Optimization of a SAE Aero Design Aircraft

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Abstract - *The Polytechnic University of Puerto Rico (PUPR) has participated in SAE Aero Design for the previous years. SAE Aero Design is an international competition in which an airplane is designed by the students into one academic calendar year. SAE Aero Design have three classes of competition: Regular, Advance, and Micro. In 2018 PUPR participated in Regular Class with an aircraft called ORCA. The objective of Regular Class is designing an aircraft able to carry as much payload as possible fulfilling every requirement and limitations. ORCA was a good design, but the aircraft do not have the capacity to carry the payload predicted in the conceptual design. This project is based on the optimization of ORCA. This new design is bigger, lighter, and can carry the payload predicted fulfilling every requirement and limitations. The design method for the aircraft was based on a full weight analysis from various aircrafts designed for the same purpose, including ORCA.*

Introducction

The design of the aircraft started from a weight analysis. This weight analysis was performed to the determination of weight fractions for every component for the aircraft. These weight fractions and assumptions like V_{stall}=30 ft/s, W/S=2.5, and AR=7 were necessary to make the initial sizing. After a few iterations considering aerodynamics and performance the aircraft was designed.

The Structure configuration of ORCA was very strong but at the same time it was heavy; this is due to the 2 g's used. For this new design just 1.2 g's and a safety factor of 5% was enough for the structure design to make it lighter. Also, other improvements for better performance and agility were the elimination of cabin bay, lighter wing and tail attachments, more wing area reducing wing loading, and using airfoil in vertical tail instead of flat plate. These modifications were the key for the optimization. For better maneuvering, bigger control surfaces were applied with an increment in static margin to guaranties the stability.

Aircraft's structure it was completely designed in Basswood and Balsawood even the wheels were in basswood. This kind of woods were selected to keep it lighter. Payload consist in 20 tennis balls and 20 metal plates making a total weight of 13lbs. which is around 66% of the Takeoff weight. The aircraft takeoff weight is 20lbs. with an empty weight below 7lbs. and just 140 feet of runway is enough to takeoff.

Design Development

The most important for the design of these aircrafts is the payload in terms of weight and space. The payload in Regular Class consist in tennis balls as passengers and metal plates as luggage. Each passenger must have their luggage and must be at a spacing no more than 0.25" in a continuous geometric plane. The conceptual design of the 2018 PUPR SAE Aero Design was for a total of 20 passengers; that's why the optimization is based on the same total of passengers. After this project the design method could be apply for any amount of payload.

The payload configuration for the aircraft can be seen in Figure 1, also shows the minimum space needed for fuselage. A good arrangement and selection of payload plates it was that the luggage itself is the passenger seat. Thanks to the space needed for the payload the dimensions of the fuselage can be established, at least to determinate the cross-section.

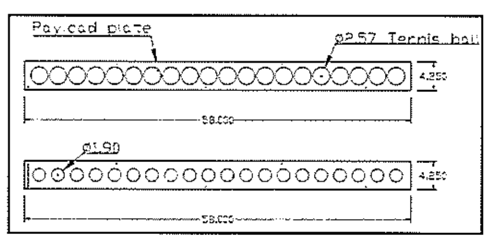

Figure 1 - Payload Configuration

Each tennis ball has a weight of 0.126 pounds and the metal plate must be between 0.5-0.75 pound. For 20 passengers and selecting 0.5lb for each luggage makes a total payload of 13 pounds, considering hardware to be secure at the airframe.

Empty Weight Fraction

Determination of empty weight fraction (W_e/W_0) a weight analysis of every component for this type of aircrafts was performed. Table 1 shows weight fractions for every component after studying three aircraft for the same mission.

Component	W_i/W_0
Tail	0.0255
Propulsion	0.0253
Wing	0.0740
Fuselage	0.1724
Electronics	0.0535

Table 1 - Average Weight Fraction

Component	W_i/W_0	W_0 guess		
		19 lbs.	20 lbs.	21 lbs.
Tail	0.0255	0.4851	0.5106	0.5361
Propulsion	0.0253	0.4816	0.5069	0.5323
Wing	0.0740	1.4065	1.4805	1.5546
Fuselage	0.1724	3.2752	3.4476	3.6200
Electronics	0.0535	1.0169	1.0704	1.1239
	W_e=	6.67	7.02	7.37
	Payload=	12.33	12.98	13.63

Table 2 - Takeoff Gross Weight

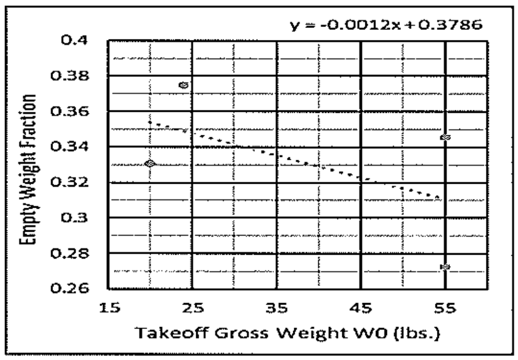

Figure 2 - Empty Weight Fraction Trends

as wing dimensions. Figure 3 shows the wing geometry and the ailerons control surface. Control surfaces like ailerons, elevator, and rudder were sized by experience too. The ailerons tipically extend from about 50% to 90% of the span [2]. In this case it was used 40% of span and 32% of chord, can be seen in Figure 3.

The diameter of a tennis ball is 2.57", keeping spacing of 0.25" between each other, and considering nose and tail the fuselage total length is about 68 inches. Remembering the payload configuration the cross-section of fuselage should be at least 6" of height and 5.5" of width.

Assuming a takeoff gross weight (W_0) and remembering the total payload (13 lbs.) a few iterations later the W0 was calculated. These weight fractions were used as a start. The final weight fractions are obtained later when the aircraft geometric and the structure are finished. For now, a great idea of weight for each component is obtained. But, an empty weight (We) around 7 pounds and 20 pounds of takeoff gross weight (W0) is secure. With these weights the W_e/W_0 can't be more than 0.35. Example of this calculation is shown in Table 2. As can be see a W_0=20 pounds match with the total payload of 13 pounds.

Also, these values can corroborate in Figure 2 which shows the empty weight fraction trends for this kind of aircrafts. This plot was considering the same three aircraft analyzed previously in the weight analysis and including the final W_e/W_0 of the aircraft.

Aircraft Geometry

The airfoil selected is S1223, considering velocity of takeoff ($V_{T.O.}$=36 ft/s) and the fact that the aircraft will be flying at sea level with a chord assumed of 12". a Reynolds number around 230K is calculated. It is very important to note that the Reynolds number is below critical Reynolds number Re_{cr}=500k [1]. This means that the aircraft will fly in laminar flow. The CL_{max} is close to 1.8 considering that a straight wing is going to be used; a wing loading (W/S) around 2.74 is obtained. Knowing that a 20 lbs. is the total lift a wing area of 1051 in^2 is needed. Remembering AR=7 assumed, a wing span (b) of 86" and a chord (c) of 12.5" are going to be used

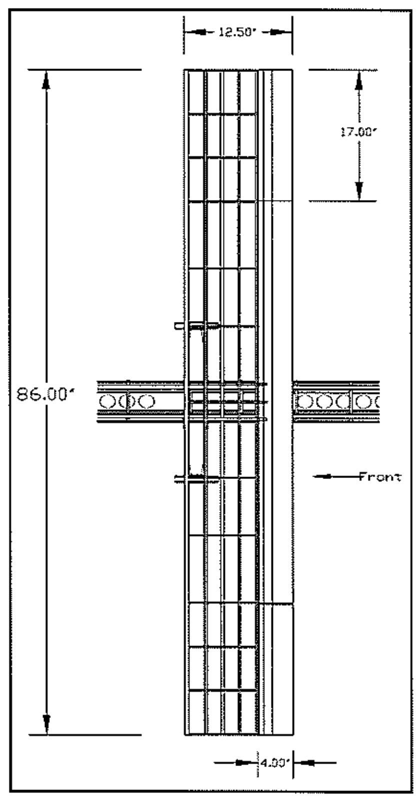

Figure 3 - Wing Geometry

For an aircraft with a front-mounted propeller engine, the tail arm is about 60% of the fuselage length [2]. However, a tail arm about 50% of the fuselage was used in this aircraft. The tail geometry was calculated using Tail Volume Coefficient. The coefficients used are based by experience. The horizontal tail coefficient (C_{HT}) is 0.6 and vertical tail coefficient (C_{VT}) is 0.055. Using these coefficieent, a tail arm of 30", and the wing geometry the areas calculated for the tail are as follows: Horizontal Tail Area (S_{HT}=268.75in^2) and Vertical Tail Area (SVT=169.49in2). The tail airfoil should be selected considering AR to ensuring that the tail never stall or if the tail stall must be later than the wing stall. Tail airfoil is NACA0010 with an Aspect Ratio (AR) for horizontal and vertical of 3.15 and 2, respectively. Figure 4 and Figure 5 shows the horizontal and vertical tail geometry. The elevator and rudder used is about 43.2% and 40% respectively, can be seen in Figure 4 and Figure 5.

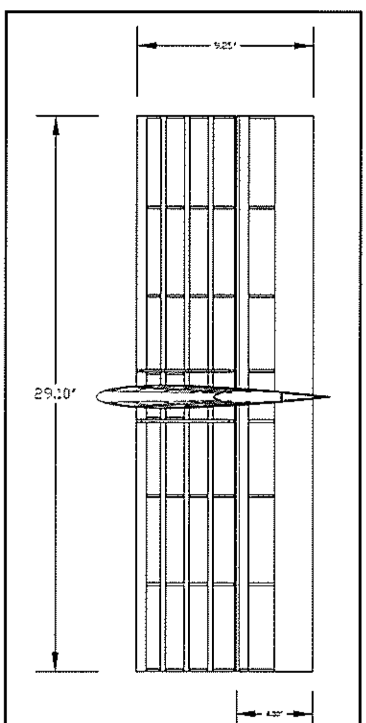
Figure 4 - Horizontal Tail Geometry

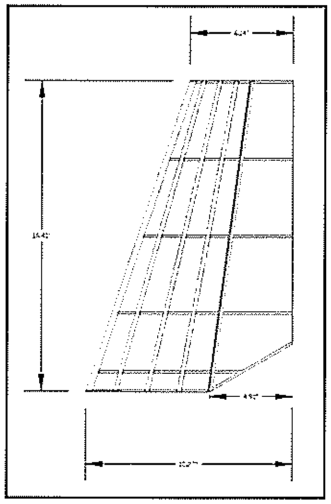
Figura 5 - Vertical Tail Geometry

Structure

The structure should be designed keeping in mind the weights calculated previously. Finding the limits in design this aircraft was designed using 1.2 g-force and a safety factor of 5%. The safety factor and g-force were selected by experiences and by the kind of maneuvers the mission requires. Using these assumptions the ultimate bending moment in wing and tail are 265lb.in. and 27lb.in., respectively. These assumptions will help to keep the aircraft lighter. The structure configuration for wing and tail are as shown in Figure 6 and Figure 7.

Wing and Tail structure configuration are based in just stringers as main support. Spars were discarded to keep it lighter. Taking advantage of the stringers for the skin (monokote) to keep the airfoil shape. Other good point of stringers is that they are as far away as possible at upper and lower surfaces obtaining great inertia. The ribs spacing was determined by experience keeping in mind factors like skin and slenderness of stringer itself to avoid buckling. Spacing used is around 8".

The distribution of stress due to bending is not going to be obtained because the areas of the stringers are very small. These stresses were considered as axial stresses in each of the stringers. Example of these calculations is presented in Table 3 and Table 4. These tables represent the axial forces and stresses generated in every stringer.

The same idea was used to design the fuselage. The Fuselage is composed of longerons and a kind of T-Beam at the bottom of the cross-section. Figure 8 shows the fuselage structure configuration.

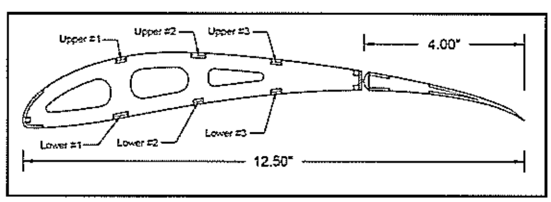

Figure 6 - Wing Structure Configuration

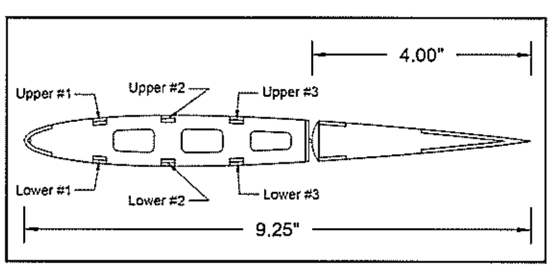

Figure 7 - Horizontal Tail Structure Configuration

The ultimate bending moment in fuselage is 130lb.in, considering the same safety factor from wing and tail. The bending effect in fuselage was analyzed statically taking the landing gears as supports. Table 5 shows the forces and stresses generated in every component of the fuselage.

	i	Ai	Ix	Pi	Pi * yi	Stress (psi)
Upper	1	0.0313	0.0092	55.5	30.0	1776
	2	0.0469	0.0203	101.0	66.3	2155
	3	0.0313	0.0077	50.7	25.1	1622
Lower	1	0.0469	0.0338	-130.5	110.8	-2785
	2	0.0313	0.0078	-51.1	25.5	-1635
	3	0.0313	0.0020	-25.5	6.4	-817
	Σ=	0.2188	0.0808	Σ=	264.1	

Table 3 - Wing Stringers Stresses

	i	Ai	Ix	Pi	Pi * yi	Stress (psi)
Upper	1	0.0313	0.0038	11.0	3.82	351
	2	0.0313	0.0050	12.6	5.03	403
	3	0.0313	0.0045	11.9	4.52	382
Lower	1	0.0313	0.0038	-11.0	3.82	-351
	2	0.0313	0.0050	-12.6	5.03	-403
	3	0.0313	0.0045	-11.9	4.52	-382
	Σ=	0.1875	0.0268	Σ=	26.75	

Table 4 - Horizontal Tail Stringers Stresses

i	Ai	Ix	Pi	Pi * yi	Stress (psi)
1	0.0938	1.5164	9.9	39.85	106
2	0.0625	0.2956	3.6	7.76	57
1	0.0938	1.5164	9.9	39.85	106
2	0.0625	0.2956	3.6	7.76	57
left	0.2500	0.1441	-3.2	1.60	-13
right	0.2500	0.1441	-3.2	1.60	-13
bottom	0.5000	1.2097	-20.5	31.87	-41
Σ=	1.3125	5.1219	Σ=	130.29	

Table 5 - Fuselage Components Stresses

Components	Weight (lbs.)	Arm (in.)	Moment (lbs. in.)
Tail	0.71875	61.00	43.844
Main Gear	0.7375	31.75	23.416
Nose Gear	0.3	-3.50	-1.050
Engine + Prop	0.66	-7.25	-4.785
Wing	1.25	32.50	40.625
Battery	1.02	42.00	42.840
ESC	0.097003	37.00	3.589
Receiver	0.000661	37.00	0.024
Payload	13	29.00	377.000
Fuselage	2.14	24.00	51.360
Power limiter	0.0375	36.00	1.350
Σ =	19.96	Σ =	578.213
	X cg (in.) =	28.97	
	X cg %MAC =	17.73	

Table 6 - Center of Gravity

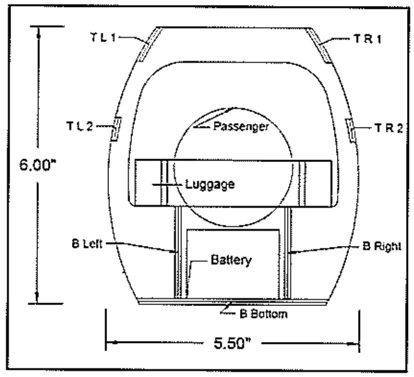

Figure 8 - Fuselage Structure Configuration

Weights and Center of Gravity

While the manufacture processes the weights of every component of the aircraft was monitored to ensure We. The center of gravity (CG) was calculated using the first Bulkhead as datum.

Table 6 shows a summary of CG determination. The CG calculation helps to locate the wing to obtain the CG as desired. A good advice is to weight every element before manufacturing starts and think about glue and skin (monokote). The final weight fraction of every component of this project is presented in Table 7. These weight fractions are very useful to design aircraft with a similar mission.

Aerodynamics

The lift-curve slope is the behavior between lift and angle of attack (AOA) of the aircraft. Theoretically the maximum of this slope is 2π [3]. For this project the slope is $4.9428=1.57\pi$. Figure 9 can see this slope and the maximum lift coefficient (CL_{max}) about 1.8 at AOA=15. On the other hand, drag polar presents the behavior between lift and drag of the aircraft. The tangent of the drag polar curve represents the maximum lift to drag ratio (L/D_{max}). As could dee in Figure 10 the L/D_{max}=14.9, but this is at AOA=1. To generate the lift needed for the aircraft should be flying around AOA=7. At this point the L/D ratio is about 12.5. The drag at zero lift angle is around 0.03 this value is due to the wetted area and considering that the aircraft have fixed landing gear. Lift-curve slope and drag polar are very important to determinate aircraft performance.

Components	$(W_i/W_0) * 100$
Tail	3.601
Wing	6.262
Fuselage	10.721
Main Gear	3.695
Nose Gear	1.503
Propulsion	3.306
Battery	5.110
Speed controller	0.486
Receiver	0.003
Power limiter	0.188
Payload	65.126
Σ =	100

Table 7 - Real Weight Fractions

Results and Discussions

The total dimensions of the aircraft are a wingspan of 86", a length of 77.40", and a height of 31.82". The final aircraft design has a W/S=2.67 with a T/W=0.38 and a takeoff weight around 20 pounds. A static margin of 21% of MAC it was necessary to guaranties the stability of the aircraft [4]. With this properties and specifica-

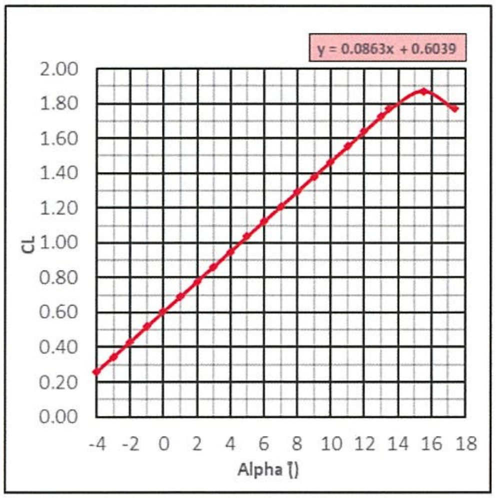

Figure 9 - Lift-Curve Slope

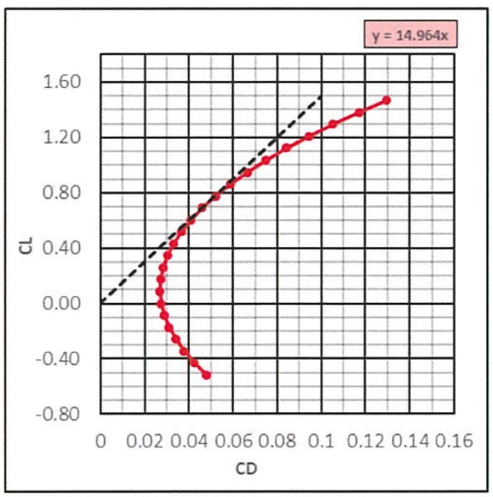

Figure 10 - Drag Polar

tions got a take-off roll distance of 140 feet.

The V_{stall} is 30ft/s at AOA=15 degrees and the L/D_{max} is around 14.9 at AOA=1 degree. But at the operating flight conditions L/D is around 12.5, close to AOA=7 degrees.

A W_e/W_0=0.348 which means that more than 65% of the aircraft is payload. Within that We/W0=0.348, just 16.47% of the aircraft is structure. After manufacturing the aircraft, valuable details to consider when designing this kind of aircraft is that the glue is about 7.5% weight increment in each structural component. In addition, the skin (monokote) is about 12.5% weight gain as well. Figure 11 demonstrates the weight distribution of the aircraft.

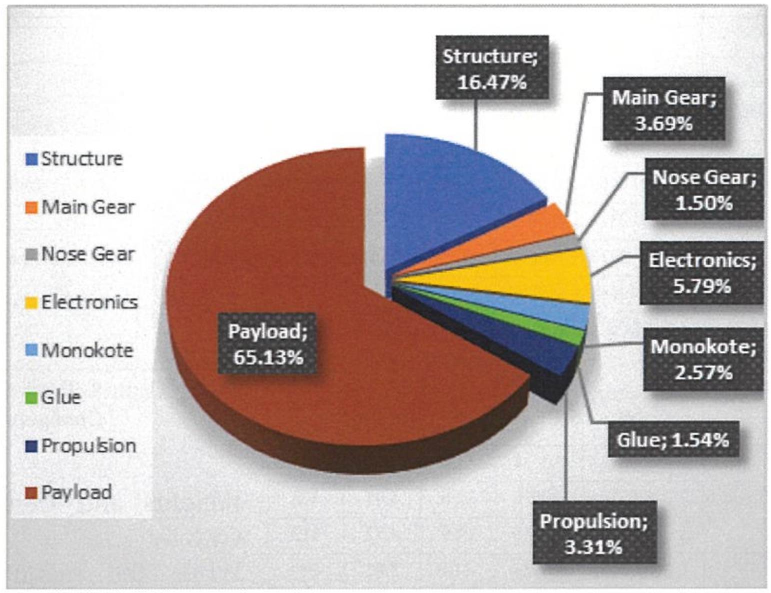

Figure 11 - Aircraft Weight Distribution

Conclusions

The objective of design an aircraft with capacity to carry the payload predicted in the conceptual design was successfully performed and manufactured. The aircraft designed in this project meets all the requirements and limitations for Regular Class in 2018 Collegiate Design Series SAE Aero Design Rules. After this project, important factors to design this kind of aircraft is that the empty weight fraction should be between 0.25 and 0.34 and the thrust to weight ratio must be at least 0.40. The empty weight fraction trends presented in Figure 2 is very important to consider at the moment to design a cargo aircraft.

The weight fractions method used to design this aircraft could be used to develop aircrafts whose mission is carry as much payload as possible within their requirements and limitations. Also, could be used for future PUPR participation in SAE Aero Design.

Future works

The structure design was one of the most studied areas in the project, so much so that the design is almost at the limit. Because this competition is very extreme a deeper analysis of structure is recommended to make it stronger "especially in the wing" keeping the same weights. This will be very helpful to flight in aggressive weather conditions; as it is where these competitions are held.

Use of telemetry to get info about the flight to corroborate the data assumed from the conceptual design. Also, implementation of sensors could be very useful to validate the aircraft's aerodynamics and performance.

Acknowledgement

This project is dedicated to José R. Pertierra who taught me and awakened my passion for aerospace. Also, special thanks to all the professors who were part of my engineering education.

References

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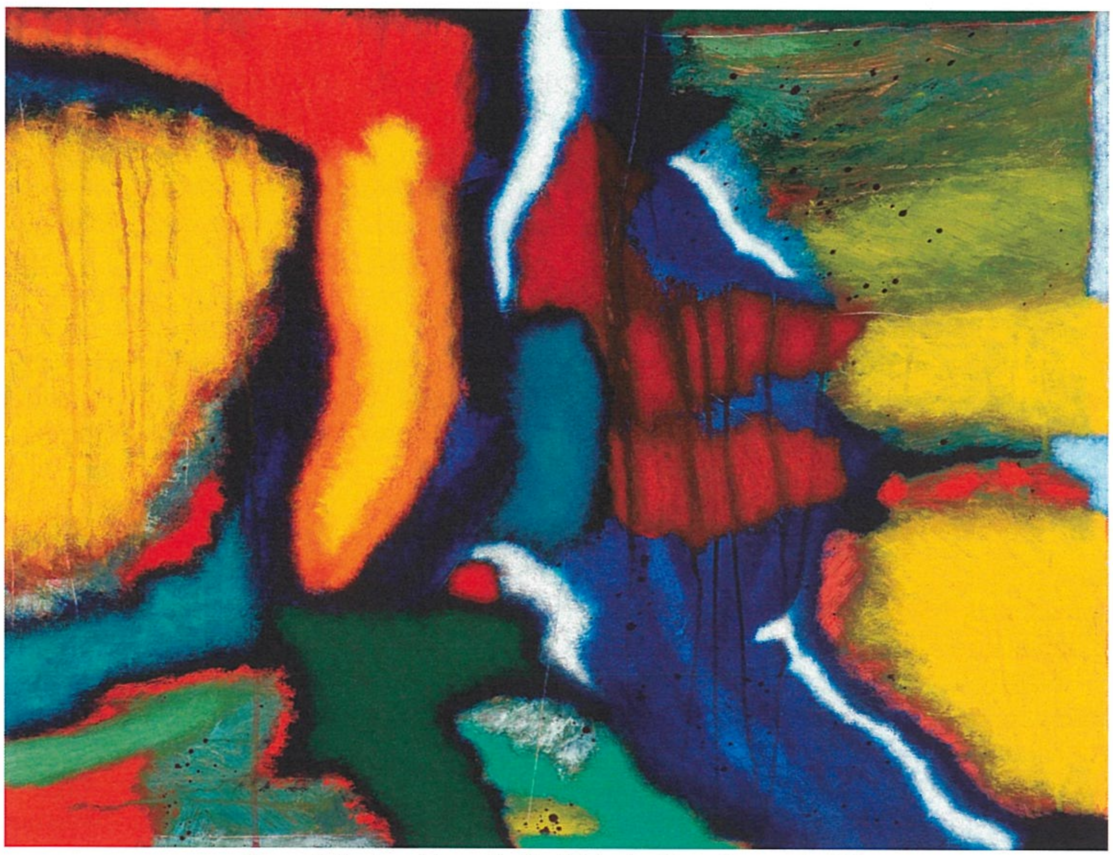

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