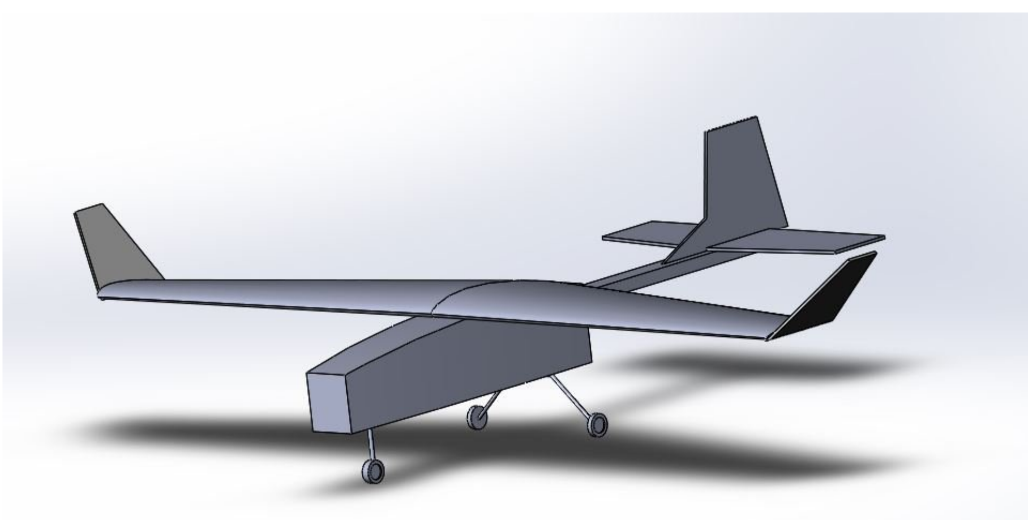
Manjara Charitable Trust's

Rajiv Gandhi Institute of Technology

TEAM NAME:

RAGING RAPTORS

TEAM: #214



SAE AERO-DESIGN WEST COMPETITION 2013

SAE

Appendix
2013 SAE AERO DESIGN

STATEMENT OF COMPLIANCE
Certification of Qualification

Team Name RAGING RAPTORS Team Number 214

School RAJIV GANDHI INSTITUTE OF TECHNOLOGY

Faculty Advisor Dr. Kiran Chaudhari

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Statement of Compliance

As Faculty Advisor, I certify that the registered team members are enrolled in collegiate courses. This team has designed, constructed and/or modified the radio controlled airplane they will use for the SAE Aero Design 2013 competition, without direct assistance from professional engineers, R/C model experts or pilots, or related professionals.

KChaudhary.
Signature of Faculty Advisor

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LIST OF SYMBOLS AND ACRONYMS

Symbol		Unit
L	Lift force of wings	N
C_L	Coefficient of lift	rad^{-1}
S	Area of wings	sq.in
v	Absolute velocity of wings	ft/s
ρ	Density of air	lb/ft^3
α	Angle of attack of aircraft	rad
A_0	Angle of incidence of aircraft	rad
MAC	Mean Aerodynamic Chord	in
CG (X)	Centre of Gravity of aircraft	-
e	Oswald's efficiency factor	-
V_{STALL}	Stall speed of aircraft	ft/s
W/S	Wing loading	lb/ft^2
C_D	Co-efficient of drag	rad^{-1}
B.F.	Blade Factor	-
S.T.	Static Thrust	lb
D.T.	Dynamics Thrust	lb
TAS	True Air Speed	ft/s
AC	Aerodynamic Center	-
LE	Leading Edge of the wing	-
TE	Trailing Edge of the wing	-
mAh	Milli-ampere hour	-
δ	Deflection of flaps	rad
g	Acceleration due to gravity	ft/s^2
S_H	Horizontal tail area	sq.in
S_v	Vertical tail area	sq.in
V_h	Horizontal tail volume co-efficient	-
V_v	Vertical tail volume co-efficient	-
b	Wing span	in

L_H	Distance from horizontal tail's aerodynamic centre to the aircraft CG	In
L_v	Distance from vertical tail's aerodynamic centre to the aircraft CG	in
NTSC	National Television Standards Committee	-
PAL	Phase Alternating Line	-
PWM	Pulse Width Modulation	-
n_{max}	Maximum load factor	
C	Control surface chord	in
H	Control surface length	in
Z1	Max control surface deflection	deg
Z2	Max servo deflection	deg
η	Load factor	-
η_{max}	Maximum load factor	-
θ_{max}	Maximum banking angle	rad
V_{TO}	Velocity at takeoff	ft/s
T	Thrust force	N
μ	Coefficient of friction of aircraft wheels	-
W	Aircraft weight	lbs
S_{wet}	Wetted area	sq.in
R_e	Reynolds no	-
fm	Function of mach no	-
C.F	Coefficient of propeller	-
C_{di}	Coefficient of induced drag	-
C_f	Skin friction coefficient	-

1. Introduction

The SAE Aero design series aims to give a real world engineering opportunity to design and fabricate an aircraft to various universities across the world. The design engineering goal of the Advanced Class is to design the most efficient aircraft capable of accurately dropping a three pound (3 lb) humanitarian aid package from a minimum of 100ft off the ground. This task needs to be carried out while simultaneously carrying ideally 15 pounds worth of static payload. This engineering goal can only be met by applying principles of aerodynamics, material sciences, physics and other principles. There are various other challenges pertaining to the design like planning and management, budget etc. In order to successfully complete the given task, the team has to design, test and enhance it along the way.

1.1 Objective

Team Raging Raptors aims at designing and fabricating an aircraft which maximizes lift, carry the stated payload and drop the expellable cargo with maximum precision while maintain a robust, light weight and easy to fabricate design. With a combination of research, testing and analysis, team Raging Raptors is meeting the objective.

1.2 Requirement Statement

Wing	No lighter-than-air or rotary wing aircraft.
Gross Weight	Maximum 26 pounds of weight to be lifted for maximum flight points.
Engine	O.S Max 46AX Engine.
Payload	15 pounds of static cargo weight and 3 pounds of expellable cargo weight.
DAS requirement	• Real time altitude reading at the ground station. • Record the altitude while releasing the expellable cargo.
Material	• Lead payload plates are prohibited. • Commercially available engine and propeller only. • Use of FRP is prohibited.
Other special requirements	First Person View (FPV) for dropping the expellable cargo.
Battery	Battery pack should be no less than 1000 mAh.
Safety requirements	• Metal propellers are not allowed. • All aircraft must utilize either a spinner or a rounded safety nut.

2. Design Process

2.1.1 Discussion of Concepts

The team made a thorough research on various aspects of the design. Compared to last year's .61 size engine, the use of a .46 size engine to lift a dead weight of eighteen pounds (18 lb) **for an optimum score**, was a very challenging task. The team had to think of various ways of minimizing the empty weight while maintaining a robust, stable and easy to fly design. Another major challenge was dropping a 3 pound expellable cargo package on the target accurately.

The following list outlines the goals that were implemented into our design:

- Minimizing the empty weight of the aircraft.
- Designing a simple mechanism for expellable cargo.
- Designing a proper landing gear to resist the impact.
- Optimizing the thrust of aircraft.
- More lift during take-off and slow speeds while landing.
- Easy to transport design.
- Proper placing of DAS and FPV system for accurate results.

To understand and meet all the goals mentioned above, a detailed research was conducted. A detailed explanation of the research and conclusions drawn from them are mentioned below.

2.1.2 Initial Aircraft Configuration

The team initially considered the following aircraft configurations:

1. Conventional	2. Bi-plane
3. Delta	4. Tail Boom

Out of these four configurations, the delta configuration was directly eliminated because of its bad performance at low speeds. The bi-plane configuration was ruled out only because it produced excessive drag. Finally, the remaining two configurations were considered appropriate for the competition, of which the team decided to use the tail boom configuration because it was comparatively lighter and had lower drag.

3. Design Analysis, Review Process and Selection

3.1 Material

In order to reduce the weight of the aircraft and simultaneously maintain its strength, a proper material had to be chosen. One of the major designing goals was to reduce the empty weight of the aircraft without reducing its structural strength, which was achieved by reinforcing balsa with thin plywood. The wing of the aircraft should be strong and at the same time be very light. The team decided to use balsa cut ribs (1/16 inch)and then strengthen the root rib by reinforcing balsa with thin ply (1/16 inch).To provide more geometrical strength ,the team decided to use a front and a rear brace. The spar material used was 1/4 inch balsa.

The material selection for fuselage was based on strength, weight, cost, availability and the ease of fabrication. The fuselage material was decided keeping in mind that static payload had to be kept inside the fuselage. The overall stress on the fuselage formers was very large and hence we used doped balsa laminated with thin ply, which provided enormous strength. This composite material was of great advantage to us in reducing the overall weight of the aircraft. The firewall was made of plywood and the covering material for wing and fuselage was decided to be silver film. The material chosen for landing gear was spring steel compared to aluminium, as it provided more resistance to impact. The boom material used was aluminium because of its high strength to weight ratio, low cost and ease of availability.

3.2 Wing

Wing designing is the most important aspect in the aircraft designing process since the wing features decide the performance of the aircraft during flight. While designing the wing, the main factors considered were:

1. Airfoil selection
2. Planform shape
3. Planform area
4. Aspect Ratio

3.2.1 Airfoil Selection Process

Choosing an appropriate airfoil is very important in the wing designing process. While selecting an airfoil the team took three factors into consideration: lift, drag and pitching moment. After comparing various airfoils the team shortlisted the following: Wortmann FX 63-137, S-1223, Clark-Y, S-1210 and NACA 6411.

NACA 6411 and Clark-Y were then selected for further analysis over other airfoils, as other high lift airfoils produced high induced drag and pitching moment. Clark- Y was chosen due to its high C_l/C_d ratios at cruise C_l of 0.9 . Clark-Y also provided other benefits such as less pitching moment and is easy to fabricate . Clark-Y has mild stall characteristics.

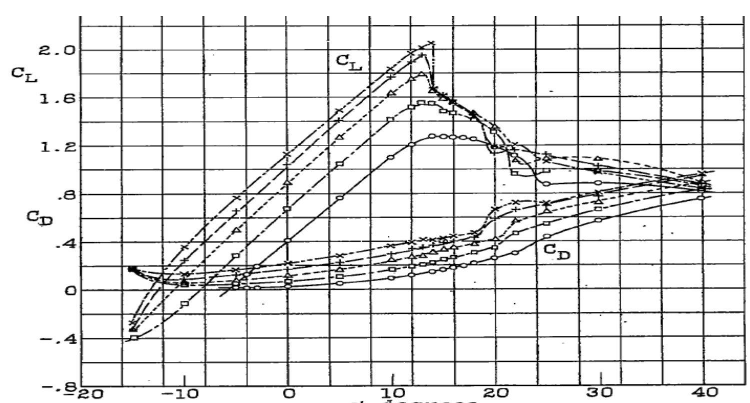
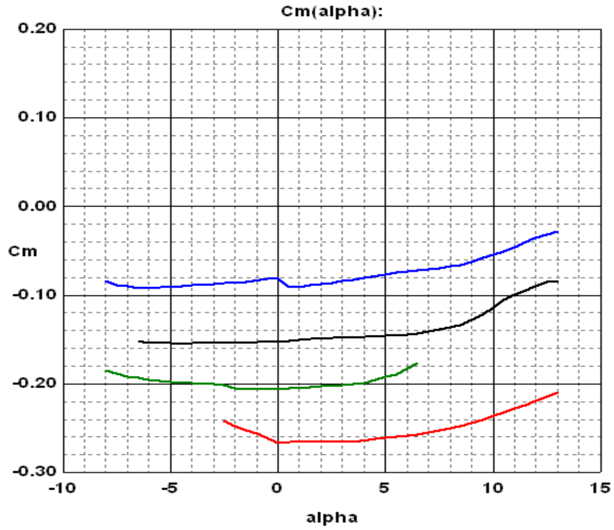
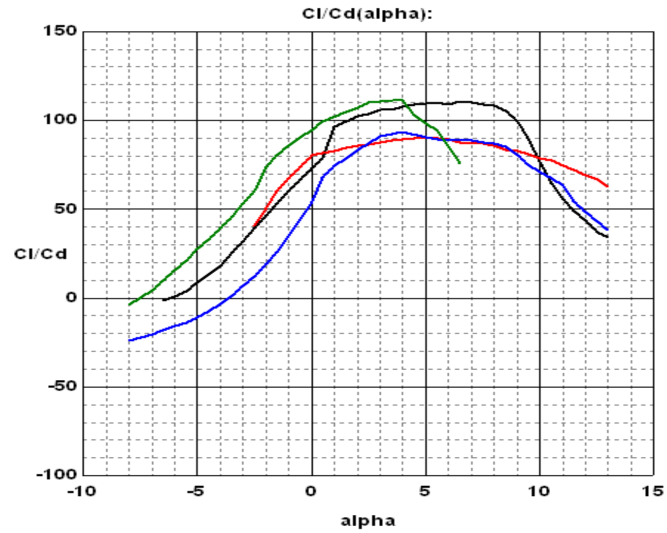


Fig. 3: Effect of plain flaps on Clark-Y with flap to chord ratio of 0.15 .





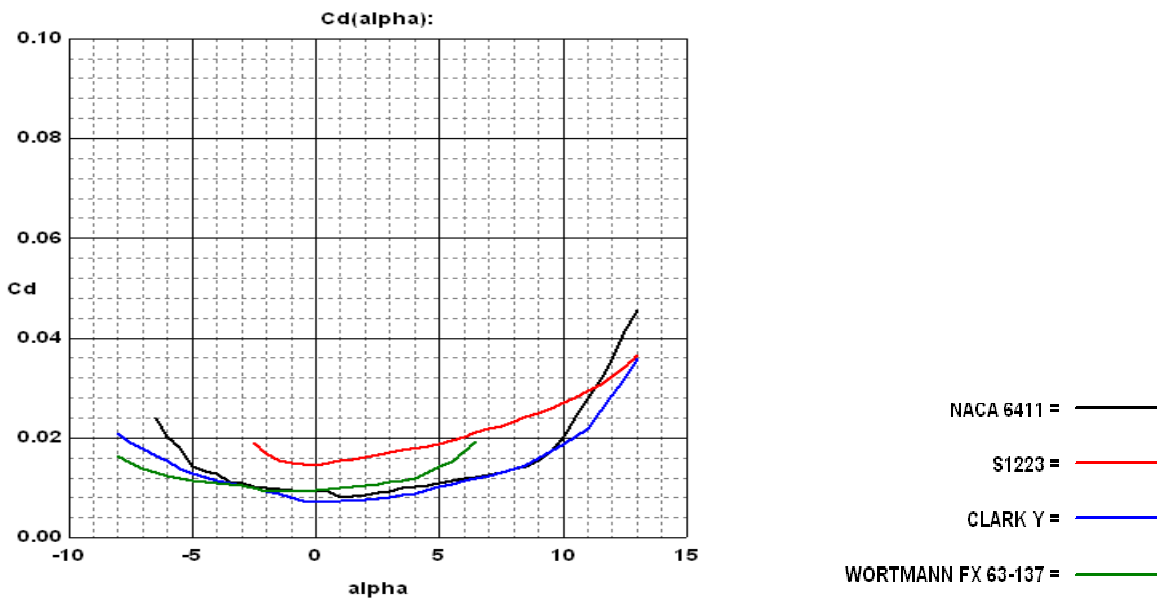


Fig 4: Graph of C_m v/s **α.**
Fig 5: Graph of C_l/C_d v/s **α.**
Fig 6: Graph of C_d v/s **α.**

3.2.2 Planform Shape

The planform shape affects various parameters such as span efficiency, vortex drag, stress, flutter, etc. The two planforms in consideration were a rectangular wing and a front swept tapered wing. Rectangular wings have better stall characteristics and are easier to fabricate. However they create more induced drag and hence are less efficient. Since the tips do not generate much lift, hence the extra area at the tips of a rectangular planform adds to more induced drag and increases wing loading.

The tapered wing, although difficult to fabricate in comparison, is more efficient (close to elliptical lift distribution). It has more area near the root and less at the tips thus reducing the wing loading. The wing was made swept forward such that the leading edge of the wing remains straight. Air flowing over swept forward wing tends to move inward thus increasing flap efficiency.

As a result, the dangerous tip stall condition of a backwards-swept design becomes a safer and more controllable root stall on a forward swept design. This allows full aileron control despite loss of lift, and also means that drag-inducing leading edge, slats or other devices are not required. The tip rib was was placed at an angle of 45 degrees to delay the tip stall and hence make the wing more resistant to stalling.

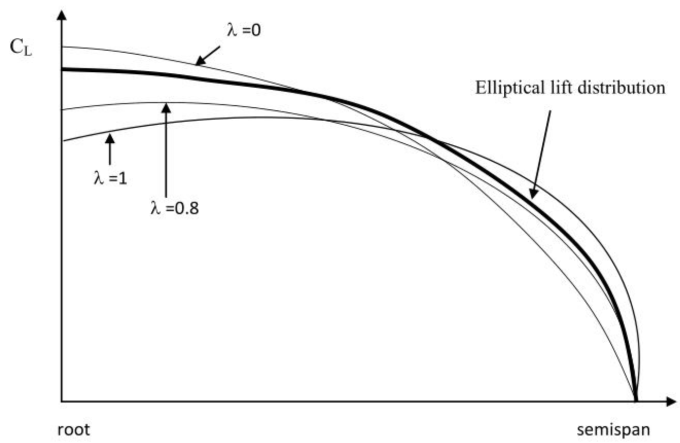


Figure 5.30. The typical effect of taper ratio on the lift distribution

3.2.3 **Planform Area**

Planform area is the parameter which controls the amount of air the wing pushes downwards, thus generating lift. As per the lift formula, a suitable planform area of 735 sq.inches was calculated. This was obtained by keeping the root chord as 14 inches and tip chord as 7 inches (taper ratio of 0.5). The span was decided to be 70 inches.

3.2.4 Aspect Ratio

A higher aspect ratio wing provides more stability and helps reducing the wing loading. They also reduce the vortex drag created by the wing. But, however create more flex and flutter during flight. Considering all the above mentioned factors an aspect ratio of 6.67 was obtained. Thus, the design team considered the aspect ratio to be appropriate for performing the required mission.

3.2.5 Positioning of wing

The team decided to use high-wing configuration because of the following reasons:

- While take off or moving to higher altitude, in low-wing planes the vertical alignment of CG and resultant lift changes which creates a pitching moment in one direction i.e. in clockwise direction as show in fig. this leads to nose up phenomenon .This is eliminated in high-wing as both CG and resultant lift produces moment in opposite direction.

- Compared to mid wing, high-wing does not occupy any inner space of fuselage. Its attachment is less complicated.
- A high-wing configuration has more lateral stability as compared to mid-wing and low-wing configuration due to the “pendulum effect”.
- A low-wing and mid-wing has low ground clearance compared to high-wing configuration as a result of which high-wing aircraft remains safe while improper landing.
- Since a high-wing is supported entirely on the fuselage, hence the stress on the wing root particular decreases.

3.2.6 Flaps

Since the aircraft had to carry more load , the team decided to use flaps which provided the following advantages.

- Extending the flaps lowers the stalling speed.
- Extending the flaps increases the wing’s angle of incidence.
- Extending the flaps effectively increases the washout, since on most planes the inboard sections have flaps while the outboard sections do not.
- Extending the flaps increases drag. This is helpful during landing, but unhelpful during climb and cruise.
- Extending the flaps gives the airfoil a shape that is more resistant to stalling. That means, among other things, that it can fly at a higher angle of attack without stalling
- Thus flaps provide a lower stalling speed, which is important for safety as well as performance.
- Increased washout increases roll damping so the airplane handles more nicely near the stall.

3.3 Fuselage

The team constantly kept a check over the empty weight of 8 lbs and at the same time considered making a fuselage which could be easily dismantled. Sticking to the objective, the team decided to keep the weight of the fuselage as low as possible and at the same time make it more aerodynamic. The team decided to consider the following points while designing the fuselage.

- Ease of construction.
- Reducing the weight of the fuselage.
- Ease of loading and unloading the components.
- Strong enough to bear the impact while landing.

The team decided to use a boom mounted fuselage over a conventional fuselage so as to reduce the overall area and hence reduce parasitic drag. Using a boom would reduce the weight by a considerable amount. The use of a rectangular cross section was the most appropriate choice as it provides more geometrical strength and at the same time is easy to construct. The team decided to use balsa wood (1/4 inch) doped with nitrocellulose and then laminated it with ply (1/16 inch) which increased the shear strength of the material. The two formers were made out of this composite material. The side walls of the fuselage were made by using warren trusses. Trusses have a high strength-to-weight ratio, are extremely rigid and are very light. To resist the impact while landing, the area below the undercarriage had to be made strong. Here the team decided to use 1/4 inch plywood.

3.3.1 Cargo Bay

Static cargo bay

The team chose the static cargo material as mild steel plates. The dimension of each plate was decided to be 4x2x0.2 inch. Each static cargo plates weighed around 0.9 lb to 1.1 lb. Static cargo plates were placed under the wing so as to have the least pitching moment.

Expellable cargo

The team met the requirement of using sand enclosed by fabric sewn material. The expellable cargo mechanism was activated using a servo and a spring mechanism.

3.4 Tail

The horizontal and vertical tail moment arms were determined using the volume co-efficient method. The moment arm of tail was limited to the fact that a longer tail created take off rotation clearance problems. The horizontal tail was sized in such a way that it could counter the negative pitching moment of the wing. The team decided to use a flat plate tail without airfoil was used to counter the negative effects that occur at such slow speeds.

The tail volume co-efficients were determined using the following realtions :

Horizontal tail volume co-efficient : $V_H = S_H \times L / (S \times MAC)$

Vertical tail volume co-efficient : $V_v = S_v \times L / (S \times b)$

Considering the horizontal tail volume co-efficient to be 0.51 and vertical tail volume co-efficient to be .03, the horizontal and vertical tail areas were determined.

The initial tail configuration was based on the following factors:

FIGURE OF MERIT	WEIGHTAGE FACTOR	CONVENTIONAL TAIL	T-TAIL	H-TAIL
STABILITY	.30	4	4	5
EASE OF CONSTRUCTION	.15	5	3	2
DRAG	.15	3	3	2

MANEUVERABILITY	.20	3	4	5
WEIGHT	.20	4	4	3
TOTAL	1.00	3.85	3.7	3.7

Table no.1: Comparing different tail configurations

3.5 Landing Gear

The Landing gear should be easy to fabricate, strong, lightweight and should have dampening effect to dampen the jerk produced while landing. The team had the following advantages of choosing a tricycle gear configuration.

- Reduced possibility of mishaps like nose over and ground looping. This is due to main wheels being behind CG.
- It eliminates the unwanted nose up condition which produces more parasitic drag.
- Tricycle gear aircraft are easier to land because they are more stable while landing and do not roll over from the wing tips.
- It will provide more clearance for external expellable payload while taxing and dropping.
- The payload weight gets equally distributed among the three wheels and hence there is minimal landing deflection.

Design Factors	**Weighting factors**	**Tricycle**	**Tail-dragger**
Ground control	.20	5	2
Weight	.10	2	5
Drag Efficiencies	.20	2	5
Stability	.20	5	2
Ease of fabrication	.10	3	3
Minimal landing deflection	.20	5	2
Total	1.00	3.9	3

Table No 2: Comparing different landing gear configurations

3.6 DAS

A new challenge is introduced in this year's event to introduce Data Acquisition Systems (DAS) which is interface between the real world of physical parameters, which are analog, and the artificial world of digital computation and control. With current emphasis on digital systems, the interfacing function has become an important one; digital systems are used widely because complex circuits are low cost, accurate, and relatively simple to implement. In addition, there is rapid growth in the use of microcomputers to perform difficult digital control and measurement functions. Computerized feedback control systems are used in many different industries today in order to achieve greater productivity in our modern industrial societies.

This is a very vast field of development with lots of potential and possibilities. But to meet the requirements and Due to lack of large sum and time as well as experience, we preferred to use easier components and methods to design this system in this particular project.

Components:

1.Laser Distance measuring Sensor
2.Gyroscope and Accelerometer Sensor
3.Atmega 16/ Arduino Micro-Controller
4.Transmitter - Receiver Kit
5.Display kit/ Laptop

Laser Distance measuring Sensor

Bosch GLM50 Laser Range Finder

Product Specification:

Barcode: 3165140600781
Measurement range 0,05 - 50 m
Measurement accuracy, typical ± 1.5 mm
Measuring time, max. 4 s
Weight, approx. 0,14 kg
Dimensions: 115 mm x 53mm x 32mm

Laser diode 635 nm, < 1 mW
Laser class 2
Measurement time, typical < 0.5 s
Power supply 2 x 1.5 V LR03 (AAA)

Gyroscope and Accelerometer Sensor

The output from the Laser sensor is actually vertical distance between sensor and ground where the laser sensor is mounted on the RC Aircraft. As we know Aircraft will tilt\bank in all possible directions thus resulting in inaccurate altitude readings. Hence Gyroscope and Accelerometer Sensors are used.

An accelerometer measures acceleration. A 3-axis accelerometer will tell you the orientation of a stationary platform relative to earth's surface, once that platform starts moving; however, things get more complicated. If the platform is in free-fall, it will show zero acceleration. If it is accelerating in a particular direction, that acceleration will simply be added to whatever acceleration is being provided by gravity, and you will not be able to distinguish.
A gyro measures rate of rotation around a particular axis. If a gyro is used to measure the rate of rotation around the aircraft roll axis, it will measure a non-zero value as long as the aircraft is rolling, but measure zero if the roll stops. So, a roll gyro in an aircraft in a coordinated turn with a 60 degree bank will be measure a rate of zero, same as an aircraft flying straight and level. You can approximate the current roll angle by integrating the roll rate over time, but you can't do so without some error creeping in. Just to make life more interesting, gyros drift with time,

so additional error will accumulate over a period of minutes or even seconds, and eventually, you'll have a totally inaccurate idea of your current roll angle relative to the horizon. So, gyros alone can't be used to keep in an aircraft in a particular orientation.

So, in a nutshell: Accelerometers are right in the long term and Gyros are right in the short term.

1. LSM303DLHC with accelerometer and 3D compass

Dimensions:

-Size: 0.5″ × 0.8″ × 0.1″ -Weight: 0.6 g

General specifications

Interface:	I²C
Minimum operating voltage:	2.5 V
Maximum operating voltage:	5.5 V
Measurement range:	±2, ±4, ±8, or ±16 g (accelerometer) ±1.3, ±1.9, ±2.5, ±4.0, ±4.7, ±5.6, or ±8.1 gauss (magnetometer)
Supply current:	10 mA

2. ST L3GD20 three-axis gyroscope

Dimensions

Size: 0.5″ × 0.9″ × 0.1″ Weight: 0.7 g

General specifications

Interface:	I²C, SPI
Minimum operating voltage:	2.5 V
Maximum operating voltage:	5.5 V
Axes:	pitch (x), roll (y), and yaw (z)
Measurement range:	±250, ±500, or ±2000°/s
Supply current:	7 mA

ATmega 16/ Arduino Micro-Controller

ATmega 16

ATmega16 is an 8-bit high performance microcontroller of Atmel's Mega AVR family with low power consumption. Atmega16 is based on enhanced RISC (Reduced Instruction Set Computing) architecture with 131 powerful instructions. Most of the instructions execute in one machine cycle. Atmega16 can work on a maximum frequency of 16MHz. ATmega16 has 16 KB programmable flash memory, static RAM of 1 KB and EEPROM of 512 Bytes. The endurance cycle of flash memory and EEPROM is 10,000 and 100,000, respectively.ATmega16 is a 40 pin microcontroller. There are 32 I/O (input/output) lines which are divided into four 8-bit ports designated as PORTA, PORTB, PORTC and PORTD.ATmega16 has various in-built peripherals like USART, ADC, Analog Comparator, SPI, JTAG etc. Each I/O pin has an alternative task related to in-built peripherals. The following table shows the pin description of ATmega16.

Arduino

The Arduino Uno is a microcontroller board based on the ATmega328.It has 14 digital input/output pins (of which 6 can be used as PWM outputs), 6 analog inputs, a 16 MHz crystal oscillator, a USB connection, a power jack, an ICSP header, and a reset button. It contains everything needed to support the microcontroller; simply connect it to a computer with a USB cable or power it with a AC-to-DC adapter or battery to get started.

Specifications:
Microcontroller ATmega328
Operating Voltage 5V
Input Voltage (recommended) 7-12V
Input Voltage (limits) 6-20V
Digital I/O Pins 14 (of which 6 provide PWM output)
Analog Input Pins 6
DC Current per I/O Pin 40 mA
DC Current for 3.3V Pin 50 mA
Flash Memory 32 KB of which 0.5 KB used by bootloader
SRAM 2 KB

EEPROM 1 KB ,Clock Speed 16 MHz

We are using Arduino/ATmega16, as the output from our sensors are in analog/digital format as well as un-calibrated with respect to each other, So as to sync them and get the desired output particular Micro-controller is necessary.

Transmitter - Receiver Kit

Refer FVP section for further details as Transmitter-Receiver Kit is shared by both

Display kit/Laptop

Preferably laptop is used to provide the digital readings of FVP and DAS system with high accuracy.

3.7 FPV

Another challenge put up this year is to have a First Person View System (FPV) in the aircraft for mainly precise cargo drop application.

Components:
Camera
Transmitter and Receiver Kit

Transmitter and Receiver:
Specifications:

TS351+RC805	New version 5.8G 200mW FPV kit
Frequency	5645-5945 MHz; 8CH
Distance	Open area (>2000 m)
Power supply	12 V
Weight	0.40 lbs

Camera:
CM-3434C

Mini Color Camera
1/4" Sharp Color CCD
1.0 Lux , 420TVL
S/N Ratio : More than 48dB
Number of Pixels : PAL: (H)500x(V)582
NTSC: (H)510x(V)492

This particular Transmitter-Receiver kit is selected as it has wide frequency range and the separate system helps us in avoiding the interference with the radio frequency (2.4 GHz). It is also lighter as compared to other similar systems. The Camera is selected in accordance with payload dropping mechanism. The resolution of camera is good along with it’s wide angle of sight.

FPV Selection Process

This particular FPV system was selected based on the fact that the S/N ratio of this system is more than 48db. The S/N ratio decides the resolution and video quality based on the interference with electromagnetic field passing through the air.

Electrical Differences

The main difference starts with the electrical power system that runs behind the color transmissions. In the NTSC power required is 60Hz whereas in PAL required power is 50Hz.

- Depending upon the configurations decision was taken to use TS351+RC805 New version 5.8G 200mW FPV kit.
- This particular FPV was taken due to its low price and was readily available in the market.

4. Calculations

4.1 Performance Prediction

LOAD FACTOR (N)

The load factor is a ratio of the lift of an aircraft to its weight and represents a global measure of stress to which the structure of the aircraft is subjected. By calculating the lift at the cruise velocity where the aircraft would achieve stability, we determine the lift to be 23.5 lbs of force .we can find the load factor using the following relation.

$$n = \frac{L}{W_{empty}} = \frac{23.5}{5.5} = 4.27$$

With regards to tuning performance, it is advisable to design the aircraft with maximum possible load factor as this determines the turning performance of the aircraft .The following equations can be used to compute maximum load factor.

$$n_{max} = 0.5 * \rho_\infty * V_\infty{}^2 * \frac{C_L}{\frac{W_{empty}}{S}} = 0.5 * 0.002377 * 55^2 * \frac{1}{0.9429} \approx 5.33$$

The maximum load factor can be used to calculate the maximum safe banking angle of an aircraft. The following expression is used to calculate the banking angle

$$\cos(\theta_{max}) = \frac{1}{n_{max}}$$

On further calculation it can be seen that maximum banking angle is 72 degree ,which implies that our aircraft has decent stability while making turns at a cruising speed of 55 fps.

WING LOADING

The wing loading is the weight of the aircraft divided by the area of the reference wing. The wing loading directly affects stall speed, climb rate, turn performance, and take-off and landing distances. The sizing of the wing is heavily influenced by the wing loading—if the wing loading is reduced, the wing is larger. The wing loading and thrust-to-weight ratio must be optimized together in order to ensure the tradeoff between better aerodynamic performance and increases in drag and weight is not detrimental to the functionality of the aircraft. Wing loading (W/S) can be found using the following equation:

4.1.1 Take off Performance

TAKE-OFF DISTANCE

Takeoff distance is a distance travelled by the plane the on runway till the main wheels leave the ground i.e., plane gets airborne .The takeoff distance is generally calculated for maximum weight of the aircraft under standard conditions.

Takeoff distance was calculated by using the following formula:

$$X_g = \frac{W}{(C_D - \mu C_L) g \rho S} \ln \left[\frac{T - \mu W}{(T - \mu W) - 1.21 (C_D - \mu C_L) \frac{W}{C_{L_{max}}}} \right]$$

The team decided the maximum take-off weight to be 23.5 pounds, and then take-off velocity was calculated by using formula:

V_{TO}= [2 W/(S ρ 0.8 C_{Lmax})]^{½}

V_{TO} was found to be 55.02 ft/s. By using the above relation take-off distance was estimated to be approximately 279 ft.

4.1.2 Lift calculations

At cruise flight, lift force on aircraft is equal to weight of aircraft. The weight of aircraft was taken to be 26 pounds. At cruise conditions, the velocity of aircraft was taken to be 65.61 ft/s. It was decided to place the wings of aircraft at 4 degree angle of incidence. The team estimated the aircraft C_L to be 0.89.Thus taking air density as 0.0764 lb/ft^3. The required planform area of wings was calculated as follows:

At cruise,

L=W

L= 0.5* ρ * V^2 * S * C_L

L= (½)*(0.0764)*(65.61)^2 * (S)*(0.89)

W= m*g

Thus planform area was calculated to be 735 sq.in.

During take-off required lift is greater than weight. Thus for desired mission performance, the required lift force was estimated to be 150 N. Thus taking the wing area as 735 sq.in and take-off velocity as 59.05 ft/s, the required C_L was calculated using as follows:

$(150*23.73)=(1/2)*(0.0764)*(59.05)^2 * (735/12)*C_L$

The C_L calculated from above equation was found out to be 1.29.Thus the team estimated the takeoff C_L to be 1.5 with the flab displacement of 15 degree.

Engine specifications and static testing:

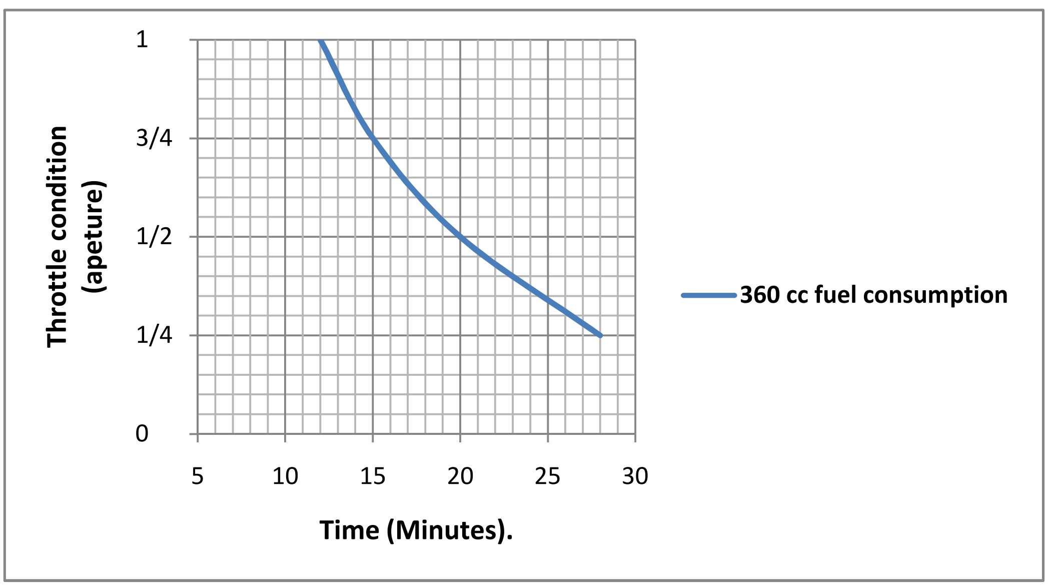
As the max displacement capacity of engine was specified, the team decided to use one which produces more power.
Following are the specification of engine:

Displacement capacity	0.455 cu. in
Bore	0.866
Stroke	0.772
Practical RPM	2000-17000
Power Output	1.63HP @ 16000 rpm

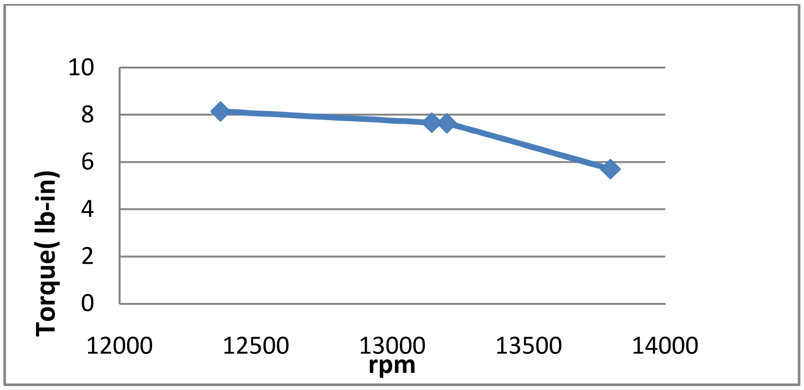
The team tested the engine in various conditions and measured the required parameters. Fuel consumption is an important parameter as it decides the run time of engine. The team drew the following graph based on practical testing:

Fuel: Methanol (20 % castor oil) **Tank capacity**: 12 oz

Needle valve: 1 ½ to 2 turns suitable to throttle condition



The team tested different propellers on engine and noted its max rpm each time. Following chart for max torque was drawn by using max rpm and respective power required for each propeller.



4.1.3 Propeller Selection

Propeller selection was done based on following parameters:

1 .Type   2. Static thrust   3. Dynamic thrust

1. Type: Master Airscrew S2 series propellers were used based on following reasons- **Material**: Fiberglass Reinforced Nylon which is much stronger than wood. **Blades**: Swept-tips with undercambered blades provide more thrust.

2. Static Thrust (S.T.)

Static thrust is the thrust generated by the propeller when aircraft is stationary i.e. when incoming air velocity is zero.
Static thrust was found by theoretical and practical methods:

Theoretical method:

Static thrust values were found theoretically based on formula.

Formula:

$$S.T.= (2.83*10^{-12}) * (RPM^2) * (D^4) * ((\rho*23.936)/29.92) * CF * B.F.$$

$2.83*10^{-12}$ = a constant originated from Reynolds number,
23.936 = metric air density conversion constant
29.92 = constant, standard air pressure : Hg mm

Practical method:

For finding static thrust practically a movable engine mount was made. This mount consisted of a slider on which the engine was mounted and was allowed to slide on low friction wheels over the base. The slider was connected to digital weight measuring hanger. A tachometer was then used to find the maximum RPM of the engine.

Conditions while testing:

Engine: Throttle: Full
Fuel: 360 cc
Propellers: Blades: 2
Needle valve: 1.5-2 turns

Practical and theoretical static thrust values table:

	10x7	**11x6**	**11x7**	**12x5**	**12x6**
Top RPM (practical)	13600	13800	13200	13145	12370
Static thrust pounds(practical)	5.09	7.58	7.04	9.61	8.51
Static thrust pounds(Theoretical)	5.13	7.73	7.07	9.93	8.8

After getting static thrust values, a decision was made to use 11x6, 11x7 and 12x5 propellers for further testing.

3. Dynamic Thrust

Dynamic thrust is an important parameter as the thrust generated by propeller changes with True Air Speed (TAS). As TAS increases the thrust produce by propeller decreases.

By knowing the static thrust values for different propellers, the dynamic thrust values were found using the following equation:

$$Dynamic\ thrust = (-0.00877 * TAS + 1) * (Static\ thrust)$$

By considering the TAS to be in between 0-45 knots, dynamic thrust for different propellers were calculated.

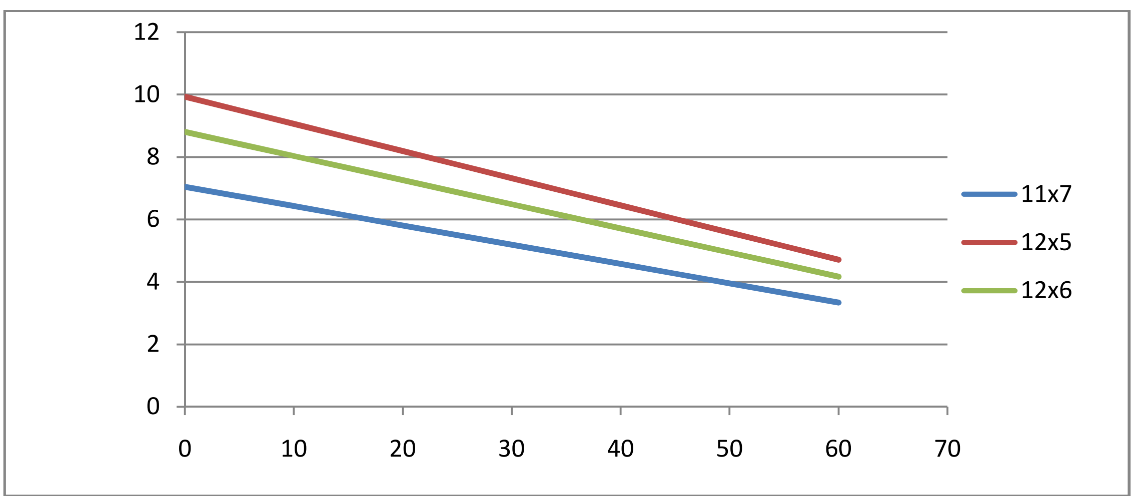


Fig 7: Dynamic Thrust for varying propellers

4.1.4 Drag Reduction and calculations

Reducing drag is a critical factor during the flight of an aircraft. There are two main forms of drag i.e. induced drag and parasitic drag. There is a increase in the induced drag due to formation of wingtip vortices. Winglets encourage the vortices to be shed nearer the wingtips, rather than somewhere else along the span. This produces more lift, since each part of the span contributes lift in proportion to the amount of circulation carried by that part of the span, in accordance with the Kutta-Zhukovsky theorem.

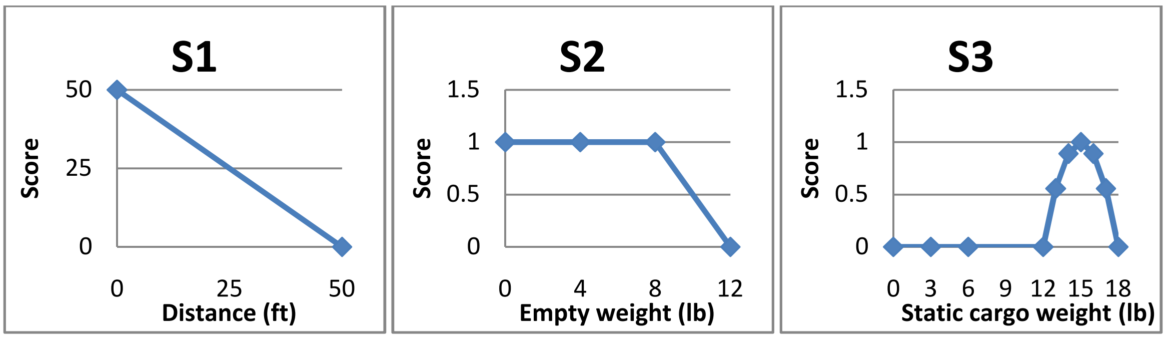
Drag of individual parts was calculated as follows:

Part	***Equations***	***Coefficient of drag (C_D)***
Wing (skin friction)	$C_D = 1.1C_d$	0.011
Wing (induced)	$C_{D,i} = \frac{C_L^2}{\pi e AR}$	0.025
Fuselage	$C_D = C_f f_{LO} f_m \left(\frac{S_{wet}}{S}\right)$ $C_f = \frac{0.455}{[\log(Re)]^{2.58}}$ $f_{LO} = 1 + \frac{60}{\left(\frac{L}{D}\right)^3} + 0.0025\left(\frac{L}{D}\right)$ $f_m = 1 - 0.08M^{1.45}$	0.025
Engine	$C_D = \frac{C_f A}{S_{Ref}}$	0.012
Horizontal tail	$C_D = C_{fht} f_{toht} f_m \left(\frac{S_{wetht}}{S}\right)\left(\frac{C_{dminht}}{0.004}\right)^{0.4}$	0.004
Vertical tail	$C_D = C_{fvt} f_{tovt} f_m \left(\frac{S_{wetvt}}{S}\right)\left(\frac{C_{dminvt}}{0.004}\right)^{0.4}$	0.008
Landing gear (wheel)	$C_D = \sum_{i=1}^{n} C_D \left(\frac{S_{gi}}{s}\right)$	0.004
Landing gear (strut)	$C_D = \sum_{i=1}^{n} C_{D,i} \left(\frac{S_s}{s}\right)$	0.004
Overall Drag		0.094

4.1.5 Competitive Scoring

To achieve a top position in competition, it was important to understand the scoring system and work accordingly. The design report and oral presentation are worth 50 points each and account for 33 % of the total score. The major part of scoring comes from the flight. The flight scoring structure is as follows:

Final score= $\frac{1}{n}\sum_{1}^{n} FS$ Where FS= (4) [S1] [S2] [S3]



Factors affecting flight score

The ideal max final flight score is 200 points. In the above equation S1 depends on how accurately the package can be dropped near the target. To get a non-zero score the team has to drop the package within 50 ft radius. S2 depends on the empty weight of aircraft. To get unit score, the team has to keep weight below 8 pounds. S3 depends on the static cargo weight. The team has to make a aircraft that can lift exact 15 pounds of static cargo to get a unit score.

The S2 and S3 scores depend totally on the designing of the aircraft. The team reduced the weight of the aircraft by- using balsa as the core material for entire aircraft, replacing tail part of fuselage with boom which was strong and light, making rectangular and triangular slots in ply of side walls of fuselage, using balsa strips for making parts like stabilizers, winglets and internal parts. The team also made the static cargo plates in such a way that is added in total to get exact 15 pounds. S1 depends partly on flying and partly on FPV and DAS systems. So the team selected suitable FPV and DAS system which could provide maximum accuracy. To optimize flight score S1, the team made use of sensors which provided an accuracy of 1.5 mm.

4.2 Stability and Control

To avoid mishaps like over-turning, nose up, yaw, rolling and pitching, stable flight is required. Stability of flight depends on many factors. Some factors can be taken care while designing like proper placement of CG and modification in design.

CG Placement

Proper placement of CG is necessary for stability of flight. So CG was first found by theoretical method.

Theoretically CG was found by using following formula:

$$CG = \frac{wing\ MAC}{6} + \frac{3 * Tail\ Moment\ \{(AC\ to\ AC) * \ horizontal\ tail\ area\}}{8 * wing\ area}$$

MAC= Mean Aerodynamic Chord

AC to AC =Distance between Aerodynamic center of wing and horizontal tail.

MAC was found out to be 10.5 in. AC of wing was taken 2.625 in from leading edge as per the thumb rule i.e. it should be at a distance of 25% of MAC from LE. AC of horizontal tail was found by joining LE of tip chords. The AC to AC distance was found out to be 30 in. Wing area was 735 in^2 and tail area was 100 in^2.

CG was found out to be 3.15 in from LE i.e. at 30 % of MAC. By knowing the position of CG a line was drawn at a distance of 3.15 in from LE and parallel to LE on wing. Then the whole plane was balanced on two strings of bench balancer by matching the string tips with the line. Then the components and payloads were placed according to location of CG to balance the plane on the strings.

The CG was calculated theoretically as follows:

Section	Components	X(from nose)	Weight(lb)	Moment(about nose)
Engine	Engine assembly	3	1.08	3.24
	Fuel tank(12 oz)	8	0.75	6
	Throttle servo and receiver	17.5	0.09	1.575
Wing	L&R aileron servos	26	0.16	4.16
	L&R flabs servos	26	0.16	4.16
	Wing and hardware	24	0.98	23.52
Fuselage	Fuselage and hardware	24.5	2.07	50.715
Tail	Rudder servo	42	0.08	3.36
	Elevator servo	44	0.08	3.52
FPV	Camera, Transmitter, Battery	20	0.66	13.2
DAS		18	0.44	7.92
Telemetry	Battery	17	0.15	2.55

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4.3 Servo Forces

The maximum torque that the servos (PZ-15138) can handle is given in the following table:

	4.8 v	6 v
Speed	0.21 sec/60 degree	0.17 sec/60 degree
Torque	3.30 lbs-in	3.73 lbs-in

The above parameters where confirmed by hanging weight to servo arm. Hence the decision was taken to use 4.8v supply for servo. The output torque for throttle was not considered due to load was negligible. The torque for nose gear was also not estimated due to the load similar to rudder as both were driven by same servo.

Servo Force Requirements

	Servo Torque	
Location	Required (lbs-in)	Max torque (lbs-in)
Aileron	1.31	3.30
Elevator	2.31	3.30
Rudder	1.90	3.30
Flaps	1.39	3.30

The formula used to calculate the torque is as follows:

Torque (lbs-in) = 5.31E-7 * (0.54 C^2 V^2 H sin(Z1) tan(Z1) / tan(Z2)]

Servo Selection

Sr. No.	Name	Output Torque (lbs-in)
1.	PZ-15138	3.30
2.	Gotech	2.60
3.	HS-5495 BH	5.55
4.	Futaba S-9254	1.20

As given in the table above, the team decided to use PZ-15138 servo due to its output torque rating and low cost. This particular servo can also rotate to almost 360^0 which was required for fine tuning of the control system.

6. Conclusion

In conclusion, after much research, and testing an aircraft that is capable of lifting 26 pounds of payload was designed. The design applies the principles of fluid and aerodynamics to reduce drag, and increase lift.

Material mechanics were applied to design an extremely light aircraft that will still be able to withstand any load that it experiences during a normal flight. The aircraft was also designed with a tapered wing with forward sweep for good lift distribution. Stability was further improved using winglets

Throughout the design process many challenges were overcome to improve upon previous designs and through new innovations.So our aircraft will be highly competitive and has great promise to place high in the Aero Design West competition.

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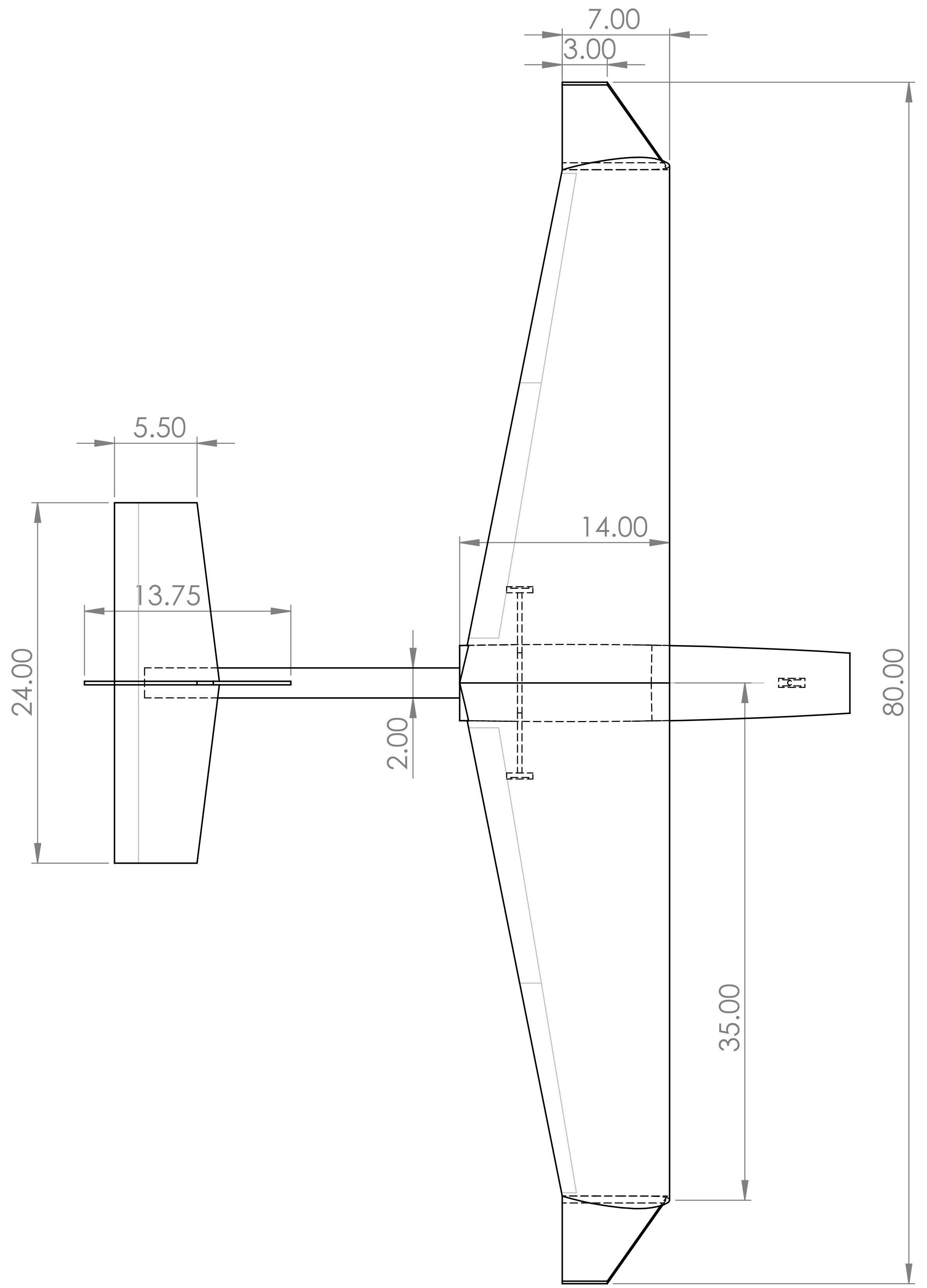

Team Raging Raptors

Rajiv Gandhi Institute of Technology

Team no. 214

Plane Parameters	
Wingspan	80 inches
Empty Weight	5 lbs.
Engine Model	OS Max 46 AX