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Conceptual Sizing, Rapid Prototyping and Drag Estimation of a Twin-Engine Trainer Aircraft

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Abstract

A final year undergraduate project based on the conceptual design of a new twin-engine trainer aircraft named MKTS-7 has been carried out at the Hindusthan Institute of Technology. The new aircraft design performed is based on the idea to build a 4-seat high wing aircraft with two light engines (Rotax 912, usually used for ultralight aircrafts) and to enter the market with a twin-engine aircraft with the same weight of a single engine aircraft. The present paper shows all main criteria for the initial sizing of the aircraft and the choice of the configuration have been based. The specification data are initially collected from the existing twin engine Italian aircraft, Tecnam P2006T. An aircraft aerodynamic investigation has been performed, especially to estimate the drag of the vehicle, both theoretically and experimentally (through wind-tunnel tests of a 1:100 scaled 3D printed model). Further, this paper focuses on the importance and application of the 3D printing technology and the ease of testing at the early stages of design. All tests and research activities are focused on the evaluation of aircraft geometric layout, aerodynamics and in particular on the estimation of the total drag value.

Keywords: *Twin-engine, Rapid prototyping, 3D printing, Drag estimation, Conceptual design, Weight estimation, Computer-aided drafting, Stereolithography, Subsonic wind tunnel, SLA Viper Si2, CATIA V5R20, Geometric sizing, Configuration layout, Lofting*

1. Introduction to Aircraft Conceptual Design

We engineers can never quite agree as to just where the design process begins. From designer point of view, the design starts with a new airplane concept. A sizing specialist knows that nothing can begin until an initial estimate of the weight is made. From customer, civilian or military point of view, design begins with the requirements. To our surprise, all those are correct. Actually, design is an iterative effort to size the entire configuration. It is in conceptual design that the basic questions of configuration arrangement, size and weight, and performance are answered.

The first question is, "Can an affordable aircraft be built that meets the requirements?" If not, the customer may wish to relax the requirements. Conceptual design is a very fluid process. New ideas and problems emerge as a design is investigated in ever-increasing detail. Each time the latest design is analyzed and sized, it must be redrawn to reflect the new gross weight, fuel weight, wing size, engine size, and other changes. Early wind-tunnel tests often reveal problems requiring some changes to the configuration. This initial sizing and wind-tunnel testing is the primary focus of this paper.

The first design step, involves sketching a variety of possible aircraft configurations that meet the required design specifications. By drawing a set of configurations, designers seek to reach the design configuration that satisfactorily meets all requirements as well as go hand in hand with factors such as aerodynamics, propulsion, flight performance, structural and control systems. This is called design optimization. Fundamental aspects such as fuselage shape, wing configuration and location, engine size and type are all determined at this stage. Constraints to design like those mentioned above are all taken into account at this stage as well. The final product is a conceptual layout of the aircraft configuration on paper or computer screen, to be reviewed by engineers and other designers.

1.1 Survey of existing aircrafts

Surveys of existing twin engine high wing aircrafts were made. The aircrafts of the same class are listed below:

1. Partenavia P68
2. Piper PA-34 Seneca
3. Beechcraft Baron G58
4. Diamond Twin Star
5. Piper Seminole
6. Tecnam P2006T

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To our surprise, of these 6 aircrafts, the Tecnam P2006T aircraft has most of its specifications matching our design need. So, the general and performance specifications of this aircraft can be taken as the initial data for our design, as they are practically proven. The three view conceptual sketch of the proposed aircraft MKTS-7 is shown in Fig.1.

DESIGN PROPOSAL : MKTS-7 TWIN ENGINE TRAINER AIRCRAFT

We propose to design a twin engine trainer aircraft with a high wing configuration. Design will be focused on making the configurations such that the aircraft could fly at a low subsonic cruise Mach number and longer range and be able to operate at higher altitudes compared to the existing aircraft. This design is a response to the existing trainer design Tecnam P2006T that has a limited range and endurance.

The proposed characteristics of the MKTS-7, along with those of the Tecnam P2006T, are given in the following table. The principle design drivers will be higher range, greater endurance, higher cruise speed and comparable cost.

Table 1: Proposed characteristics of MKTS-7 versus those of Tecnam P2006T

Proposed Characteristics	***MKTS-7***	***Tecnam P2006T***
Maximum Cruise Speed (km/hr)	300	287
Cruise Altitude (m)	4200	4200
Range (km)	1,500	1,148
Endurance (hrs)	4.5	4.25
Max. Take-Off Weight (kg)	1,200	1,180
Empty Weight (kg)	750	760
Initial Rate of Climb (m/s)	8	6.4
Take-Off Distance (m)	500	518
Landing Distance (m)	500	518
Wing Loading (kg/sq. m)	80	78

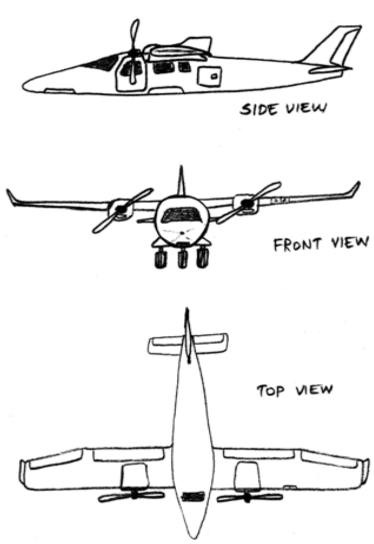

Fig. 1 Three views of the proposed design MKTS-7.

2. Sizing from the Conceptual Sketch

Following the design proposal and initial conceptual sketch, the first step in the design of our MKTS-7 is to obtain an estimate of the Gross Takeoff Weight. There are many levels of design procedure. The simplest level just adopts past history. For our design, if we need an immediate estimate of the takeoff weight of MKTS-7 to replace the existing Tecnam P2006T, we can take 1,180 kg. This is what the Tecnam weighs, and is probably a good number to start with. To get the right answer takes several years, many people, and lots of money. Actual design requirements must be evaluated against a number of candidate designs, each of which must be designed, analyzed, sized, optimized, and redesigned any number of times. The flight profile for the design mission necessary for calculation of the total takeoff weight is shown in Fig.2.

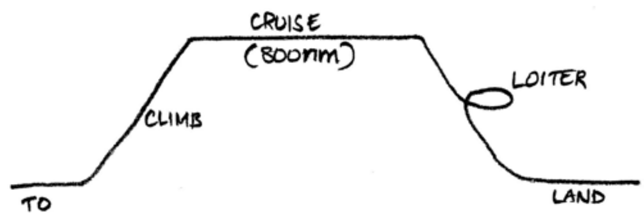

Fig. 2 Mission Profile of MKTS-7.

2.1 Empty Weight Estimation

The empty weight fraction $\left(\frac{W_e}{W_0}\right)$ can be estimated statistically from historical trends as shown in Table 2. The available empty weight consists of the initial estimated take-off weight minus all the removable weights including fuel weight and expendable and nonexpendable payload weights. This is then compared to the required empty weight, which is the structure weight we can expect for a particular type of aircraft, based on historical data. This is a fixed percentage of the take-off weight defined as the structural factor given as, $s = \frac{W_{empty}}{W_{TO}}$.

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Table 2: Structural factor for selected aircraft as a function of take-off weight

$s = AW_{TO}{}^{C}$	A	C
Sailplane (powered)	0.91	-0.05
General Aviation (single engine)	2.36	-0.18
General Aviation (twin engine)	1.51	-0.10
Twin Turboprop	0.96	-0.05
Jet Trainer	1.59	-0.10
Jet Fighter	2.34	-0.13
Jet Transport	1.02	-0.06

Structural factor for MKTS-7, s = 1.51 x $(1{,}200)^{-0.10}$ = 0.7431 i.e., $\left(\frac{W_s}{W_0}\right)$ = 0.7431

2.2 Fuel-fraction Estimation

Only a part of the aircraft's fuel supply is available for performing the mission ("mission fuel"). The other fuel includes reserve fuel as required by civil design specifications, and also includes "trapped fuel", which is the fuel which cannot be pumped out of the tank. The required amount of mission fuel depends upon the mission to be flown, the aerodynamics of the aircraft, and the engine's fuel consumption. The aircraft weight during the mission affects the drag, so the fuel used is a function of the aircraft weight.

The total amount of fuel used during the flight mission is based on considering the individual amounts used within each flight phase. For any of the flight phase, the fuel used (by weight) is represented as the ratio of the fuel weight leaving (final) to that entering (initial) that flight phase, namely,

$$\text{Fuel Weight Fraction} = \left(\frac{W_f}{W_0}\right)_{fuel}$$

The total fuel fraction for the complete flight plan is equal to the products of the individual weight fractions in the respective flight phases.

Table 3: Mission segment weight fractions

Mission Segment	$\left(\frac{W_i}{W_{i-1}}\right)$
Warm-up and takeoff	0.970
Climb	0.985
Cruise	0.9229
Loiter	0.9888
Landing	0.995

The warm-up and takeoff, climb and landing weight fractions can be estimated historically, but the cruise and loiter weight fractions are calculated from the Breguet Range and Endurance equation by using the SFC and L/D ratio estimated for those phases.

The total mission segment weight fraction can be calculated as,

$$\frac{W_5}{W_0} = \left(\frac{W_1}{W_0}\right) \text{x} \left(\frac{W_2}{W_1}\right) \text{x} \left(\frac{W_3}{W_2}\right) \text{x} \left(\frac{W_4}{W_3}\right) \text{x} \left(\frac{W_5}{W_4}\right) \qquad (1)$$

$$\frac{W_5}{W_0} = 0.970 \text{ x } 0.985 \text{ x } 0.9229 \text{ x } 0.9888 \text{ x } 0.995 = 0.8675$$

Since, our trainer aircraft mission segments does not involve payload drops, all weight lost during the mission must be due to fuel usage. The mission fuel fraction must therefore be equal to $\left(1 - \frac{W_5}{W_0}\right)$. If we assume, typically, a 6% allowance for reserve and trapped fuel, the total fuel fraction can be estimated by using the following equation.

$$\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_5}{W_0}\right) \qquad (2)$$

$$\frac{W_f}{W_0} = 1.06(1 - 0.8675) \qquad (3)$$

$$\frac{W_f}{W_0} = 0.14045$$

This shows that 14% of the total takeoff weight accounts for fuel weight.

The total fuel fraction for the complete flight plan is the product of the individual weight fractions for the respective flight phases. The total fuel weight then corresponds to the estimated take-off weight minus the weight after landing minus any expendable (dropped) weight, plus 5 percent reserve and 1 percent trapped fuel.

2.3 Mission Requirements

Table 4: Mission Requirements for MKTS-7

Parameter	Value
Maximum Cruise Speed (km/hr)	300
Cruise Speed (km/hr)	250
Cruise Altitude (m)	4,200
Engine: TSFC Min.	0.5
Engine: TSFC Max.	0.5
Engine: Power (kW)	73.5161
Aspect Ratio	8.8
Loiter: Time (min)	35
Loiter: Altitude (m)	1,200
Fuel Reserve (%)	5
Trapped Fuel (%)	1
Structural Factor	0.7431
Payload (kg)	(80*4) + (15*4) = 380

3. Wing Geometry, Power Loading and Wing Loading

Before the design layout can be started, values for a number of parameters must be chosen. These include the airfoil(s), the wing and tail geometries, wing loading, horsepower-to-weight ratio, estimated takeoff gross weight and fuel weight, estimated wing, tail, and engine sizes, and the required fuselage size. These are calculated in the next three units. This chapter covers the selection of airfoil, wing and tail geometry.

This section also deals with the selection of power loading and wing loading. The horsepower-to-weight ratio (*hp/W*) and wing loading (*W/S*) are the two most important parameters affecting aircraft performance. Optimization of these parameters forms a major part of the analytical design activities conducted after an initial design layout. However, it is essential that a credible estimate of the wing loading and power loading be made before the initial design layout is begun. Otherwise, the optimized aircraft may be so unlike the as-drawn aircraft that the design must be completely redone.

3.1 Airfoil Shape Selection

The airfoil, in many aspects, is the heart of the airplane. The airfoil affects the cruise speed, takeoff and landing distances, stall speed, handling qualities (especially near the stall), and overall aerodynamic efficiency during all phases of the flight. The airfoil is selected by using the basic parameters such as thickness ratio and maximum lift coefficient which are 0.15 and 1.6 respectively. The selected airfoils are NACA 4415 for the root chord and NACA 4412 for tip chord.

3.2 Wing Geometry Selection

For our design, Aspect Ratio, A = 8.8
Wing span, b = 11.5 m

Wing area, $S = \frac{b^2}{A} = 15.0284\ m^2$

Other design considerations for the wing are: high wing configuration, sweep angle of 0˚, taper ratio of 0.45, wing incidence of 2˚, dihedral between 0˚ and 2˚ and upswept wing-tip.

Root chord,

$$C_{root} = \frac{2S}{b(1+\lambda)} = \frac{2 \times 15.0284}{11.5 \times (1+0.45)} = 1.8025\ \text{m} \quad (4)$$

Tip chord, $C_{tip} = \lambda C_{root} = 0.45 \times 1.8008 = 0.8111$ m

Mean aerodynamic chord,

$$\bar{C} = \left(\frac{2}{3}\right) C_{root} \frac{1+\lambda+\lambda^2}{1+\lambda} = \left(\frac{2}{3}\right) 1.8025 \frac{1+0.45+0.45^2}{1+0.45} = 1.3684\ \text{m} \quad (5)$$

Span-wise location of mean aerodynamic chord,

$$\bar{Y} = \left(\frac{b}{6}\right)\left(\frac{1+2\lambda}{1+\lambda}\right) \quad (6)$$

$$= \left(\frac{11.5}{6}\right)\left(\frac{1+2 \times 0.45}{1+0.45}\right) = 2.5114\ \text{m}$$

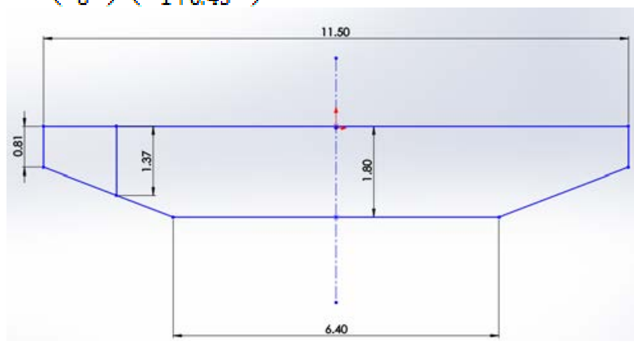

Fig. 3 Main wing geometric layout using the software SolidWorks 2013.

The airfoil selected for the tail is the NACA 0012 considering the symmetry of the airfoil.

3.3 Power Loading Selection

Using our existing data, the power loading is $W/hp = \frac{1200}{(98.6 \times 2)} = 6.085$ kg/hp

So, the horsepower-to-weight ratio is $hp/W = 1/6.085 = 0.1643$ hp/kg

This value is close to the statistical estimate for general aviation twin engine aircrafts.

Using this power loading value, the initial weight estimated is, $W_0 = 6.085 \times (98.6 \times 2) =$ 1,199.962 kg ≈ 1,200 kg

3.4 Wing Loading Selection

The wing loading is the weight of the aircraft divided by the area of the reference (not exposed) wing. As with the thrust-to-weight ratio, the term "wing loading" normally refers to takeoff wing loading, but can also refer to other flight conditions. Wing loading affects stall speed, climb rate, takeoff and landing distances, and turn performance. The wing loading determines the design lift coefficient, and impacts drag through its effect upon wetted area and wing span.

Comparing the calculated values of W/S for stall speed, takeoff, landing and cruise along with the statistical estimate, we see that the lowest value of Wing loading is 80 kg/m^2 which is the wing loading of the existing aircraft. Hence, we can select this value as the wing loading as,

$$W/S = 80 \text{ kg/m}^2$$

$$S = \frac{1200}{80} = 15 \text{ m}^2$$

This will be the surface area of the reference wing.

4. Initial Sizing, Configuration Layout and Loft

Aircraft sizing is the process of determining the takeoff gross weight and fuel weight required for an aircraft concept to perform its design mission. The sizing introduced in section 2 was a quick method based upon minimal information about the design. That sizing method was limited to fairly simple design mission. In this chapter, we will carry out a refined sizing.

An aircraft can be sized using some existing engine or a new design engine. The existing engine is fixed in size and thrust, and is referred to as a "fixed-engine" ("fixed" refers to engine size). The new design engine can be built in any size and thrust required, and is called as "rubber-engine" because it can be stretched during the sizing process to provide any required amount of thrust. However, even if we use an existing engine, we must begin with rubber-engine design study to determine what characteristics to look for in the selection of the existing engine.

The process of aircraft conceptual design includes numerous statistical estimations, analytical predictions, and numerical optimizations. However, the product of aircraft design is a drawing. While the analytical tasks are vitally important, we must remember that these tasks serve only to influence the drawing, for it is the drawing alone that ultimately will be used to fabricate the aircraft. All the analytic efforts done till now were performed for laying out the initial drawing. These drawings will be used as reference for creating a CAD model (section 6). In this section, we will discuss the key concepts followed to develop a credible initial drawing of the conceptual aircraft design. These concepts include the development of smooth, producible, and aerodynamically acceptable external geometry. These drawings are made using the initial sized data the previous section.

The outputs of this configuration layout task will be design drawings as well as the geometric information required for further analysis and modeling. Here, we will place the wing and tail in the fuselage, find the placement of engines on wings and loft the fuselage along with the cross sections. Lofting is the process of defining the external geometry of the aircraft. This lofting is not that advanced as the one done in preliminary design. This will bring the outer layout of the geometry, and they include the common pencil drafting as well as 2D CAD drafting.

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4.1 Fuel Weight Calculations

The empty weight is estimated using the improved statistical equations,

$$\frac{W_e}{W_0} = a + bW_0^{C1}A^{C2}\left({hp}/{W_0}\right)^{C3}\left({W_0}/{S}\right)^{C4} V_{max}^{C5} \quad (7)$$

$$\frac{W_e}{W_0} = -0.90 + 1.32 \text{ x } W_0^{-0.10} \text{ x } 8.8^{0.08} \text{ x } 6^{0.05} \text{ x } 80^{-0.05} \text{ x } 186.411^{0.20}$$

$$\frac{W_e}{W_0} = -0.90 + 3.2677 \text{ x } W_0^{-0.10} \quad (8)$$

The equation given here is in a much better statistical fit, with only about half the standard deviation of the equation used in section 2, because it uses the constants which better reflect the weight impact of major design variables such as aspect ratio, horsepower ratio, wing loading, and maximum speed.

For, W_0 = 1200 kg, $\frac{W_e}{W_0}$ = 0.7081 which means W_e = 849.7688 kg

Comparing it with the data of Tecnam, W_0 = 1180 kg, W_e = 760 kg ; $\frac{W_e}{W_0}$ = 0.644

So, the empty weight fraction of our aircraft is a higher value than for the existing design. We use a fudge factor to adjust the equation.

$$\frac{W_e}{W_0} = \frac{0.644}{0.7081}\left(-0.90 + 3.2677 \text{ x } W_0^{-0.10}\right) \quad (9)$$

$$\frac{W_e}{W_0} = -0.8185 + 2.9718 \text{ x } W_0^{-0.10} \quad (10)$$

Therefore, for W_0 = 1200 kg, $\frac{W_e}{W_0}$ = 0.64401 which means W_e = 772.8225 kg.

We already know that, the weight fractions for warm-up & takeoff, climb, and landing are predetermined.

$\left(\frac{W_1}{W_0}\right)$ = 0.970 ; $\left(\frac{W_2}{W_1}\right)$ = 0.985 ; $\left(\frac{W_5}{W_4}\right)$ = 0.995

We need to calculate the weight fractions for cruise and loiter phases.

Cruise:

Wing loading, W/S = 80 x 0.970 x 0.985 = 76.436 kg/m^2

Dynamic Pressure, q = 1933.7942 N/m^2 = 197.1169 kg/m^2

$$\frac{L}{D} = \frac{1}{\frac{197.1169 \text{ x } 0.02}{76.436} + 76.436 \text{ x } \frac{1}{197.1169 \text{ x } \pi \text{ x } 8.8 \text{ x } 0.8}} \quad (11)$$

$$\frac{L}{D} = 14.4697$$

$$\frac{W_3}{W_2} = exp\frac{-RC_{bhp}}{550\eta_p\left(\frac{L}{D}\right)} \quad (12)$$

$$\frac{W_3}{W_2} = exp\frac{-(800 \text{ x } 6076)0.00005753}{550 \text{ x } 0.8 \text{ x } 14.4697} \quad (13)$$

$$\frac{W_3}{W_2} = 0.957$$

Loiter:

$\frac{L}{D}$ = 0.866 x 14.4697 = 12.5307

$$\frac{W_4}{W_3} = exp\frac{-EVC_{bhp}}{550\eta_p\left(\frac{L}{D}\right)} \quad (14)$$

$$\frac{W_4}{W_3} = exp\frac{-2100 \text{ x } 227.8361 \text{ x } 0.000071917}{550 \text{ x } 0.8 \text{ x } 12.5307} \quad (15)$$

$$\frac{W_4}{W_3} = 0.9937$$

Therefore, total weight fraction,

$$\frac{W_5}{W_0} = 0.970 \text{ x } 0.985 \text{ x } 0.957 \text{ x } 0.9937 \text{ x } 0.995 = 0.90406 \quad (16)$$

Fuel weight fraction,

$$\frac{W_f}{W_0} = 1.06(1 - 0.90406) \quad (17)$$

$$\frac{W_f}{W_0} = 0.10169$$

Fuel weight, $W_f = 0.10169 \text{ x } W_0$ = 122.0356 kg

Usable fuel weight,

$$W_{f\,(usable)} = \frac{122.0356}{1.06} = 115.128 \text{ kg}$$

4.2 Rubber Engine Sizing

Since the empty weight was calculated using a guess of the takeoff weight, it is necessary to iterate towards a solution. This is done by calculating the empty-weight fraction from an initial guess of the takeoff weight and using the equation below to calculate the resulting takeoff weight. If the calculated takeoff weight did not equal the initial guess, a new guess is made somewhere between the two.

$$W_0 = \frac{W_{crew} + W_{payload}}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_e}{W_0}\right)} \quad (18)$$

$$W_0 = \frac{380}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_e}{W_0}\right)} \quad (19)$$

Here,

$$\frac{W_e}{W_0} = -0.8185 + 2.9718 \text{ x } W_0^{-0.10} \quad (20)$$

Table 5: Rubber-engine sizing iterations

W_o Guess	W_f	W_e	W_o Calculated
1200.000	122.028	772.8225	1458.894426
1329.447	135.1915	836.3725	1379.926328
1354.687	137.7581	848.5663	1366.429737
1360.558	138.3552	851.3942	1363.366873
1361.963	138.498	852.07	1362.638495
1362.301	138.5323	852.2326	1362.46344
1362.382	138.5406	852.2718	1362.421263

But this heavier W_0 would give reduced performance with a fixed-size engine.

4.3 Fixed Engine Sizing

The sizing procedure for the fixed-size engine is similar to the rubber-engine sizing, with several exceptions. These result from the fact that either the mission range or the performance must be considered a fallout parameter, and allowed to vary as the aircraft is sized.

We must vary the value of total weight fraction, $\frac{W_5}{W_0}$, until $W_{0_{calc.}} = W_{0_{drawn}}$ = 1200 kg

This occurs when $\frac{W_5}{W_0}$ = 0.963

So, fuel weight fraction becomes,

$$\frac{W_f}{W_0} = 1.06(1 - 0.963) \tag{21}$$

$$\frac{W_f}{W_0} = 0.0.03922$$

Fuel weight, $W_f = 0.03922 \text{ x } W_0$ = 0.03922 x 1200 = 47.064 kg

Usable fuel weight,

$$W_{f_{(usable)}} = \frac{47.064}{1.06} = 44.4 \text{ kg}$$

In a similar fashion, we can iterate to calculate the takeoff weight (Table 6).

Table 6: Fixed-engine sizing iterations

W_o Guess	W_f	W_e	W_o Calculated
1200.000	47.064	772.8225	1199.641828
1199.820914	50.87241	772.7334	1211.891651
1205.856283	51.12831	775.7357	1209.062701
1207.459492	51.19628	776.5326	1208.316058
1207.887775	51.21444	776.7454	1208.116939
1208.002357	51.2193	776.8024	1208.063692
1208.033024	51.2206	776.8176	1208.049442

Now, we have to solve $\frac{W_3}{W_2}$ to determine range,

$$\frac{W_3}{W_2} = exp\frac{-RC_{bhp}}{550\eta_p\left(\frac{L}{D}\right)} \tag{22}$$

And,

$$\frac{W_3}{W_2} = \frac{\frac{W_5}{W_0}}{\left(\frac{W_1}{W_0}\right) \text{x} \left(\frac{W_2}{W_1}\right) \text{x} \left(\frac{W_4}{W_3}\right) \text{x} \left(\frac{W_5}{W_4}\right)} = \frac{0.963}{0.970 \text{ x } 0.985 \text{ x } 0.9937 \text{ x } 0.995} = 1.0193 \tag{23}$$

$$\frac{W_3}{W_2} = exp\frac{-(\text{R}) \text{ x } 0.00005753}{550 \text{ x } 0.8 \text{ x } 14.4697} \tag{24}$$

$$\frac{W_3}{W_2} = 1.0193 = exp\frac{-(\text{R}) \text{ x } 0.00005753}{550 \text{ x } 0.8 \text{ x } 14.4697} \tag{25}$$

Range, R = 2115521.843 = $\frac{2115521.843}{6076}$ = 348.6932 nm

This range of 349 nm is much less than the goal of 800 nm. We will layout the design anyway, and use refined sizing methods and optimization techniques to maximize range and performance.

4.4 Geometric Sizing Data

Fuselage Length = 7.1774m
Tail arm length = 3.9475m
Propeller Diameter = 1.78m

Main Wing Data:
Span = 11.489m
Wing area = 15m^2
Wing aspect ratio = 8.8
Taper ratio = 0.45
Root chord = 1.8008m
Tip chord = 0.8103m
MAC = 1.3681m

Vertical Tail Data:
Span = 2.0683m
Area = 3.0559m^2
Aspect ratio = 1.4
Taper ratio = 0.4
Root chord = 2.1106m
Tip chord = 0.8442m
MAC = 1.5678m

Horizontal Tail Data:
Span = 3.5321m
Area = 4.1588m^2
Aspect ratio = 3
Taper ratio = 1

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Root chord = 1.1739m
Tip chord = 1.1739m
MAC = 1.1739m

Fuel Tank Data:
W_f = 122.0356 kg
1 kg = 0.3675 gallons
So, W_f = (122.0356 x 0.3675) gallons = 44.8480 gallons
1 gallon of fuel fits in 0.135 ft^3
Therefore, the capacity of the fuel tank is, 44.8480 gallons = 6.0544 ft^3 = 0.1714 m^3

Tire Sizing Data:
Main Wheel Diameter = 13.57 inches
Main Wheel Width = 5.0911 inches
Nose Wheel Diameter = 8.028 inches
Nose Wheel Width = 3.1842 inches

4.5 Fuselage Lofting

The fuselage lofting is carried out by using a number of control points and connecting conic lines.

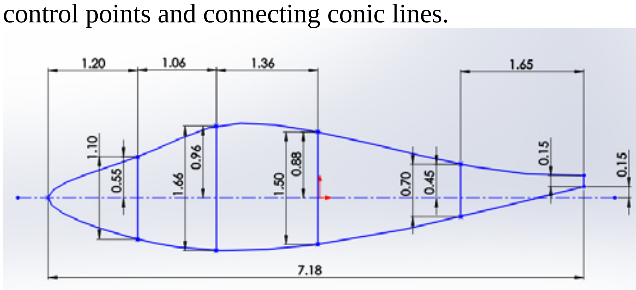

Fig. 4 Computer aided drafting of the fuselage lofting done in SolidWorks 2013

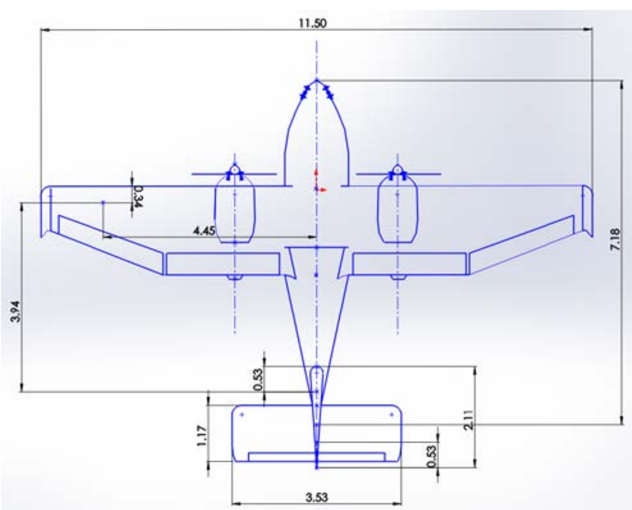

Fig. 5 Computer aided drafting for tail placement done in SolidWorks 2013

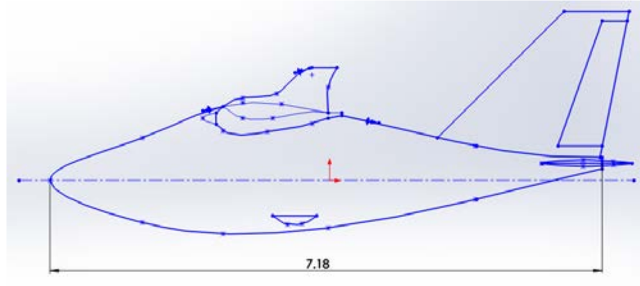

Fig. 6 Side View of MKTS-7

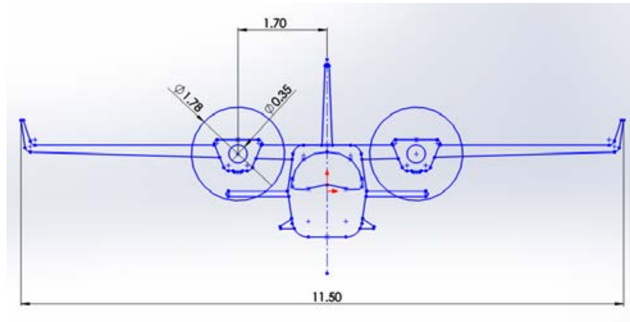

Fig. 7 Front View of MKTS-7

4.6 Initial Configuration Layout of MKTS-7

Since the lofting of fuselage and three views are generated using computer-aided drafting tools, we can now go for the final configuration layout of the aircraft. This will be much like the initial conceptual sketch drawn in Figure 1, but with a better dimensional accuracy. The configuration layout is illustrated in Figure 8.

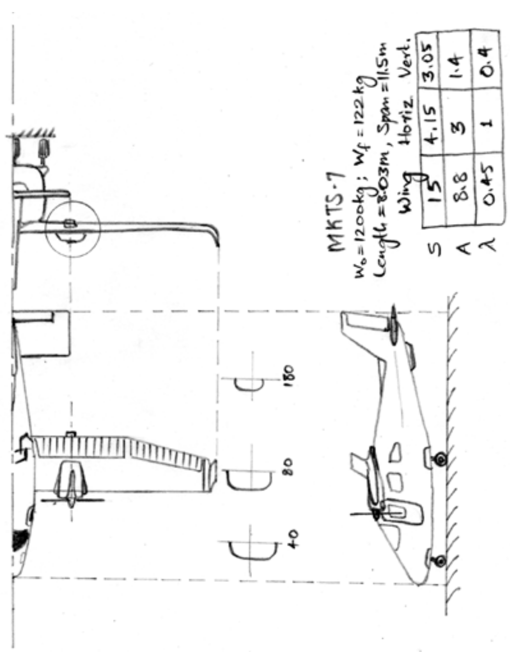

Fig. 8 Final Configuration layout of MKTS-7

The Wetted and Exposed areas determined using the drawing are as follows:
Exposed wing area = 7.3646m^2
Wetted area of main wing = 15.0768 m^2
Wetted area of tails = 11.3677 m^2
Wetted area of fuselage = 25.1872 m^2
Wetted area of engines = 4.1328 m^2

5. Theoretical Drag Estimation

In the previous sections, we have done the design layout of a credible aircraft configuration. The initial sizing was based upon rough estimates of the aircraft's aerodynamics, weights, and propulsion characteristics. In this chapter, we will calculate the aerodynamic forces acting on our aircraft, especially a theoretical estimate of the drag value. The wetted and exposed areas are obtained already in the section 4. We use those values for the calculations carried out here.

The subsonic lift-curve slope is given by,

$$C_{L_\alpha} = \frac{2\pi A}{2+\sqrt{4+\frac{A^2\beta^2}{\eta^2}\left(1+\frac{\tan^2 \Lambda_{\max t}}{\beta^2}\right)}}\left(\frac{S_{exposed}}{S_{ref}}\right)(F) \qquad (26)$$

C_{L_α} = 4.07809 per radian = 4.07809 x (π/180) = 0.07117 per degree

5.1 Parasite (Zero-lift) Drag

For estimating the subsonic parasite drag, we use the component buildup method.
The required data are:
Cruise velocity, V = 250 km/hr = 227.8359 ft/s
Maximum lift coefficient, $C_{L_{max}} \cong 0.9 C_{l_{max}}$ = 1.44
h = Sea level to determine friction C_f

The component buildup method estimates the subsonic parasite drag of each component of the aircraft using a calculated flat-plate skin-friction drag coefficient (C_f) and a component "form factor" (*FF*) that estimates the pressure drag due to viscous separation. Then the interference effects on the component drag are estimated as a factor "*Q*" and the total component drag is determined as the product of the wetted area, C_f, *FF*, *Q*.

Reynolds number, $R = \frac{\rho V l}{\mu}$ (27)

The subsonic drag coefficient is given by,

$$(C_{D_0})_{subsonic} = \frac{\sum\left(C_{f_c} FF_c Q_c S_{wet_c}\right)}{S_{ref}} + C_{D_{misc}} + C_{D_{L\&P}} \qquad (28)$$

This includes miscellaneous drag, $C_{D_{misc}}$ and drag due to leakages and protuberances, $C_{D_{L\&P}}$.

So, the total parasite drag coefficient is(sum plus 5% for leaks and protuberances),

$$C_{D_0} = 1.05 \text{ x } [(C_{D_0})_{fuselage} + (C_{D_0})_{wing} + (C_{D_0})_{tail} + (C_{D_0})_{engine}] \qquad (29)$$

C_{D_0} = 1.05 x [0.00756388 + 0.0063064 + 0.0033563 + 0.00095396]
C_{D_0} = 1.05 x 0.01818 = 0.01908

Adding the cooling drag and miscellaneous engine drag, the parasite drag coefficient becomes,
C_{D_0} = 0.01908 + 0.000706 + 0.000244427
C_{D_0} = 0.020039

5.2 Drag due to Lift (Induced Drag)

The induced-drag coefficient at moderate angles of attack is proportional to the square of the lift coefficient with a proportionality factor called the "drag-due-to-lift factor" or "*K*". Here, we follow the classical method based upon *e*, the Oswald span efficiency factor.

$$K = \frac{1}{\pi A e} \qquad (30)$$

The Oswald efficiency factor is typically between 0.7 and 0.85.
Oswald Span Efficiency, $e = 1.78\,(1 - 0.045\,A^{0.68}) - 0.64$
$e = 0.78853$ (31)
Drag-due-to-lift factor, $K = \frac{1}{\pi \text{ x } 8.8 \text{ x } 0.78853}$ = 0.04587
Design lift coefficient, $C_L = \frac{W/S}{q} = \frac{80}{\left(\frac{2953.86}{9.80665}\right)}$ = 0.26559
Induced Drag Coefficient, $C_{D_i} = K\ C_L^2$ = 0.04587 x (0.26559)^2 = 0.0032357 (32)

5.3 Total drag estimation

The component-wise parasite drag and their percentage from the total parasite drag can be shown as in Table 7.

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Table 7: Component-wise parasite drag and their percentage from the total parasite drag

Component	C_{D_0} of Component	Percentage from total C_{D_0} (%)
Fuselage	0.007563	37.74
Wing	0.006306	31.46
Tails	0.003356	16.74
Engines	0.000953	4.76
Cooling Drag	0.000706	3.52
Miscellaneous Drag	0.000244	1.21
Other Components	0.000915	4.57
Total C_{D_0}	0.020039	100

This component built-up parasite drag coefficient should now be added with the induced drag coefficient to get the total drag coefficient for the aircraft.

Total Drag Coefficient = Parasite Drag Coefficient + Induced Drag Coefficient

$$C_D = C_{D_0} + C_{D_i} = C_{D_0} + KC_L{}^2 \qquad (33)$$

$$C_D = 0.020039 + 0.0032357 = 0.02327$$

Using this C_D value, the Total Drag force can be calculated as,

$$D = \frac{1}{2}\rho V^2 S C_D \qquad (34)$$

$$= \frac{1}{2}\ x\ 1.225\ x\ 69.4444^2\ x\ 15\ x\ 0.02327$$

D = 1031.0226 N (we know that, Maximum L/D = 16)

Therefore, Lift force, L = 16 x 1031.0226 = 16,496.36 N

6. Computer-aided Drafting and Modeling

The modeling of the structural components are not necessary at this stage of design, so they are not included in this model. Only the exterior layout of the aircraft is designed as a single part which makes it convenient for being 3D printed. Each and every cross section of the fuselage and engines are lofted in the section 4 and those diagrams are used to develop the drawing into a three dimension model.

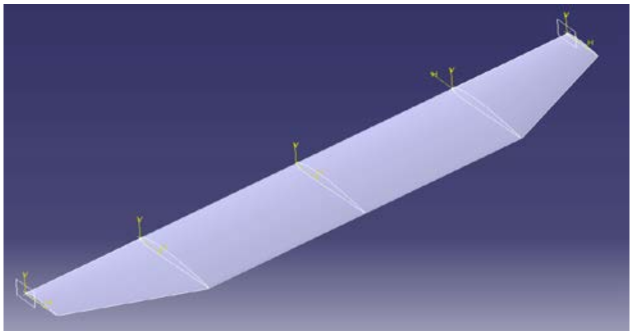

Fig. 9 Wing design done in CATIA V5R20

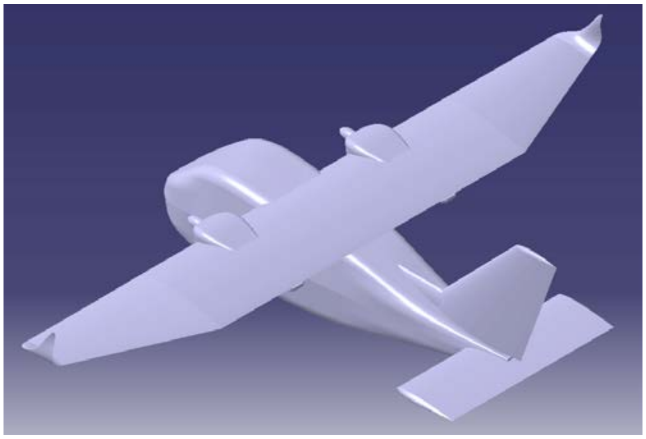

Fig. 10 CAD model of MKTS-7 done in CATIA V5R20

7. Rapid Prototyping and Experimental Drag Estimation of MKTS-7 model

3D printing or additive manufacturing is a process of making three dimensional solid objects from a digital file. The creation of a 3D printed object is achieved using additive processes. In an additive process an object is created by laying down successive layers of material until the entire object is created. Each of these layers can be seen as a thinly sliced horizontal cross-section of the eventual object. The 3D printing technique used for our model is Stereolithography.

7.1 Stereolithography (SLA)

The main technology in which photo-polymerization is used to produce a solid part from a liquid is SLA. This technology employs a vat of liquid ultraviolet curable photopolymer resin and an ultraviolet laser to build the object's layers one at a time. For each layer, the laser beam traces a cross-section of the part pattern on the surface of the liquid resin. The liquid resin used for this model is Accura 60, a clear (semi transparent) & tough plastic with the appearance of Polycarbonate. Exposure to the ultraviolet laser light cures and solidifies the pattern traced on the resin and joins it to the layer below.

After the pattern has been traced, the SLA's elevator platform descends by a distance equal to the thickness of a single layer, typically 0.05 mm to 0.15 mm (0.002″ to 0.006″). Then, a resin-filled blade sweeps across the cross section of the part, re-coating it with fresh material.

On this new liquid surface, the subsequent layer pattern is traced, joining the previous layer. The complete three dimensional object is formed by this project.

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Stereolithography requires the use of supporting structures which serve to attach the part to the elevator platform. This technique was invented in 1986 by Charles Hull, who also at the time was the founder of the company, 3D Systems.

7.2 1:100 Scaled model of MKTS-7

The figures below shows the final 1:100 scaled down model of the MKTS-7 aircraft. The process of four and a half hours took the virtual model in the digital world to be a perfectly scaled physical object held in the hands. The model weighs around 16.5 grams.

Fig. 11 The final 3D printed model of MKTS-7 aircraft

7.3 Wind Tunnel Testing

The experiment was done in the subsonic wind tunnel situated at the aerodynamics laboratory at the Hindusthan Institute of Technology, Coimbatore on 25th March 2015. This tunnel can run at a maximum test section airflow velocity of 23 m/s. This wind tunnel is an open non-return type tunnel operated by a suction type electrically driven fan at the rear section of the tunnel. Air is drawn from the room into a large settling chamber fitted with a honeycomb and several screens. The honeycomb is there to remove swirl imparted to the air by the fan. The screens break down large eddies in the flow and smooth the flow before it enters the test section. Following the settling chamber, the air accelerates through a contraction cone where the area reduces (continuity requires that the velocity increase). The test (working) section is of constant area (300 mm x 300 mm).

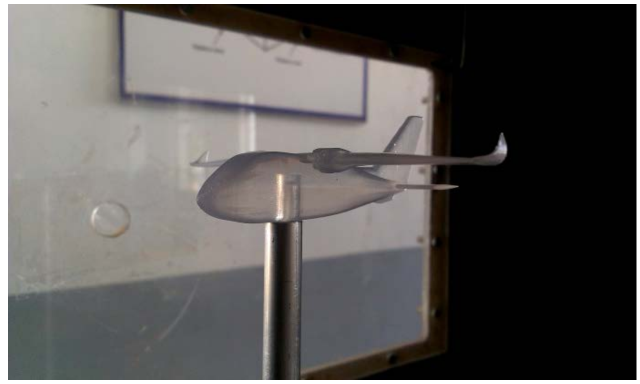

Fig. 12 The MKTS-7 model mounted on the support rod within the test section of the subsonic wind tunnel

7.4 Drag Estimation

Velocity constraint faced for scaled model:
Reynolds Number,

$$R_e = \frac{\rho V l}{\mu} = \frac{1.225 \text{ x } 69.4444 \text{ x } 7.1774}{0.00001789} = 34129515.92$$

Using the same Reynolds number for the scaled model, we can determine the velocity required in the wind tunnel for the drag estimation.

$$34129515.92 = \frac{\rho V_{scaled}\, l}{\mu} \tag{35}$$

$$34129515.92 = \frac{1.225 \text{ x } V_{scaled} \text{ x } 0.0803}{0.00001789}$$

$$V_{scaled} = 6207.101327 \text{ m/s}$$

This is a very high velocity value, meaning that for the 1:100 scaled model to experience the same magnitude of drag forces as a full scaled aircraft, the wind speeds in which it needs to run in the wind tunnel would need to be almost 90 times the wind speed needed for the actual full scale aircraft. This comes to a Mach number of 15.89, requiring an hypersonic wind tunnel. We only have a subsonic wind tunnel available for testing. Thus, we can conclude that this model is too small for a subsonic wind tunnel analysis. Now, we can plot the variation of the drag coefficient value with the velocity increment.

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Table 8: Tabulation for drag calculation from the subsonic wind tunnel

Fan Speed	Dynamic Pressure	Flow Velocity	Lift	Drag	Coeff. of Drag
RPM	mm/H_2O	m/s	kgs	kgs	C_D
276	2	3	0	0	0
424	4	6	0	0	0
594	9	9	0	0	0
757	15	12	0	0	0
935	23	15	0	0.01	0.1237
1109	33	18	0.01	0.01	0.0859
1176	37	19.5	0.01	0.02	0.1464
1286	45	21.5	0.01	0.03	0.1807
1381	52	23	0.02	0.04	0.2105

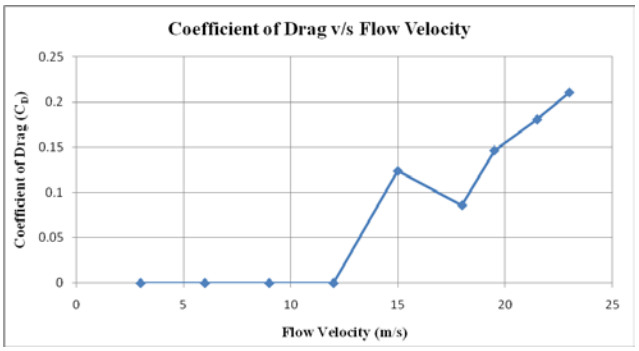

Fig. 13 Coefficient of Drag v/s Flow velocity

8. Result and Discussions

8.1 General Specifications of MKTS-7

- Crew - 2
- Seating capacity - 4
- Capacity - 380 kg payload with full fuel
- Length - 8.03 m
- Wing Span - 11.50 m
- Height - 2.90 m
- Wing Area - 15 m^2
- Empty Weight - 772 kg
- MTOW - 1,200 kg
- Fuel Weight - 122 kg
- Power plant - 2 x Rotax 912S3 98.6hp (73.5 kW)
- Propellers - 2-bladed propeller, 1.78 m diameter

Aerodynamic Characteristics:

- Total Drag - 1.03 kN
- Thrust Force - >1.03 kN
- Lift Force - 16.5 kN
- Weight Force - 11.7 kN

8.2 Performance Specifications of MKTS-7

- Maximum Speed - 300 km/hr
- Cruise Speed - 250 km/hr
- Stall Speed - 172 km/hr
- Never Exceed Speed - 309 km/hr
- Max. Control Speed - 115 km/hr
- Range - 1,500 km
- Endurance - 4.5 hrs
- Service Ceiling - 4,200 m, 13,800 ft
- Rate of Climb - 8 m/s
- Wing loading - 80 kg/m^2 at MTOW
- Power loading - 6.085

8.3 Design Summary

- The overall design was relatively successful. As indicated in the general and performance specifications, most of the characteristics that were initially proposed for MKTS-7 were closely met in the design.
- The design was having the proposed weight with optimized geometric characteristics. It uses the next version of the Rotax engine which customizes our design and the engine is perfectly matched without any scaling.
- This design should then be an initiative for a better twin engine propeller aircraft for training purpose.

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9. Conclusions

This paper can be concluded with the completion of the initial sizing of the twin engine trainer aircraft. Most of the general and performance specifications are acquired from the existing data of the Tecnam P2006T. The project has to be extended further with lot of iterative procedures to optimize and achieve the design proposals given in section 1.1. The initial sizing process carried out in this project gives only the closer values of wing loading, power loading which gives optimized gross takeoff weight and wing area. Geometric sizing was done to initially layout the configuration and model the aircraft in a 3-dimensional computer aided drafting software. This model was scaled down to a factor of 1:100 for printing in additive manufacturing techniques. The intent is to show the application of rapid prototyping in the initial design stages of an aircraft. This model was tested in a subsonic wind tunnel to see the variation of the drag with increase in the velocity. Although, we faced few constraints to learn the aerodynamic characteristics, as the model was too small for testing, this was an efficient attempt to learn the techniques of the 3D printing technology and the ease of testing at the early stages of aircraft design.

This project can be worked on further at the post-graduate level, as numerous optimizations have to be carried out to get the conceptual design completed. Further, this project presents the theoretical and experimental drag estimation methods. The CAD model created in this project can be used for the numerical aerodynamic analysis, and the 3D printed model will be preserved as a specimen for each significant stage of design.

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Mukesh.K is a Graduate in Aeronautical Engineering from the Hindusthan Institute of Technology, Coimbatore affiliated to Anna University, Chennai. He worked for the final year project under the guidance of S.Tamilselvan. A team lead by him was awarded the first position in the school level science exhibition in the year 2010 for demonstrating the 'solar updraft tower' for energy generation. He has attended an International conference on ‘Additive Manufacturing and 3D Printing’ conducted by the ‘ICAM 3D’, a National level conference on ‘Advanced Technology as Change Agent to make India an Economic Superpower’ conducted by the ‘ITC-2014’ and a National Workshop on ‘Sustainability through Green Chemistry and Catalysis’ conducted by ‘The Royal Society of Chemistry – South India’. Currently he is pursuing his Post-Graduation in Aeronautical Engineering in the Park College of Engineering and Technology, Coimbatore.