been put into account for a wind speed event up to 86 km/h. The choosing of six motors have been considered a good compromise between redudancy and weight. A significant reduction in the platform size has been possible by restricting to nine months the time of year in which is required to fly and would be obtainable, in my opinion, with the to-day technologies; indeed, a very large dimensions would be necessary for flying in North Italy.

The airframe is about 55% of the total weight, the solar cells 15% and the fuel cells 17%. In comparison to the $ 60 million cost of the HALE-UAV Tier II, this type of platform is expected to cost $ 2-3 million or $ 230-345 per hour of flight in one year.

The structural design of the wing has been completed. A very beneficial effect, on shear and bending moment loads applied to the wing at the maximum limit load, has been obtained by booms, motor and fuel cells inertia loads.

Design of the Prototype HELIPLANE

The design of the flying scale-sized technological demonstrator HELIPLANE (HELIos AirPLANE) has been completed aiming to the realization of a proof-of-concept aircraft. It would have a more usual configuration and initially it would be piloted (Fig. 6).

To minimize the power required for an horizontal flight, we preferred to reduce coefficient of drag instead of only increase the coefficient of lift, also because of the greater structural load.

The profile chosen for our wing has been the DU 97-155 having a C_{Do} of only 0.004-0.005 for a C_L varying between 0.2 and 0.6 (Re = 1.8E6). At the highest solar speed of 100 km/h, obtainable with a solar power of 1000 W/m^2, and low coefficient of lift, the wing profile drag is 45% of the total drag and the wing-induced drag is only 10%; the parasite drag (fuselage, interaction with wing, tails) is 45% of the total drag, requiring than an accurate aerodynamic study of the stream lines of the wing profile for designing the fuselage shape. The wing has a rectangular plan to reduce the manufacturing cost of the very expansive mold for the autoclave cure of the main wing-box. Vertical and horizontal tails do not give an high contribution to the overall drag by choosing the Wortman profile FX 71-L-150/30 or 20.

The final HELIPLANE configuration would be a twin electric brushless motor having the following characteristics:
S_w=28.8m^2; b=24m; λ=20; Power=13kW; n=+4.5,-2.75. V_D=120km/h; V_{s1}=46km/h; V_{sol}=100km/h (1000W/m^2); Best Glide=36
Weight=3200N (Pilot=900N; Wing=650N; Solar cells=300N; Batteries=400N; Motors=250N).

A computer program has been developed for designing the anisotropic wing box, lay-up and thicknesses. The wing box, composed of two sandwich panels and two webs, would be manufactured in 3 boxes, 7.5 m long each, by using a M40J graphite/epoxy prepreg and Nomex core, and autoclave cured. The main lamina properties would be: E_1=200 GPa; σ_1=2000 MPa; ϵ_1=1.2%; ρ_c=1.65 kg/dm^3.

After curing, lower and upper panels would be riveted along the neutral axis of the webs, co-cured with each panel (Fig. 7); then the 3 boxes are joined by 3 bolts for each side in the half meter of overlapping between two adjacent wing-boxes. Leading and trailing edges would be manufactured by foam and bonded to the wing box. The upper and lower steel molds for the autoclave curing has been manufactured and after that, the wing boxes have been manufacturing and will be tested to shear- bending - torsion up to the limit load, to verify the theoretical behaviour. The shear and bending moment diagram along the span is reported in Figure 8; an important load alleviation has been obtained by not only the wing weight (including the solar cells) but also by the motor and batteries placed into the wing. A maximum tip deflection (Fig. 8) of 1.5m and angle of twist of 2 degrees would be fulfilled; as a consequence, the maximum strain that would be reached in the spars is 0.0030. Although an higher strain should be supported by the structure, a fatigue life greater than 10 million cycles should be obtained without any damage propagation. The overall wing box is planned to be tested by shear - bending - torsion load and all the mechanical equipment for applying the load is being prepared. Since the mechanical fittings between the three wing boxes are critical points, several specimens are being tested to verify the bearing stress and failure modes.

A 6 meter long wing-box was been already manufactured for getting experience with a real long structures[9]; since a 2m maximum autoclave length was available in our laboratories, the wing box was manufactured in three pieces and joined togheter by 4 bolts for each side. A shear - bending- torsion test has been carried out up to the failure load. A good correlation has been obtained between the theoretycal and experimental results. The compression sandwich panel showed a good postbuckling behaviour up to a load 50% greater than buckling load; an eccentric load is applied to panels because of the sandwich tapering due to the bonding to the spars and ribs. Failure has occurred, at a bending moment of 19 kNm, by debonding of the spar from the skin panel because of an erroneous use of liquid release agent in manufacturing one spar.

A particular design has been developed for manufacturing the fuselage; a very light pin-jointed CFRP truss-structure would be manufactured to carry the applied load. CFRP square tubes should be bonded one to each other, by CFRP joints, to form up to seven connections; thin layers of glass fibers would be modelled around a foam to obtain the aerodynamic fuselage shape (Fig. 9). A sample 3.6 m long truss-structure has been manufacturing (Fig. 10) for getting experience with the technology. A T400-epoxy UD pre-preg was used to manufacture the square tube having the lay-up: $(0/\pm 20_5/0)$. The thin walled-tube was fabricated by utilizing an expandable rubber mandrel and an aluminium female mold; after the pre-preg was cut and 2 layers stacked with the desired fiber orientation, the wrapping procedure began by aligning the rubber with one edge of the composite layers; five successive layers were used to obtain the desired thickness; after on, the assembly was placed in the female mold, vacuum bagged and autoclave cured at a temperature of 180 °C and pressure of 0.5 MPa; each tube is 1900 mm long and cross-section dimension of 26 mm. After the several elements have been cut, they have been assembling together by proper CFRP joints, also them autoclave cured, and bonded by a two components epoxy resin. Overall length is 3600 mm, height and width 300 mm. The truss-structure should be subjected to a three point bending test, by simply-supporting the ends and applying the load, step-by-step, in the middle section by an hydraulic jack;

The MSC/NASTRAN finite element program was used to model the truss-structure in order to predict the applied load in each beam, the maximum deflections (Fig. 11), natural frequencies and mode shapes; the comparison between the analytical and the experimental results will be reported.

Conclusion

The results of this parametric study show that should be possible to realize an high altitude very long endurance platform for earth observation and telecommunication applications at least for low latitude sites of Europe. The most significant reduction in platform size, or expanding mission range or altitude or latitude sites, could occur by increasing the fuel cells specific energy to 600 Wh/kg and efficiency to 65%; furthermore it should be very important to obtain these performances at a production cost sensibly lower than $ 300.000. The solar cells efficiency and specific weight should also play an important role to satisfy the flight performances. The to-day solar cell efficiency of 21% could be very useful if it should be obtained at a low production cost. The flying altitude play a significant role in the dimensions of the platform; then, it is very important to determine the mission requirements of the platform. Of course, the platform size and weight can be reduced, or the flying altitude can be increased, by reducing the required period of the year in which the platform should have to fly, and would be manufactured with the to-day technologies.

The first results obtained by the technological demonstrator show the feasibility of the airplane while a good correspondence between the experimental and theoretical results are expected from the many tests in progress.

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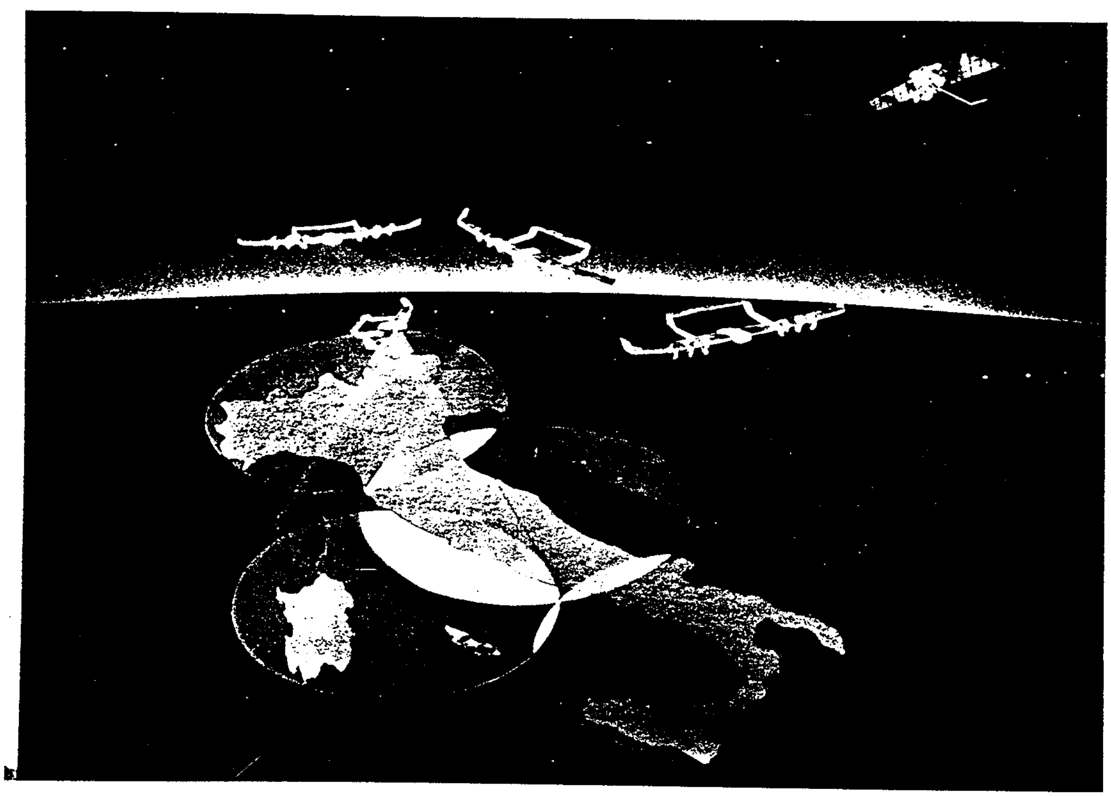


Fig.1 - Solarpowered UAV-HAVE Platform Scenario

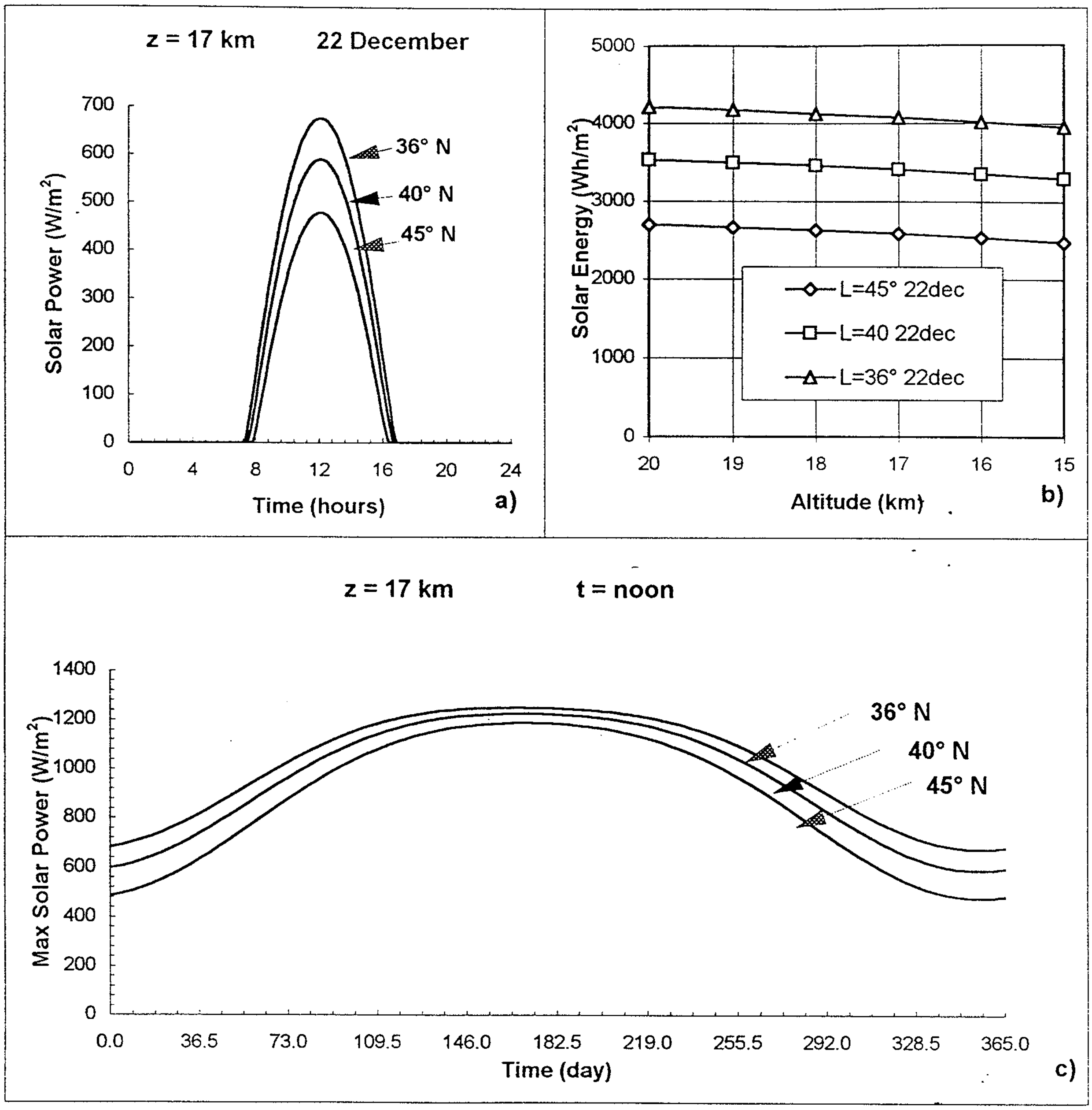


Fig.2 - Daily (a) and yearly (c) solar power and solar energy (b) at several latitude

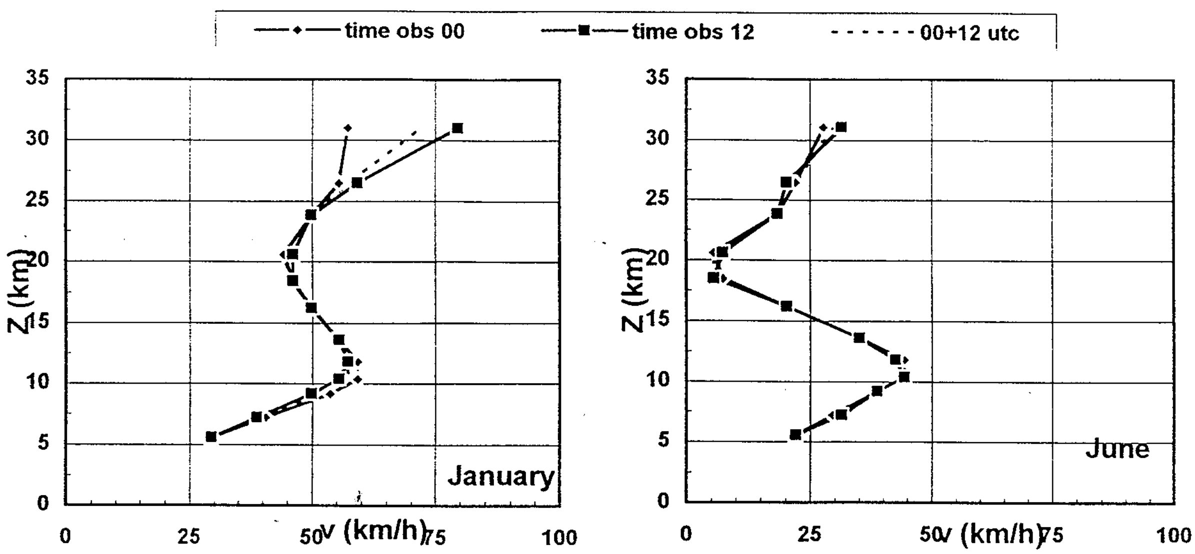


Fig.3 - Wind speeds as function of altitude at Milano Linate

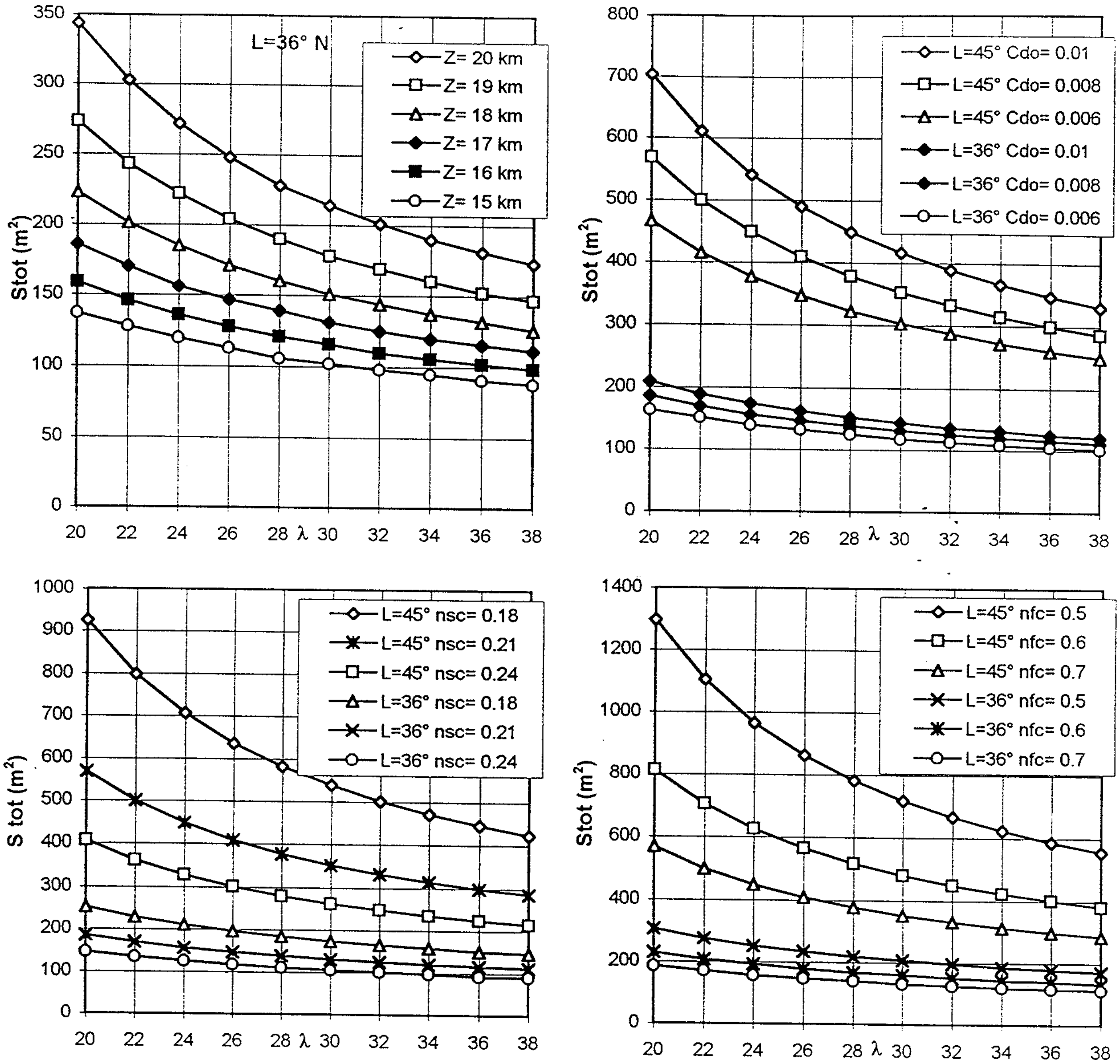


Fig.4 - Required total planform area as function of: (a) altitude (b) drag coefficient (c) solar cells efficiency (d) fuel cells efficiency

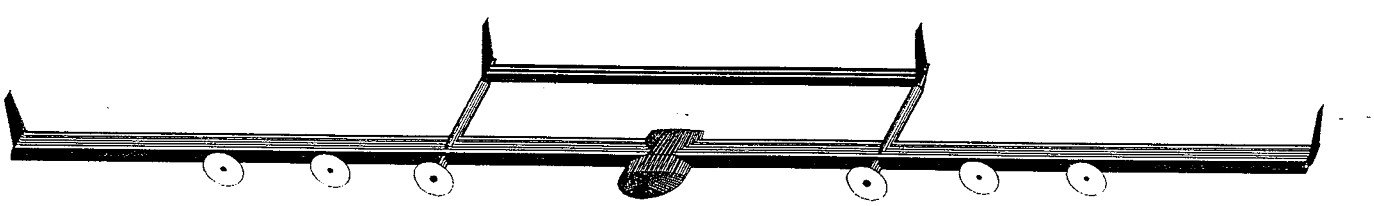


Fig.5 - UAV-HAVE HELIPLAT configuration

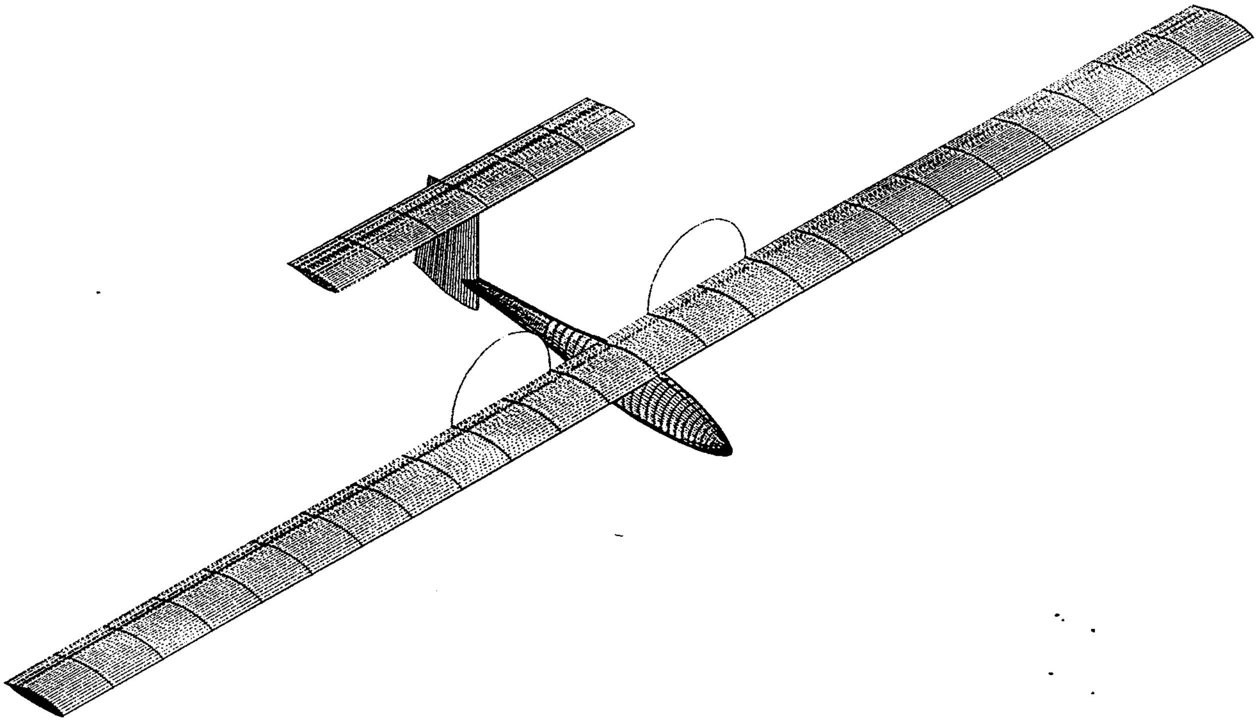


Fig.6 - Solarpowered airplane HELIPLANE

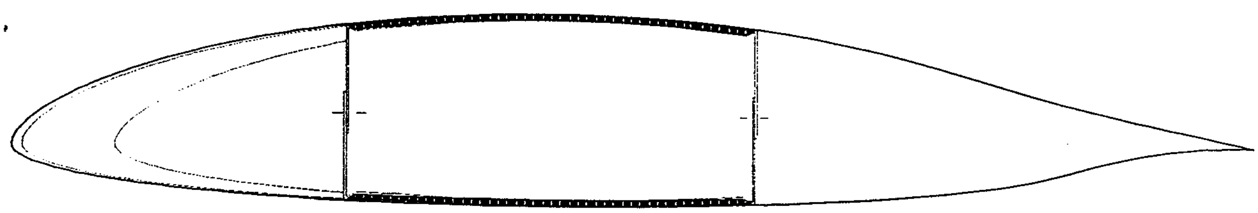


Fig.7 - Wing-section structural configuration

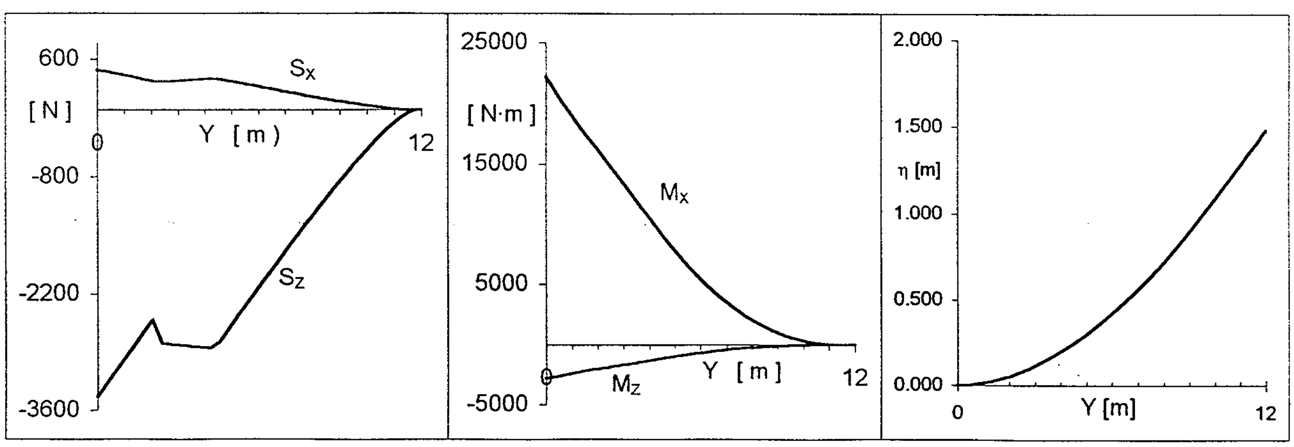
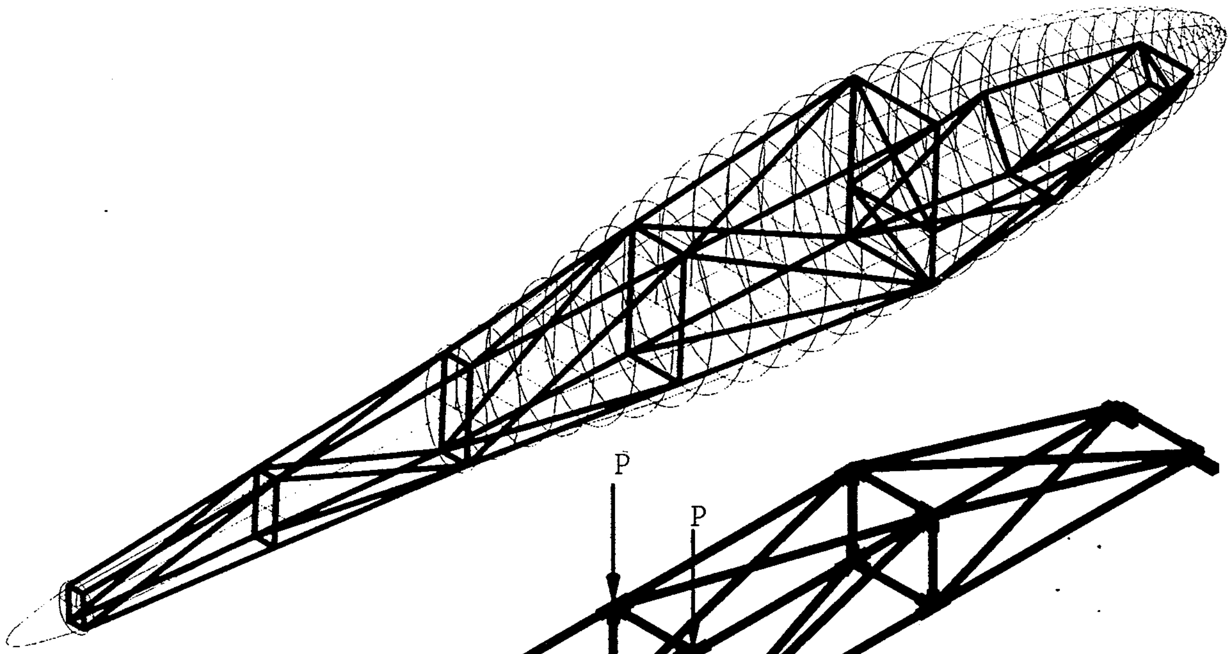


Fig.8 - Shear, bending moment and deflection along half wing span



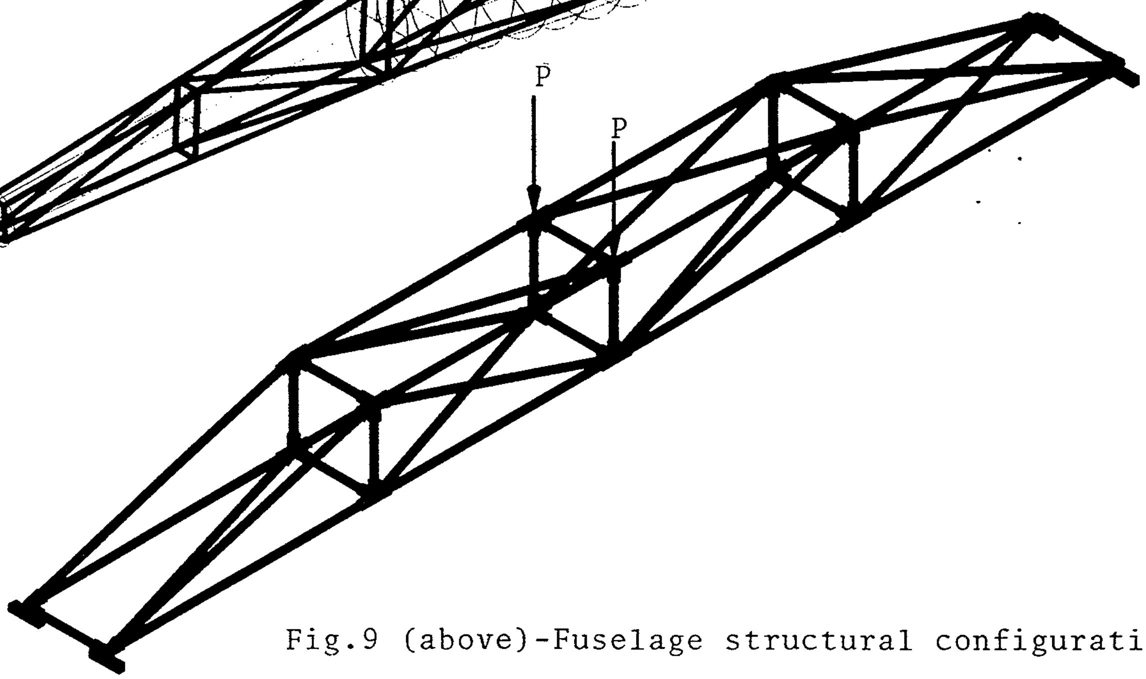


Fig.9 (above)-Fuselage structural configuration.
Fig.10(below)-Fuselage truss-structure sample.

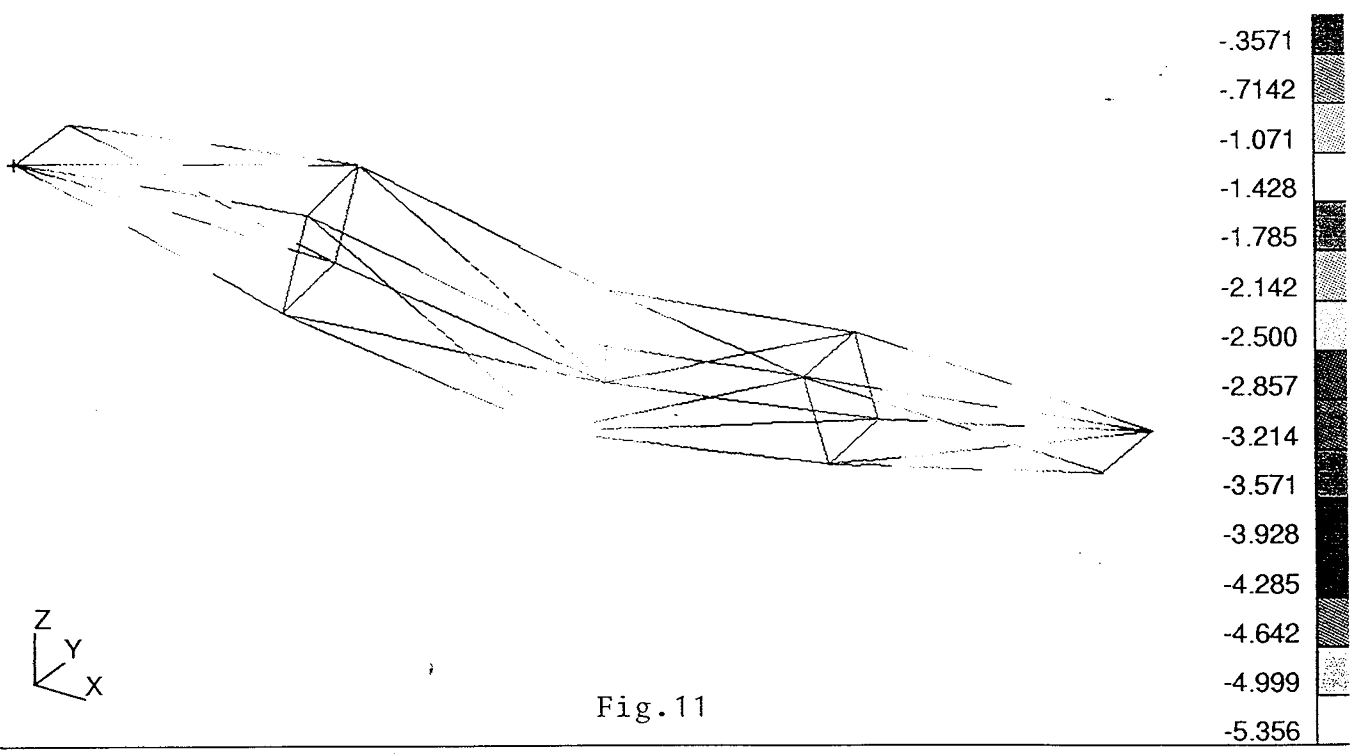


Fig.11